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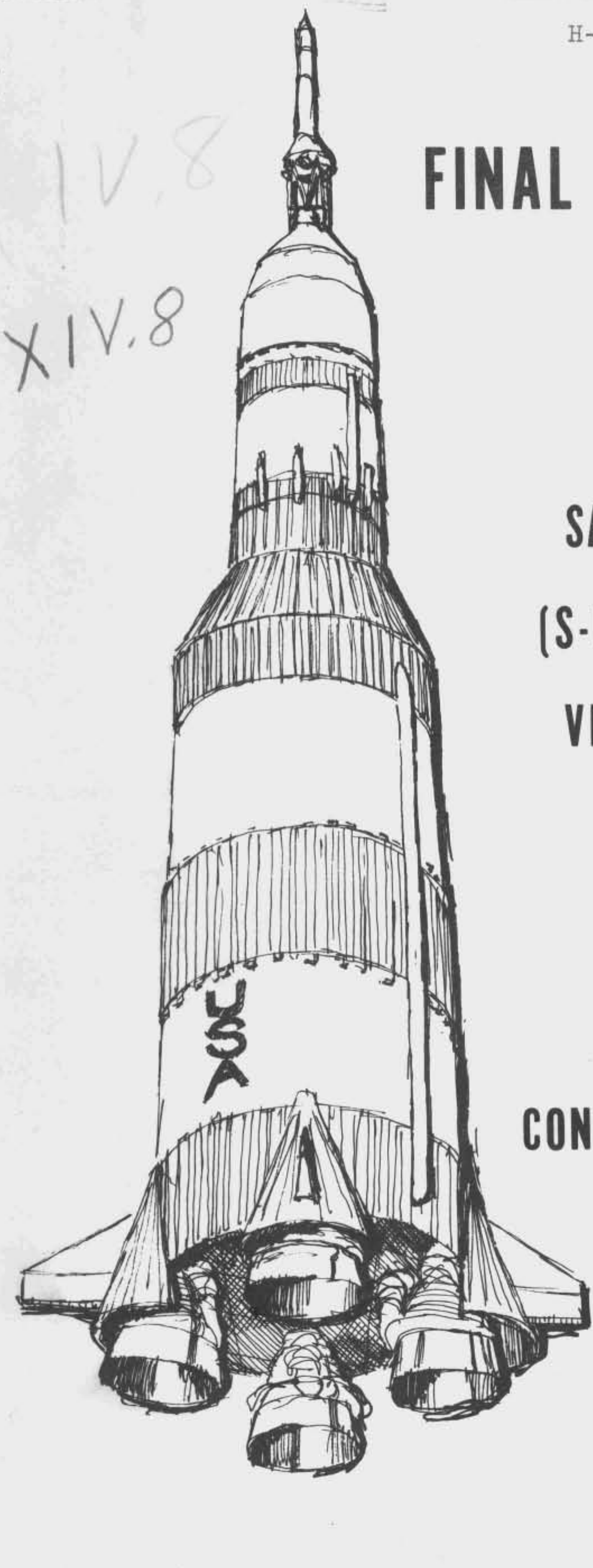
VOLUME I

H-52 Mr. Akens A&TS-MS-H

FINAL TECHNICAL REPORT



SATURN V DERIVATIVE (S-IC/S-IVB/I.U.) LAUNCH VEHICLE SYSTEM STUDY

SEPTEMBER 15, 1969

CONTRACT NO. NAS 8-30506

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THE BOEING COMPANY SPACE DIVISION LAUNCH SYSTEMS BRANCH

REVISIONS

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ABSTRACT

This document contains the results of a study to define in detail a Saturn V derivative (S-IC/S-IVB/I.U.) launch vehicle and to determine its implementation and production costs and schedules. The S-IC/S-IVB/I.U., or INT-20, has payload capabilities in the intermediate range between Saturn IB and Saturn V. The study was conducted under NASA/MSFC Contract NAS8-30506. Phase I of the study included parametric technical and resources analyses that lead to the selection of a 4 F-1 S-IC/S-IVB/I.U. baseline configuration for detailed analysis and preliminary design. In Phase II, design criteria were prepared that identified baseline vehicle weights, aerodynamics, loads, controls and flight environment characteristics.

Design studies were done to ascertain the capability of the existing Saturn V components and hardware to meet the new criteria. A preliminary design was delineated for each stage and the Instrument Unit. Performance data were prepared for MLV and Big Gemini payload shapes, for the use of a J-2S engine on the S-IVB, and for the use of Centaur and Service Module injection stages. A Phase III resources analysis detailed the Design, Development, Test, and Evaluation (DDT&E) Plan for INT-20 implementation and production. Both retrofit of existing Saturn V hardware and new production (in-line) implementation were considered. Data were included for changes and additions to Launch Complex 39 of Kennedy Space Center. The study concluded that the 4 F-1 INT-20 had wide application for both manned and unmanned missions and had a very small development cost.

Data to supplement this document are presented in the following documents:

D5-17009-1	Executive Summary
D5-17009-2, Vol. II	Appendices

LIST OF KEY WORDS

Intermediate Vehicle
INT-20
4 F-1 S-IC/S-IVB/IU
Saturn V Derivatives
MLV payload shape
Big G (Gemini) Spacecraft
Apollo Spacecraft
Retrofit/INT-20
INT-20/MLV
INT-20/Apollo
INT-20/Big "G"
J-2S/INT-20
Earth Orbit Logistics
Lunar Logistics
Interplanetary Probes
Synchronous Orbit Communication & Navigation
Intermediate Payload Range

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TABLE 4.2.5.3-II. COMPARISON OF INT-20 INTERFACE LOADS WITH PRESENT SATURN V IU CAPABILITY

INT-20 STATION 2245 (IU/S-IVB INTERFACE)

Load or Flight Condition	INT-20		Current Capability		Factor of Safety		Required Factor of Safety	
	N_c* (Lb/In)	N_t* (Lb/In)	N_c (Lb/In)	N_t (Lb/In)	N_c	N_t	N_c	N_t
95% Ground Wind for Access Door Removal and Installation	183	0	224	N/A	1.22	N/A	1.0	1.0
99.9% Ground Wind with Access Door Removed	208	0	324	N/A	1.56	N/A	1.4	1.4
Max Q Alpha	1000	148	1435	397**	1.43	2.68	1.4	1.1
End of Boost	780	0	1400	N/A	1.80	N/A	1.4	N/A

NOTE: (*) N_c is the computed interface running compression load (unfactored)
N_t is the computed interface running tension load (unfactored)

(**) Capability to yield of Interface Channel at Bolt Holes

4.2.5.3 (Continued)

111,600 maximum. Since Max Q Alpha condition is designing the baseline INT-20 IU structure, the g's and the bending moment are both lower than the present Saturn V vehicle, permitting the accommodation of the higher payload. IU structure static tests using an S-IVB forward skirt and an aft SLA skirt have demonstrated the capability of the IU structure to 1435 lbs/in. The Saturn V IU structure is therefore considered qualified to the maximum loads required for INT-20. It must be emphasized that the proper design of the payload structure is necessary to assure that the IU structure qualification remains. For example, shell and interface dimensional stability must remain as good as the present SLA structure.

The maximum acoustic environment for the four F-1 engine INT-20 vehicle at Station 2245 is presented in Table 4.2.5.3-III. These INT-20 levels for the launch and inflight environments are within the levels measured during Saturn V flights (AS-501, 502 and 503) and should present no problems to the INT-20 IU structure and components.

The IU structure and component vibration is a function of the four F-1 engine generated acoustic pressures during the launch period of flight and of the aerodynamic pressures created by boundary layer fluctuations in-flight. Since the maximum INT-20 acoustic and dynamic pressures (Table 4.2.5.3-III) are less than those measured on the Saturn V (S-IU-501, 502 and 503), the corresponding structure and component vibration can be assumed to be within the qualification levels for the Saturn V IU (Reference 3.1.2.6-1).

Of highest concern on the Saturn V vehicle was the ST-124 Guidance Platform (Location 21). Satisfactory performance is expected in the INT-20 imposed environment. Vibration damping compound applied to the IU structure in the ST-124 area of the Saturn V IU's has been successful in attenuating the vibratory amplitudes during test and flight conditions. The vibration damping compound will also be used for the INT-20 IU.

The INT-20 acceleration requirement is less than the S-IU-502 flight accelerations and should not impose any problems to the structure or components. It is therefore assumed that the imposed acceleration levels should require no additional qualification testing over that presently existing for the Saturn IU.

TABLE 4.2.5.3-III. DYNAMIC ENVIRONMENT COMPARISONS

Item	4F-1 INT-20	Saturn V IU Flight Data
Liftoff OASPL (dB)		
Specification	153.5	154.0*
Environment	150.5	151.9
MAX Inflight OASPL (dB)		
Specification	155.0	158.0*
Environment	152.0	156.9
Acceleration (G's)		
Max Q	1.86	2.0
SIC End Boost	4.68	4.8
Dynamic Pressure (Lb/Ft^2)		
Max Inflight	728.5	783.0

* IN-P&VE-S-63-2

4.2.5.3 (Continued)

The combination of vibration and acceleration loads at Max Q Alpha period of flight are equal to or less than the current Saturn V IU loads and present no problem for further consideration in this study.

(c) Conclusions of Study

A review of the loads and vibration and acoustic data presented indicate there is no concern for the adequacy of the current Saturn V IU configuration for the INT-20 vehicle. The Saturn V structure has been subjected to load levels in excess of the INT-20 requirements of this study.

It should be noted that some effects which could significantly affect IU loads were not taken into consideration at this level of study. Each must be addressed in the final loads definition. These include the following:

Any localized loading effects at the interfaces from adjacent stages.

Engine-out tension capability during boost which is required on the Saturn V vehicle to allow safe abort before vehicle break-up.

Shock load at S-IC/S-IVB Separation as higher tension loads would be expected since the IU is closer to the S-IC/S-IVB interface.

Bending Moment at End Boost should be established for future evaluation.

Any significant rearrangement or addition of components which may be required as compared to Saturn V IU's.

It was assumed that the INT-20 IU temperature environment is no more severe than present Saturn V.

6. Emergency Detection System (EDS)

(a) Requirements

Reduced requirements are:

S-IC Engine No. 5 monitoring not required.
S-II Engine monitoring not required.

4.2.5.3 (Continued)

(b) Implementation

The EDS in the IU has been studied for impact resulting from the reduced requirements. The conclusion of study is that no change is required.

The S-IC Engine No. 5 IU monitoring is energized by the stage circuitry only in the event of an engine out (thrust not OK). Therefore, the removal of this stage circuitry will present an open circuitry will present an open circuit to the IU and look like thrust OK to EDS. This means the EDS will essentially ignore the absence of the engine and it will not treat it as an engine out.

The absence of the S-II stage is handled with sequencing and the EDS simply will not monitor the S-II stage.

An open item that requires further investigation is the Rate Switch Settings of 4^{o}/sec for Pitch and Yaw during first stage burn and 9.2^{o}/sec thereafter.

7. Sequencing

(a) Requirements

S-II stage sequencing not required.

Isolate the S-IC outboard engine out discrete input to an opposed pair of engines.

(b) Implementation

The removed S-II stage removes all of its sequencing requirements, however, this only means removal of the S-II associated time base TB3 and a few S-II functions done in TB2 and 4. The wiring associated with S-II switch selector, discrete inputs, and interrupts can be left spare with no ill effect on IU performance.

As discussed in paragraph 4.2.5.3 b.1.(b) (2) with one engine out on the S-IC stage it is necessary to isolate to an opposed pair of engines so that first cutoff of two S-IC stage engines will include the engine already out. Without this feature there would be a 50/50 chance of having one burning engine after first cutoff. The isolation is done by taking existing engine cutoff indicators and pairing the two from

4.2.5.3 (Continued)

opposed engines. A spare LVDA discrete input is used and only a slight wiring change is necessary. This scheme also has the advantage of being usable on Saturn V missions with only a slight software modification. Figure 4.2.5.3-1 illustrates the modification which takes place entirely within the IU.

8. Electrical Support Equipment

The purpose of the Electrical Support Equipment/Ground Support Equipment is the definition of ESE/GSE modifications required by INT-20 vehicle configuration.

(a) Component Acceptance Test

The component acceptance test ESE will require no modifications to acceptance of the components to be implemented on the INT-20 vehicle.

(b) System Test

The system test ESE modifications are discussed below according to IU subsystems.

(1) Instrumentation and Communications (I&C)

Choice of the Saturn V CCS as the INT-20 command will require no impact to present S-V ESE.

(2) Guidance and Control

The Flight Control Computer will require modification in order to meet the additional requirements of the INT-20 configuration. The basic requirements are:

FOUR S-IC Switch Points.

NO S-II Stage.

Elimination or modification of unused S-II hardware.

The IU networks provide the FCC interface with nine switch points. The first six are presently used and the last three are terminated

SAT V - S-IC OUTBOARD ENGINE OUT (DI-14)

INT-20 - S-IC ENGINE NO. 1 OR NO. 3 OUT (DI-14)

- S-IC ENGINE NO. 2 OR NO. 4 OUT (DIS-4)

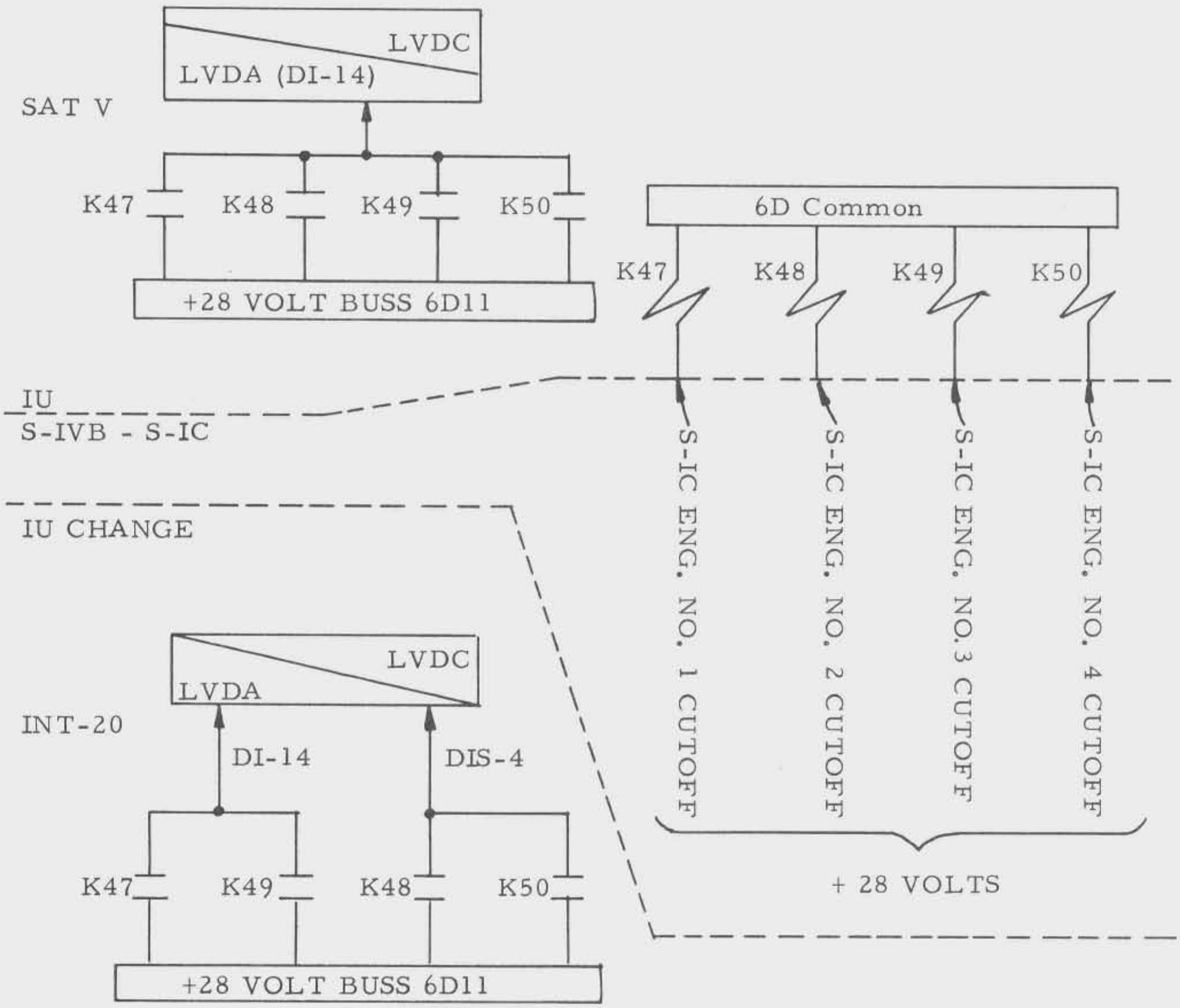


FIGURE 4.2.5.3-1. OUTBOARD ENGINE OUT ISOLATION

4.2.5.3 (Continued)

at the FCC interface. Therefore, two of these will be routed to the S-IC filters. This will require four wires to be added to the FCC cable harness and Motherboards 6 and 7 to be redesigned.

Modifications to the FCC will not require additional ESE, however, procedures changes will be discussed in ESE/GSE software.

(3) Networks

Minimum modifications of unused S-II functions can be handled by changes to electrical network design which encompasses interconnecting cabling within the IU including interstage interface wiring and switch selector functions.

Modifications such as necessary to accomplish INT-20 configuration are not unusual in the fabrication assembly and checkout of IU's on the current Saturn V Program.

(4) Software

The automatic checkout program modifications resulting from INT-20 configuration affect the subsystem Automated Checkout Programs and the IU Overall Checkout Program.

Subsystem Automated Checkout Programs

The subsystem automated checkout programs used to check out the subsystems of AS-505 will require modifications/deletions in areas specified below.

Control Subsystem.

A_1 Gain.

A_0 Gain.

Control System Nulls.

Engine Deflection.

Control Computer Comparators.

Control Computer Relay Redundancy.

4.2.5.3 (Continued)

Electrical Subsystem.

Power Distribution and Control.

General Networks.

Simulated Plug Drop.

IU Overall Checkout Program

The IU Overall Checkout Program is called the Vehicle Test Program (VTP) at IBM and is a general program applicable to all vehicles that provides on-site capability to vary program parameters and reduces the number of program deliveries. The objectives of the VTP are to:

Provide early test program availability.

Eliminate program problems due to change activity.

Reduce effort expended to debug interim programs.

Provide additional test flexibility.

Provide capability to sequence the vehicle through a simulated plus-time and dynamically test all LVDA interfaces.

Provide the test engineer with a means of making a quick look evaluation of any test run, while it also provides vehicle checkout capabilities equivalent to those in the simulated flight mode of the Flight Program, uses the current LVDC Preflight Program, and minimizes impact on the Ground Checkout Computer System (GCCS) programs.

The VTP performs the following operations:

Sequencing.

Time Base initiation and maintenance.

Vehicle Discrete Inputs.

4.2.5.3 (Continued)

Discrete Outputs.

Switch Selectors.

Normal Functions.

Special Functions.

Alternate Functions.

Platform Interface Testing.

Accelerometers.

Earth's Rate Drift Test.

Delta Count Test.

Gimbal Angles.

Earth's Rate Drift Test.

Disagreement Bit Test.

Zero Test.

Reasonableneness Test.

Attitude Error Processing.

Standard Routines.

Mission Dependent Routines.

Functions that affect other operations, i.e., sequencing switch selectors.

Functions that do not affect other operations performed, i.e., maneuver inhibits and navigation updates.

4.2.5.3 (Continued)

CIU Monitoring.

Continuous station gain assumed.

No compressed data storage over Preflight Program.

Real-Time Telemetry.

TLC Processing.

No attempt to recover and continue.

Immediate test termination.

Telemetry.

Same as Flight Program for operations similarly performed.

Special selected sequence of items.

These functions are performed by the VTP in such a manner that mission and vehicle dependent parameters can be easily changed with no impact to the basic program. The mission and vehicle dependent parameters can be loaded into the VTP via user controlled data tables. The user controlled data tables for the VTP are:

Switch Selector Table.

Ladder Profile Table.

CIU Address Table.

DCS Allow Table.

Telemetry Table.

Vehicle Dependent Parameter Table.

(Reasonableness constants test-site latitude, time back-up times, etc.)

4.2.5.3 (Continued)

The overall test requirements for INT-20 vehicle are such that basic program can be modified by using the user controlled data tables and create no impact on the VTP.

(c) KSC Operations

The KSC GSE requirements are essentially the same as those at Huntsville. Therefore, the same modifications will be required at KSC. However, the GSE at KSC must be modified by the various support contractors that control the GSE. Q-Ball requirements will be deleted from KSC operations.

The IU hardware modifications will likewise necessitate modification to the automated subsystem checkout procedures at KSC. The checkout programs to be modified are listed below with the missions that require the change. Physical GSE modification to LC-39 to accommodate the INT-20 configuration are not discussed in this section.

Electrical Subsystems.

Launch Vehicle Operations for Space Vehicle Overall Test #1 (Plugs In), V-20010.

Launch Vehicle Operations for Space Vehicle Overall Test #2 (Plugs Out), V-20012.

Q-Ball Checkout Procedure, V-27068, V-27156.

Switch Selector Functional Verification, V-21107.

Power Distribution and Control Switching Test, V-21263.

Control Subsystem.

Flight Control Computer Comparator Test, V-23169.

Flight Control Computer Redundancy Test, V-23171.

FCC Systems Gain Test, V-23176.

4.2.5.3 (Continued)

As discussed in the section on Huntsville test software, the IU overall test program requires no modifications to the basic program.

Development

No new test requirements must be developed for the INT-20 vehicle IU's. Existing test specification can be used with some modifications to reflect the hardware design changes. The specification modifications will result in less than 10 percent change in the existing specification documents for the INT-20 missions. The test specifications will be determined and released once the hardware design changes are released.

4.3 ASSOCIATED INVESTIGATIONS

Several associated investigations were made along with the INT-20 design study. It was evident that the structural effects on the vehicle imposed by design winds varied with payload length, so these effects were investigated.

An analysis was made of the INT-20/Big Gemini (Big G) configuration performance and of INT-20 performance with the J-2S engines in the S-IVB stage.

Removal of S-IVB re-start capability was considered since the INT-20 baseline mission does not require S-IVB restart.

Originally, a requirement for IBM to investigate an alternate (6 g) vehicle existed. This requirement was deleted so instead, IBM submitted a design for an improved flight control system.

These associated investigations are described in the paragraphs following.

4.3.1 BASELINE VEHICLE PAYLOAD SENSITIVITY STUDY

This payload sensitivity study was designed to determine the relationship between payload length, payload weight, and wind speed for an INT-20 vehicle with a 260 inch diameter MLV payload shape. Only two wind profiles were used in this study. These profiles represented the most severe wind month (March) and the least severe wind month (August).

The structural design of the S-IC, the S-IVB, and the I.U. is based upon wind criteria formulated for March, the month which has the highest wind speed. For August the design wind criteria is less severe and, consequently, the structural loading will be reduced. Payload length may be increased to take advantage of this load relief. Still further load margin and increased payload length may be gained by reducing the specified factor of safety or by making structural modifications to the critical stations (critical stations being defined as the stations most apt to fail with any further increase in structural loading).

4.3.1.1 STUDY GROUND RULES

4.3.1.1.1 Vehicle Trajectory and Aerodynamic Environment

a. Trajectory

Two vehicles were used in this study. One of the vehicles was the baseline INT-20 vehicle which has a payload weight of 132,026 pounds (Reference 4.3.1-1). The second study vehicle was a baseline vehicle with an 80,000 pound payload (Reference 4.3.1-2). All ground rules, stage weights, propellant capacities, and propulsion characteristics were the same for both of the study vehicles.

b. Aerodynamic Environment

The payload sensitivity study used normal force coefficient distributions for an INT-20 vehicle with a 260 inch diameter payload cylinder and a modified launch vehicle (MLV) nose cone as is shown in Reference 4.3.1-3. The normal aerodynamic forces were assumed to vary lineraly with changes in angle of attack.

4.3.1.1.2 Payload Envelopes

The payload envelopes were chosen to approximate the expected allowable payload lengths which would be permitted without structural modifications. Figure 4.3.1.1-1 shows the three payload envelopes which were used in this study. It was recognized that the longest payload envelope would not be as long as the payload length which could be obtained for the

4.3.1.1.2 (Continued)

August wind, unmanned factor of safety condition. However, the longest payload lengths are most affected by the structural dynamics of the payload and since the payload for this study is undefined, no attempt was made to define the very longest allowable payload length. Instead the emphasis was put on defining the payload lengths from 43 feet to 73 feet.

Payload densities were based on payload weights of 132,026 pounds and 80,000 pounds. These payload weights were assumed to be uniformly distributed throughout the entire payload envelope and the MLV nose cone was considered to be usable volume. Figure 4.3.1.1-2 gives a plot of payload density and volume versus payload length for both of the weights.

4.3.1.1.3 Structural Capability

The structural capabilities which were used in this payload sensitivity study were supplied by the stage contractors in References 4.3.1-4 through 4.3.1-7. A tabulation of the compressive structural capabilities for the MAX ($Q\alpha$) condition and the maximum acceleration condition are given in Table 4.3.1.1-1.

4.3.1.1.4 Wind Criteria

The ground wind profiles which were used in this payload sensitivity were developed from Reference 4.3.1-8 and the inflight wind profiles used in the analysis were also obtained by using the methods given in Reference h. Two inflight wind profiles were used in this payload sensitivity study and are shown in Figure 4.3.1.1-3. These profiles are for a 95 percentile March wind (peak wind speed equal 75 meters/second) and for a 95 percentile August wind (peak wind speed equal 22 meters/second). Superimposed upon each inflight wind profile is an embedded jet gust. Both the shear buildup of the wind profiles and the gust magnitudes were reduced 15 percent.

The two inflight wind profiles which were used represent the most severe design wind criteria and the least severe criteria.

4.3.1.2 TECHNICAL APPROACH

Ground wind bending moments were determined for all payload configurations for both the 99.9 percentile pre-launch wind and the 99 percentile launch wind by the technique given in Reference i.

Flight simulations were performed for the INT-20 payload sensitivity study vehicles having payload lengths as shown in Figure 4.3.1.1-1 and payload weights of 132,026 pounds and 80,000 pounds. These simulations provided the flexible body responses during first stage boost. Rigid body translation and rotation in the yaw plane, one free-free bending mode, and two nozzle degrees of freedom were included in the simulation.

4.3.1.2 (Continued)

The wind profiles shown in Figure 4.3.1.1-3 were applied in the yaw plane and were used in each flight simulation.

Ground wind and inflight bending moment distributions were determined for each payload length, payload weight, and wind speed. The ultimate compressive combined load was then obtained from the following formula:

$$N_C \text{ULT.} = \left[\frac{BM(X)}{\pi R^2(X)} + \frac{P(X)}{2\pi R(X)} \right] F.S. - \frac{P_{U\ MIN}\ R(X)}{2}$$

where: BM(X) = distributed bending moment

P(X) = distributed longitudinal force including aerodyanmic forebody drag

R(X) = distributed body radius

$P_{U\ MIN}$ = minimum ullage pressure (applicable to tank shells only)

F.S. = factor of safety of 1.4 for manned missions and 1.25 for unmanned missions

Critical vehicle stations were then identified by investigating ultimate compressive combined load as a function of payload length at all stations. The critical vehicle stations then determined the maximum allowable payload lengths for the various payload weights and factors of safety for March design criteria and for August design criteria.

4.3.1.3 RESULTS AND DISCUSSION

4.3.1.3.1 Payload Sensitivity Study Results

a. On-Pad Conditions

Representative plots of ultimate combined compressive load versus payload length for the on-pad, fueled, unpressurized condition and the emergency shut-down condition and are shown in Figures 4.3.1.3-1 through 4.3.1.3-6 for the 132,026 pound payload vehicle. These representative plots show that the on-pad conditions are not critical for the 132,026 pound payload vehicle. Therefore, they will not be critical for the 80,000 pound payload.

b. Max ($Q\alpha$) Condition

Figures 4.3.1.3-7 through 4.3.1.3-14 present ultimate compressive combined load versus payload length for the 132,026 pound payload vehicle at the Max ($Q\alpha$) condition. These figures give data for March and August design winds and for factors of safety of 1.4 and 1.25. Identical data for the 80,000 pound payload vehicle is shown in Figures 4.3.1.3-15 through 4.3.1.3-22. Plots are given for the following vehicle stations: 1541 Fwd., 1768 Fwd. and Aft., 1854 Aft., 1854 Fwd., 2123 Aft., 2123 Fwd., 2245 Fwd. and Aft., and 2281 Aft. These

4.3.1.3.1 (Continued)

are the vehicle stations which give the smallest allowable payload lengths.

The S-IC stage has sufficient structural capability to accept longer payload lengths than the other stages. Therefore, none of the S-IC stations were identified as critical stations. Plots which show the magnitude of the ultimate compressive combined load on the S-IC stage for each of the payload lengths and payload weights are presented in Figures 4.3.1.3-23 through 4.3.1.3-30. These figures give data for both factors of safety and both design wind criteria.

c. Maximum Acceleration Condition

Figure 4.3.1.3-31 shows a plot of ultimate combined compressive load at the maximum acceleration condition versus payload weight for the three most critical vehicle stations. Data for both factors of safety of 1.4 and 1.25 is shown on the figure for each of the critical stations.

4.3.1.3.2 Discussion of Results

The purpose of this study was to determine which MLV payload envelopes might be used on a baseline INT-20 vehicle for March design wind criteria and for August design wind criteria. Both manned and unmanned missions (factors of safety of 1.4 and 1.25, respectively) were considered. Results were obtained for two payload weights so that the effects of changes in payload weight could be assessed. In addition, it was desired to identify the gains in payload length that could be obtained with minor structural modifications.

Figures 4.3.1.3-1 through 4.3.1.3-6 show that for the 132,026 pound payload the on-pad design conditions (on-pad, fueled, unpressurized condition and emergency shutdown condition) do not give critical compressive loads for any of the payload lengths which were investigated. Consequently, since the on-pad compressive axial forces for the 80,000 pound payload are less than for the 132,026 pound payload, the on-pad design conditions for the 80,000 pound payload will also not give critical compressive loads. Furthermore the on-pad loads for both payload weights are of such a low magnitude compared to structural capability that it can be concluded that the on-pad conditions are not the critical conditions which will determine the allowable payload lengths for the INT-20 vehicle.

Figures 4.3.1.3-7 through 4.3.1.3-14 present max ($Q\alpha$) data for the 132,026 pound payload. Reference to these figures shows that for a factor of safety of 1.4 and March design wind criteria, vehicle station 2245 Forward determine the allowable payload length. This critical station, located at the aft end of the instrument unit, limits the overall payload length to 43.2 feet. This station is also critical when a factor of safety of 1.25 is used. The reduction in factor of safety from 1.4 to 1.25 yields a 4.4 ft. increase in payload length for the 132,026 pound payload with March design criteria. For August design wind criteria the allowable payload length for a factor of safety of 1.4 is 69.6 feet and the critical station is station 2245 forward. When a factor of safety of 1.25 is considered the allowable payload length will be greater than 73 feet. Table 4.3.1.3-I contains a summary

4.3.1.3.2. (Continued)

of the data which is presented in Figures 4.3.1.3-7 through 4.3.1.3-14.

Data for the 80,000 pound payload at the Max ($Q\alpha$) condition is presented in Figures 4.3.1.3-15 through 4.3.1.3-22. The critical vehicle station is once again the aft end of the instrument unit and this station limits the allowable payload length for a safety factor of 1.4 and March design criteria to 45.6 feet. For a safety factor of 1.25 and March criteria, an overall payload length of 49.4 feet can be accepted without exceeding the structural capability of the instrument unit.

For the 80,000 pound payload with August design wind criteria, both factors of safety will yield allowable payload lengths which are greater than 73 feet. A summary of the 80,000 pound payload data is given in Table 4.3.1.3-II.

If structural modification of the instrument unit is considered, then additional payload length can be obtained. For the 132,026 pound payload, additional structural capability for the instrument unit will result in an allowable payload length of 48.9 feet for a factor of safety of 1.4 and March design wind criteria. Table 4.3.1.3-I shows that for instrument unit structural modification, safety factor of 1.4, and March design wind criteria, the critical vehicle station will become station 2245 aft (the forward end of the S-IVB forward skirt). Reference to Figures 4.3.1.3-13 and 4.3.1.3-14 shows that to obtain a payload length of 48.9 feet the structural capability of the aft end of the instrument unit must be increased to 1670 Lb/In and the capability of the forward end of the instrument unit must be at least 1540 Lb/In. For a safety factor of 1.25 and March criteria, structural modification of the instrument unit will make station 2245 aft the critical station. This station will limit the overall payload length to 53.6 feet. This payload length will necessitate a structural capability of 1670 Lb/In at the aft end of the instrument unit and a capability of 1560 Lb/In at the forward end of the instrument unit.

Structural modification of the instrument unit on the 80,000 pound payload vehicle with a safety factor of 1.4 and March design criteria will cause vehicle station 1768 aft (forward end of the S-IC/S-IVB interstage) to become the critical station thereby limiting the allowable payload length to 48.7 feet. Then to accomodate this payload length the instrument unit must have the following structural capabilities: 1580 Lb/In at the aft end (station 2245 forward); and 1440 Lb/In at the forward end (station 2281 aft). For a factor of safety of 1.25 and March design criteria, station 2245 aft becomes the critical station when the structural capability of the instrument unit is increased. This critical station, located at the forward end of the S-IVB forward skirt, limits the overall payload length to 55 feet. To obtain this payload length the instrument unit must have a capability of 1670 Lb/In at its aft end and 1540 Lb/In at its forward end.

Table 4.3.1.3-III gives a summary of the allowable payload lengths for both payload weights and for no structural modification and structural modification

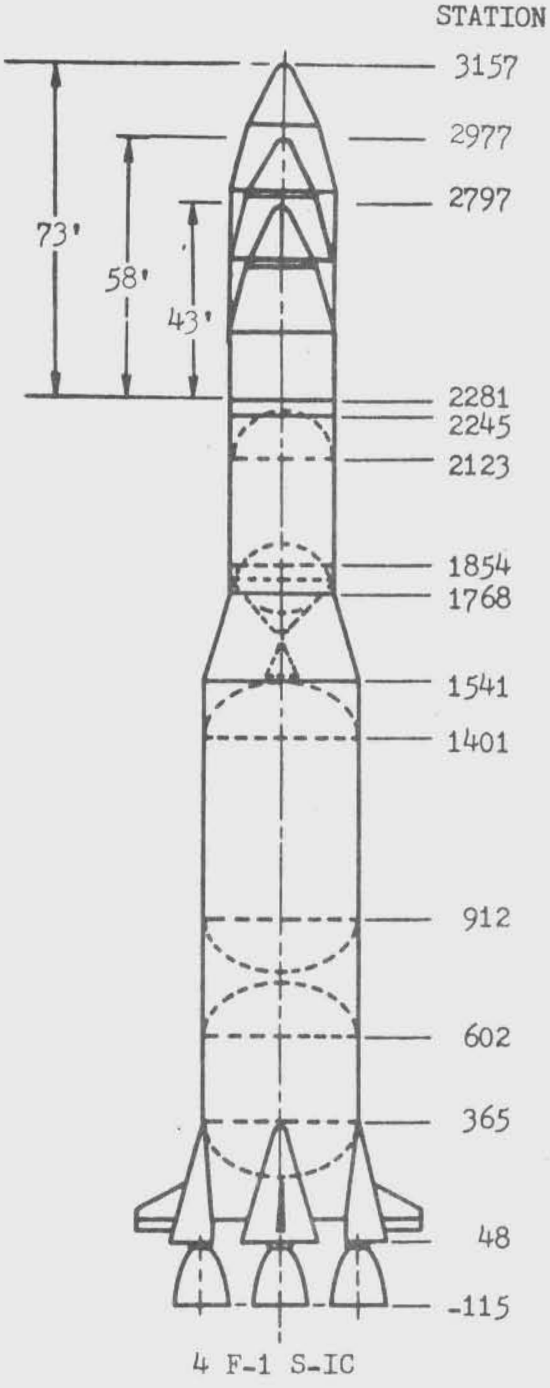
4.3.1.3.2 (Continued)

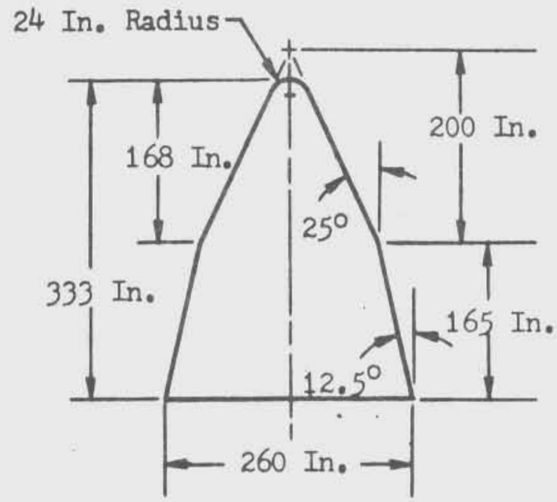
to the instrument unit.

Additional modification steps which would result in longer allowable payload lengths could also be considered. Critical stations could be determined from Tables 4.3.1.3-I and 4.3.1.3-II and the necessary modification levels could be determined from Figures 4.3.1.3-7 through 4.3.1.3-22.

Figure 4.3.1.3-31 shows that when the structural capability from Reference g is used none of the INT-20 stations experience a structural overload during the maximum acceleration condition. This is true for both the 80,000 pound payload and the 132,026 pound payload. Table 4.3.1.3-IV presents the limiting accelerations, factors of safety, and allowable payload weights for the most critical vehicle stations.

The Max (Q_{α}) design condition determines the allowable payload lengths for the INT-20 vehicle with a 260 inch diameter MLV payload shape. Vehicle station 2245 forward, the aft end of the instrument unit, is the most critical station. For a factor of safety of 1.4 and March design wind criteria, this station limits the total payload length to 43.2 feet when a 132,026 pound payload is used and to 45.6 feet for an 80,000 pound payload. Allowable payload lengths associated with vehicle stations 1541 FWD, 1768 AFT & FWD, 1854 AFT &FWD, and 2123 AFT are greater for the 132,026 pound payload than for the 80,000 pound payload. However, the 132,026 pound payload gives shorter allowable payload lengths at vehicle stations 2123 FWD, 2245 AFT & FWD, and 2281 AFT & FWD. Thus changes in payload weight have varying effects upon different sections of the vehicle.





MLV NOSE CONE

NOTE:
1. INT-20 BASED ON SATURN V AS-511 DESIGN
2. J-2 ENGINE ON S-IVB
3. LAUNCH ESCAPE SYSTEM (LES) NOT CONSIDERED FOR PHASE II ANALYSIS AND DESIGN

FIGURE 4.3.1.1-1 INT-20 CONFIGURATIONS FOR PAYLOAD SENSITIVITY STUDY

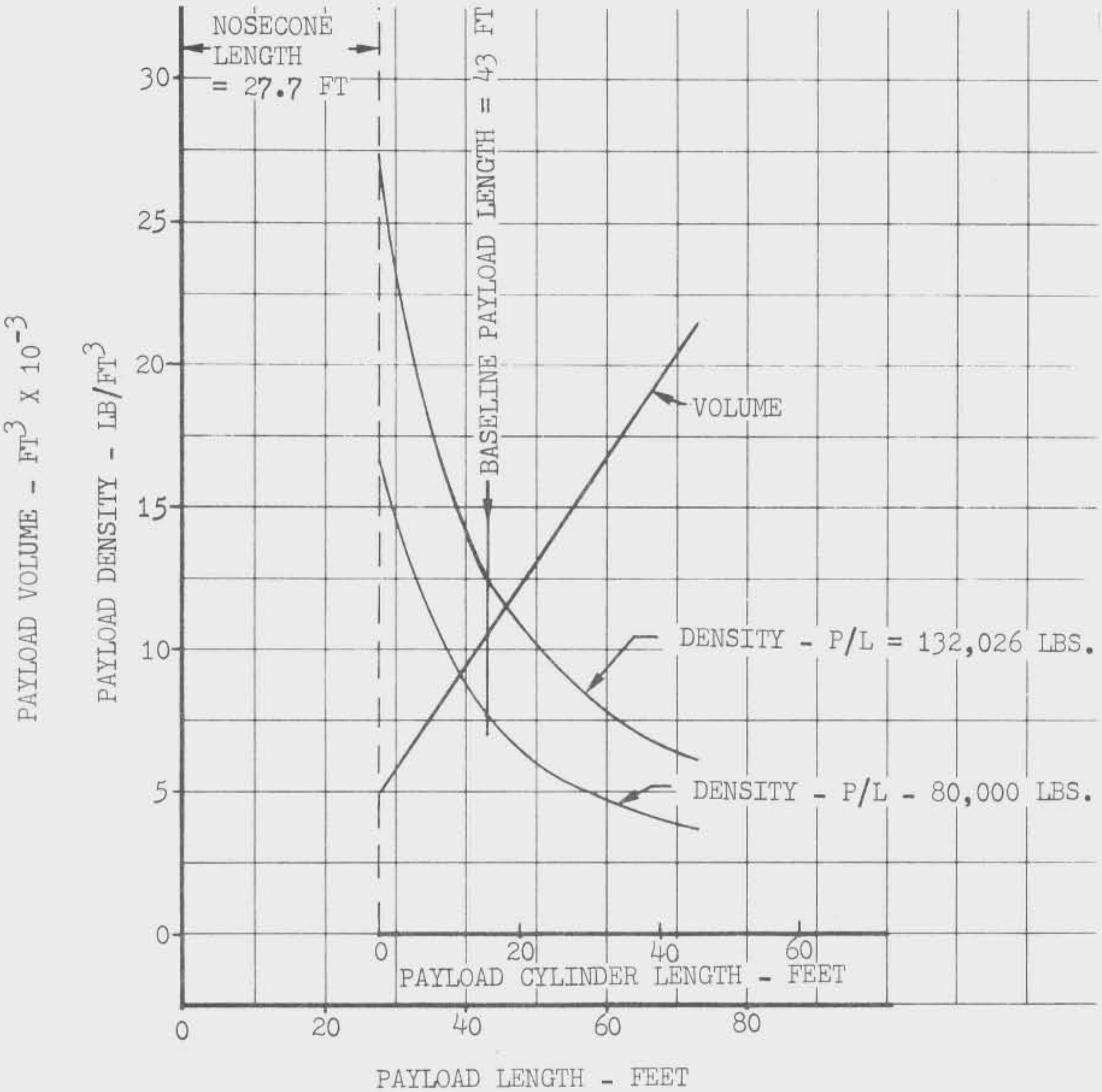


FIGURE 4.3.1.1-2 INT-20 PAYLOAD VOLUME AND DENSITY VERSUS PAYLOAD LENGTH

TABLE 4.3.1.1-I

STRUCTURAL CAPABILITY FOR THE INT-20 PAYLOAD SENSITIVITY STUDY

VEHICLE STATION (IN)	COMPRESSIVE CAPABILITY AT MAX ($Q\alpha$) (LB/IN)	COMPRESSIVE CAPABILITY AT MAX. ACCELERATION (LB/IN)
2281A	1435	1400
2245F	1435	1400
2245A	1670	1250
2123F	2200	1700
2123A	1420	—
1854F	1420	—
1854A	3900	3460
1768F	4270	3800
1768A	3920	3400
1541F	2570	2210
1541A	7340	7340
1401F	7350	7350
1401A	6470	6200
912F	6240	6630
912A	11,400	11,400
602F	11,400	11,400
602A	7910	7750
365F	7160	7300

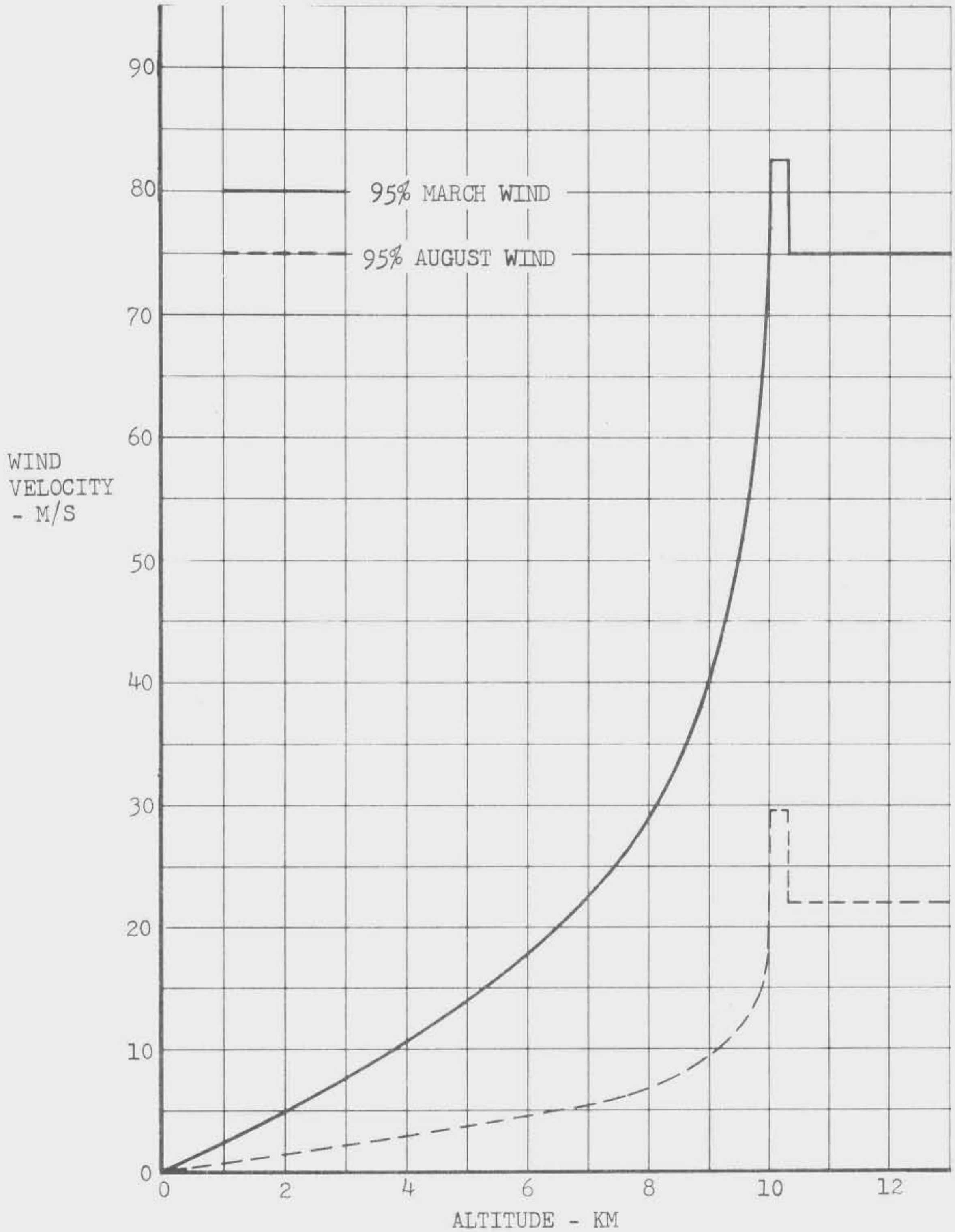


FIGURE 4.3.1.1-3 INFLIGHT WIND PROFILE FOR MARCH AND AUGUST

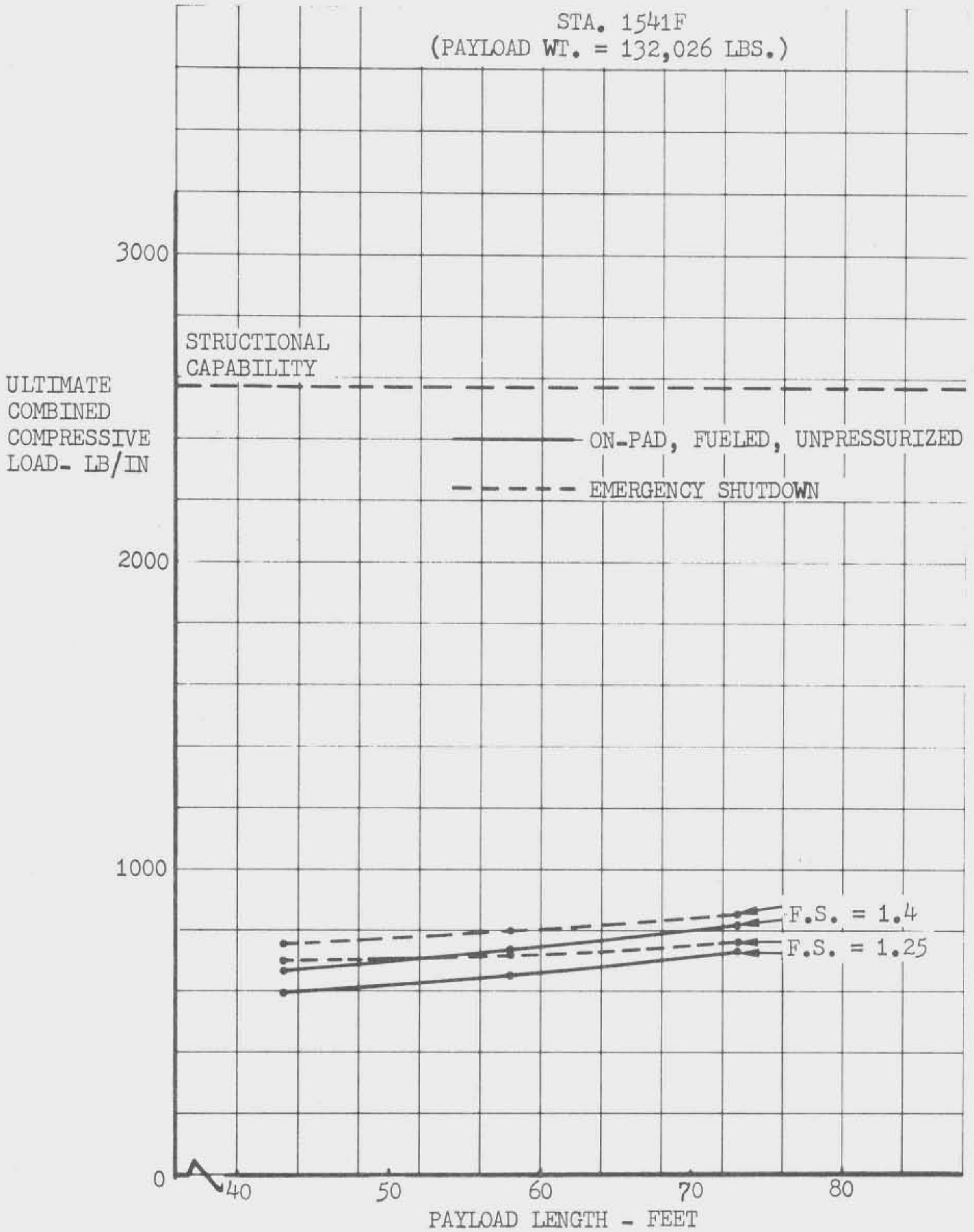


FIGURE 4.3.1.3-1 ON-PAD N_C ULTIMATE @ STA. 1541F VERSUS TOTAL PAYLOAD LENGTH FOR THE INT-20 VEHICLE WITH A 132,026 LB. PAYLOAD

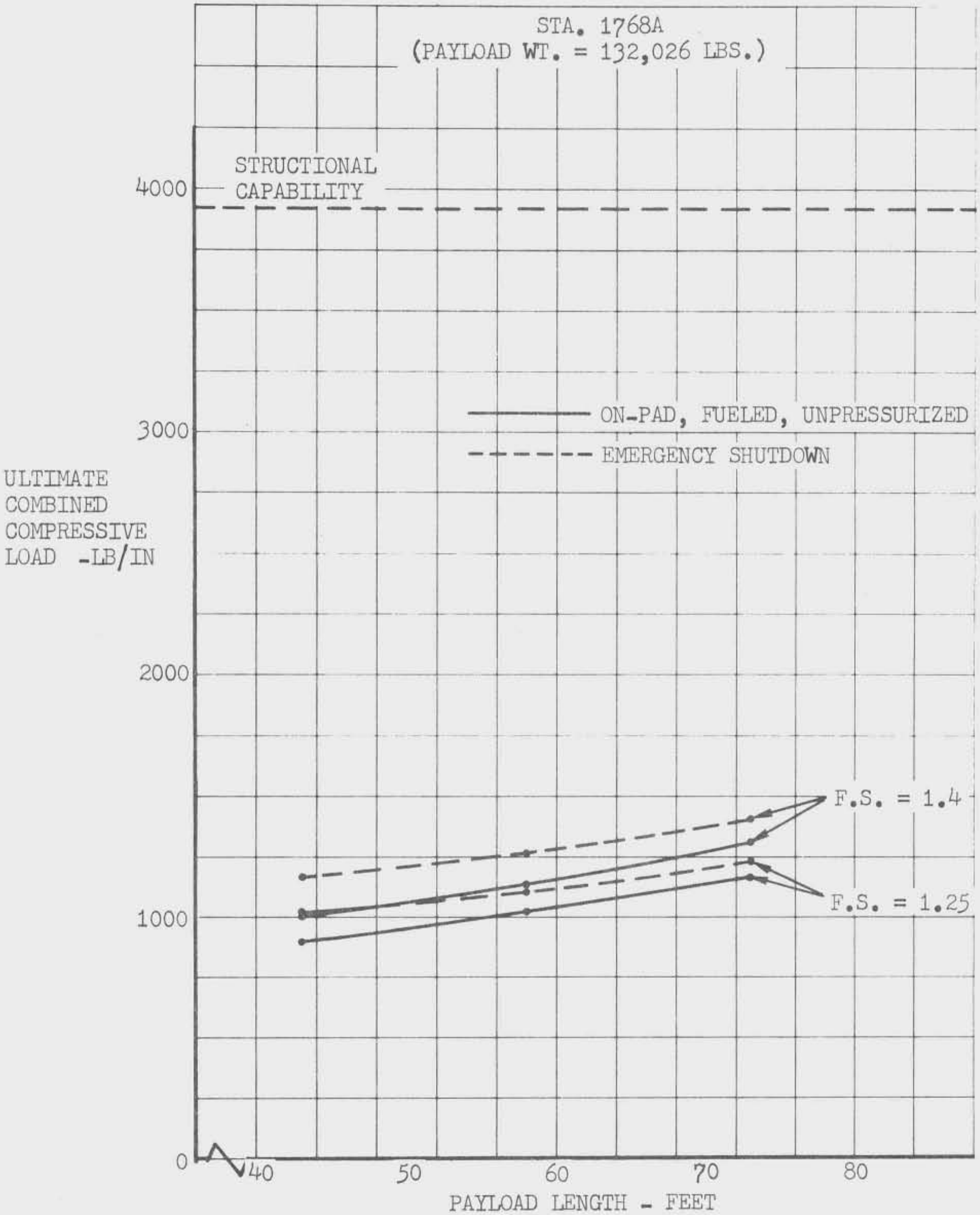


FIGURE 4.3.1.3-2 ON-PAD N_C ULTIMATE @ STA. 1768A VERSUS TOTAL PAYLOAD LENGTH FOR THE INT-20 VEHICLE WITH A 132,026 LB. PAYLOAD

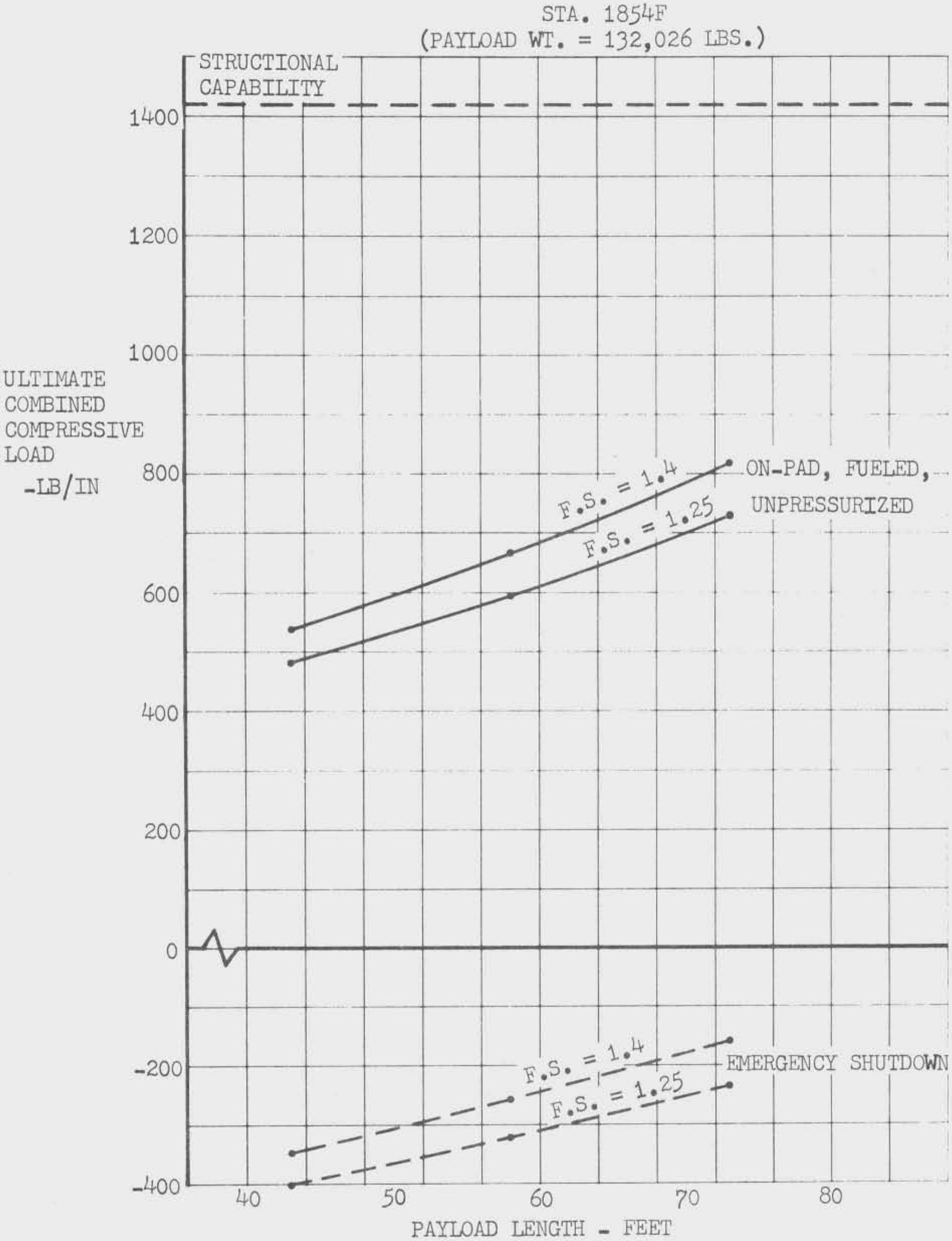


FIGURE 4.3.1.3-3 ON-PAD N_C ULTIMATE @ STA. 1854F VERSUS TOTAL PAYLOAD LENGTH FOR THE INT-20 VEHICLE WITH A 132,026 LB. PAYLOAD

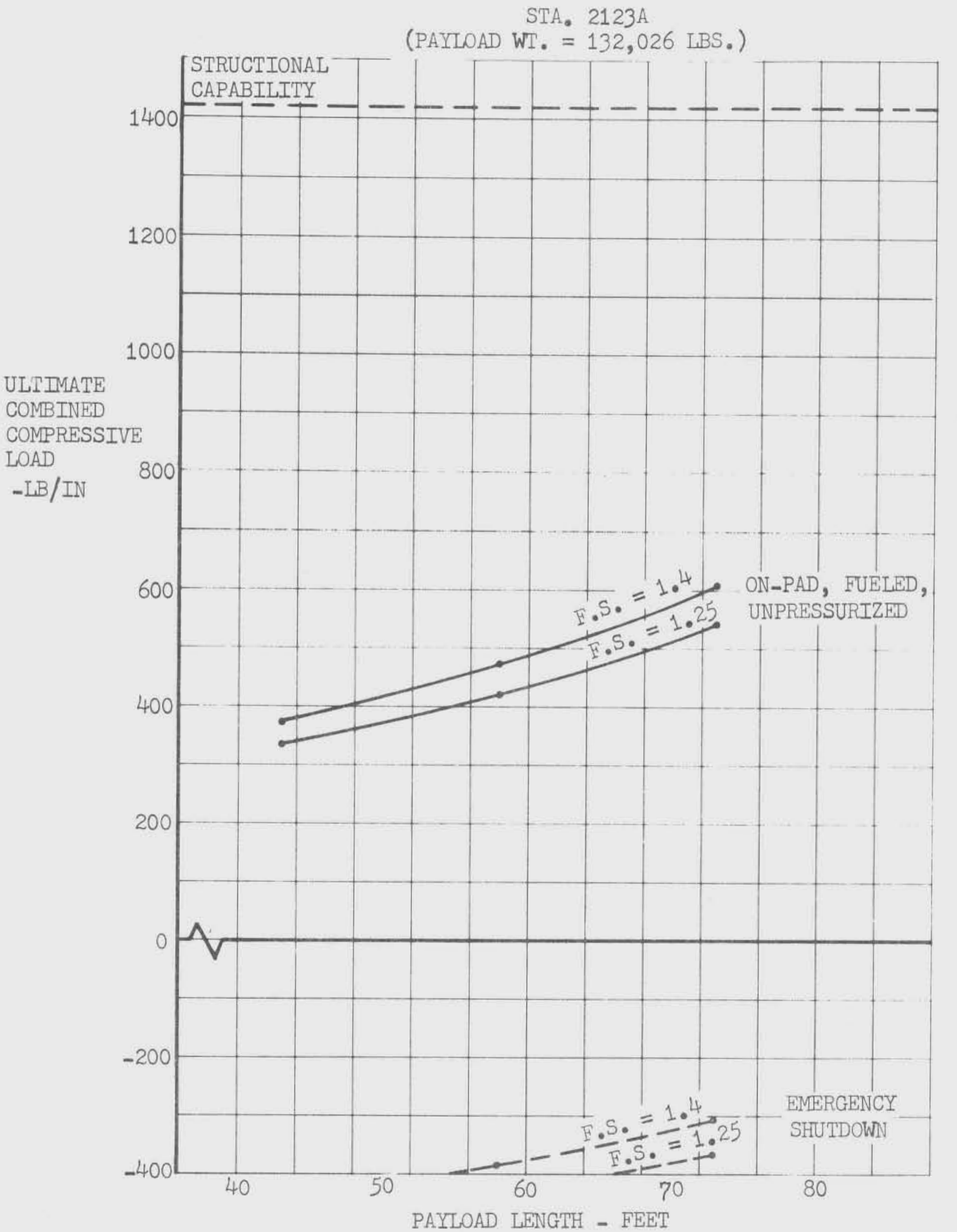


FIGURE 4.3.1.3-4 ON-PAD N_C ULTIMATE @ STA. 2123A VERSUS TOTAL PAYLOAD LENGTH FOR THE INT-20 VEHICLE WITH A 132,026 LB. PAYLOAD

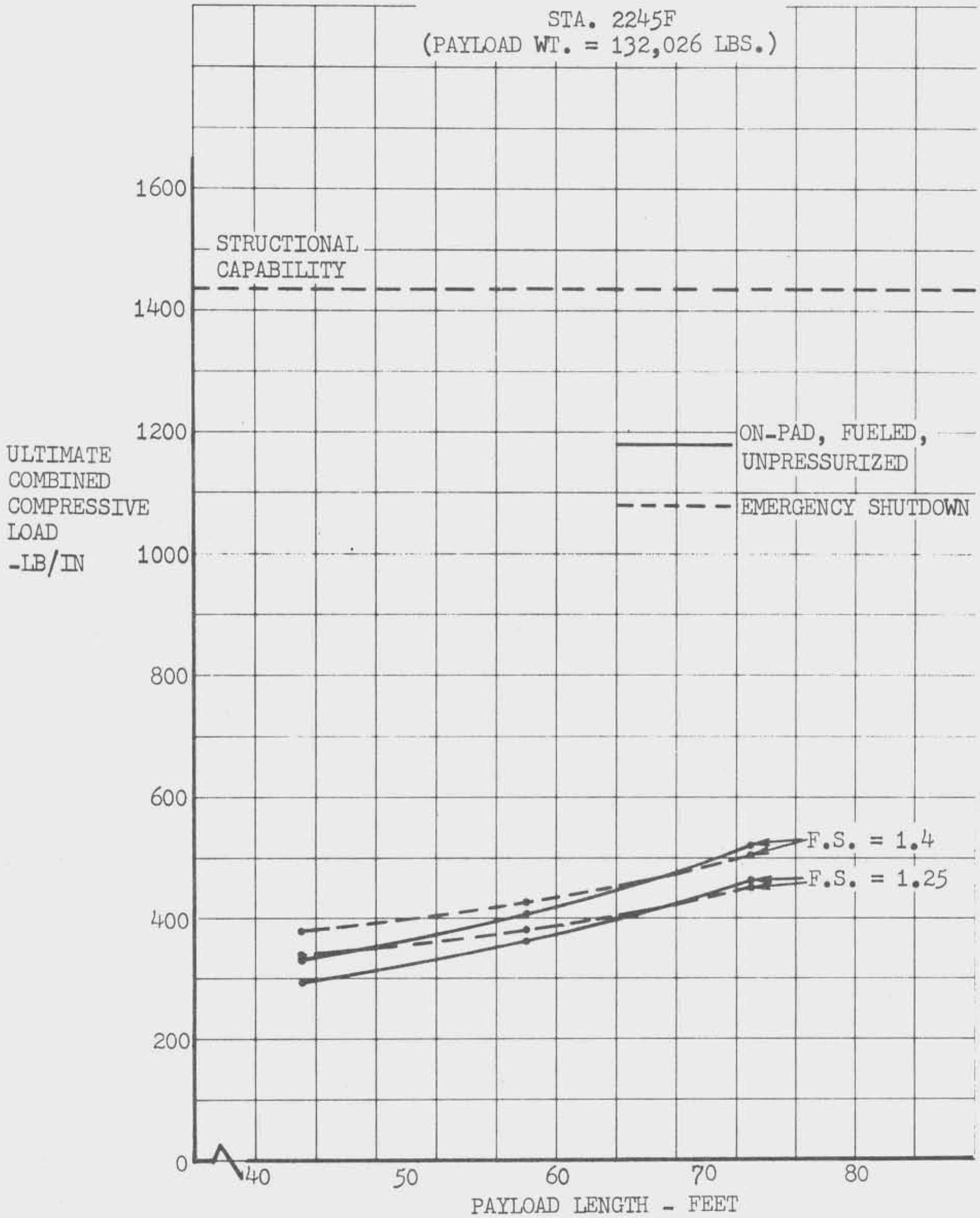


FIGURE 4.3.1.3-5 ON-PAD N_C ULTIMATE @ STA. 2245F VERSUS TOTAL PAYLOAD LENGTH FOR THE INT-20 VEHICLE WITH A 132,026 LB. PAYLOAD

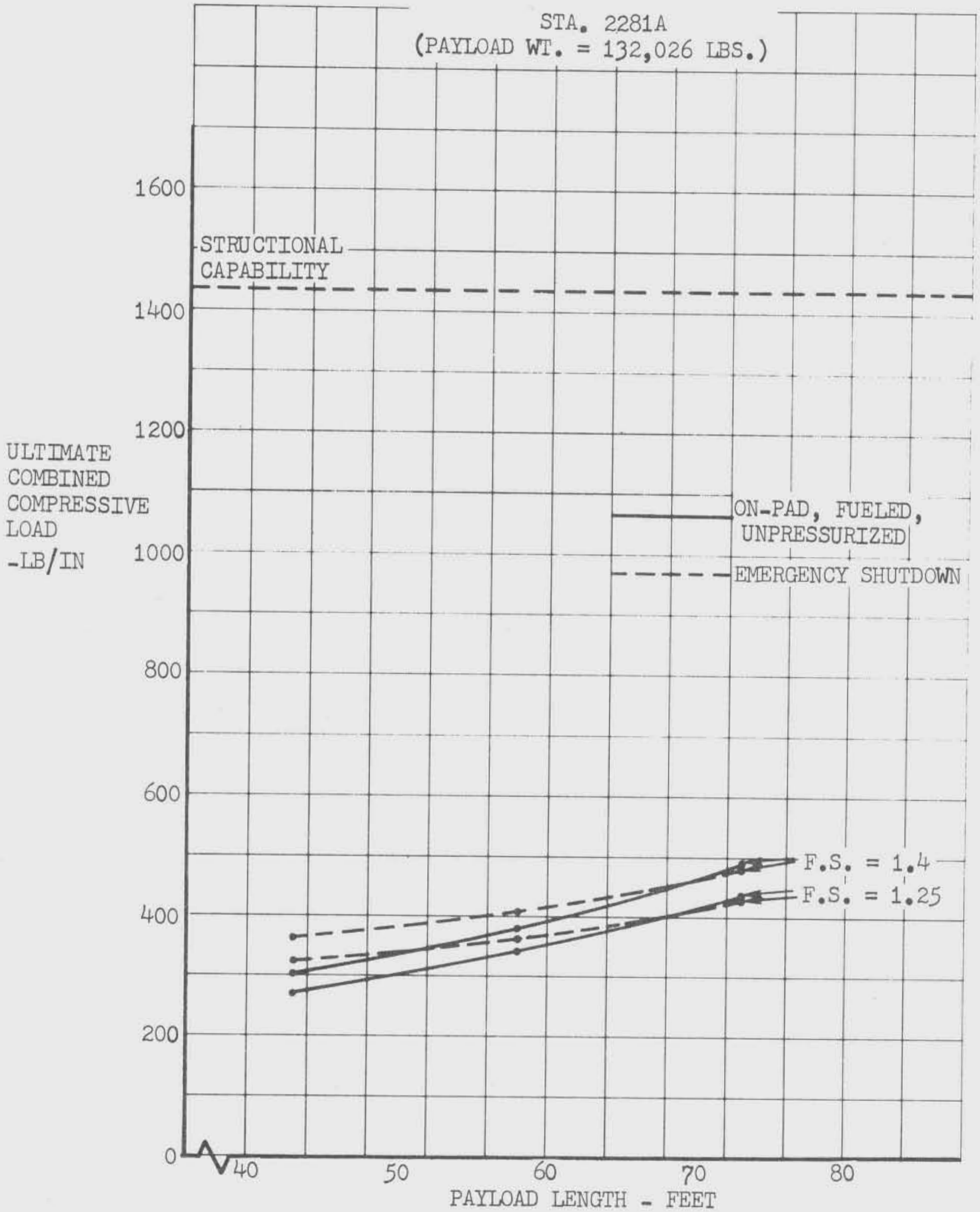


FIGURE 4.3.1.3-6 ON-PAD N_C ULTIMATE @ STA. 2281A VERSUS TOTAL PAYLOAD LENGTH FOR THE INT-20 VEHICLE WITH A 132,026 LB. PAYLOAD

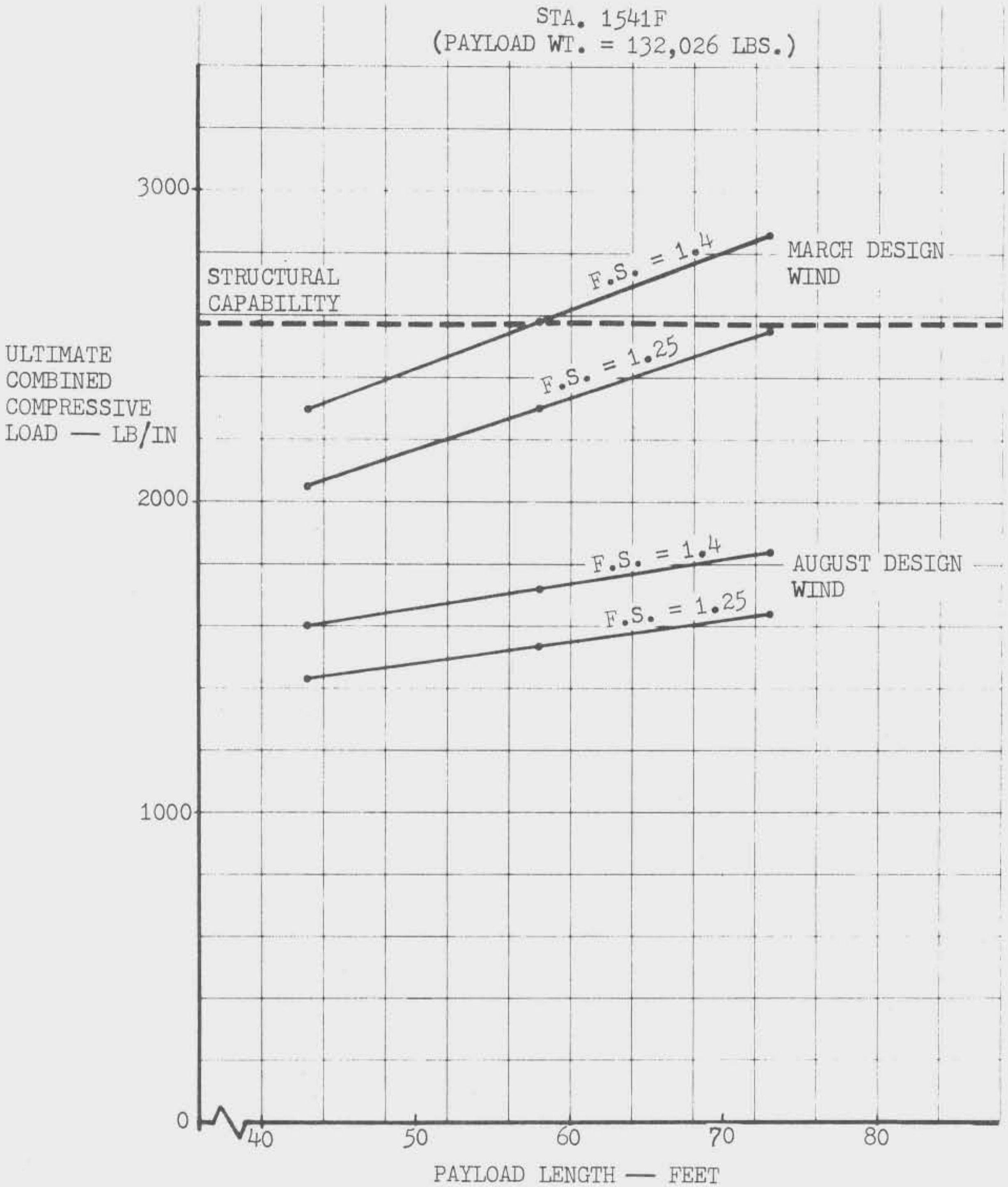


FIGURE 4.3.1.3-7 MAX ($Q\alpha$) N_C ULTIMATE @ STA 1541F VERSUS TOTAL PAYLOAD LENGTH FOR THE INT-20 VEHICLE WITH A 132,026 LB. PAYLOAD

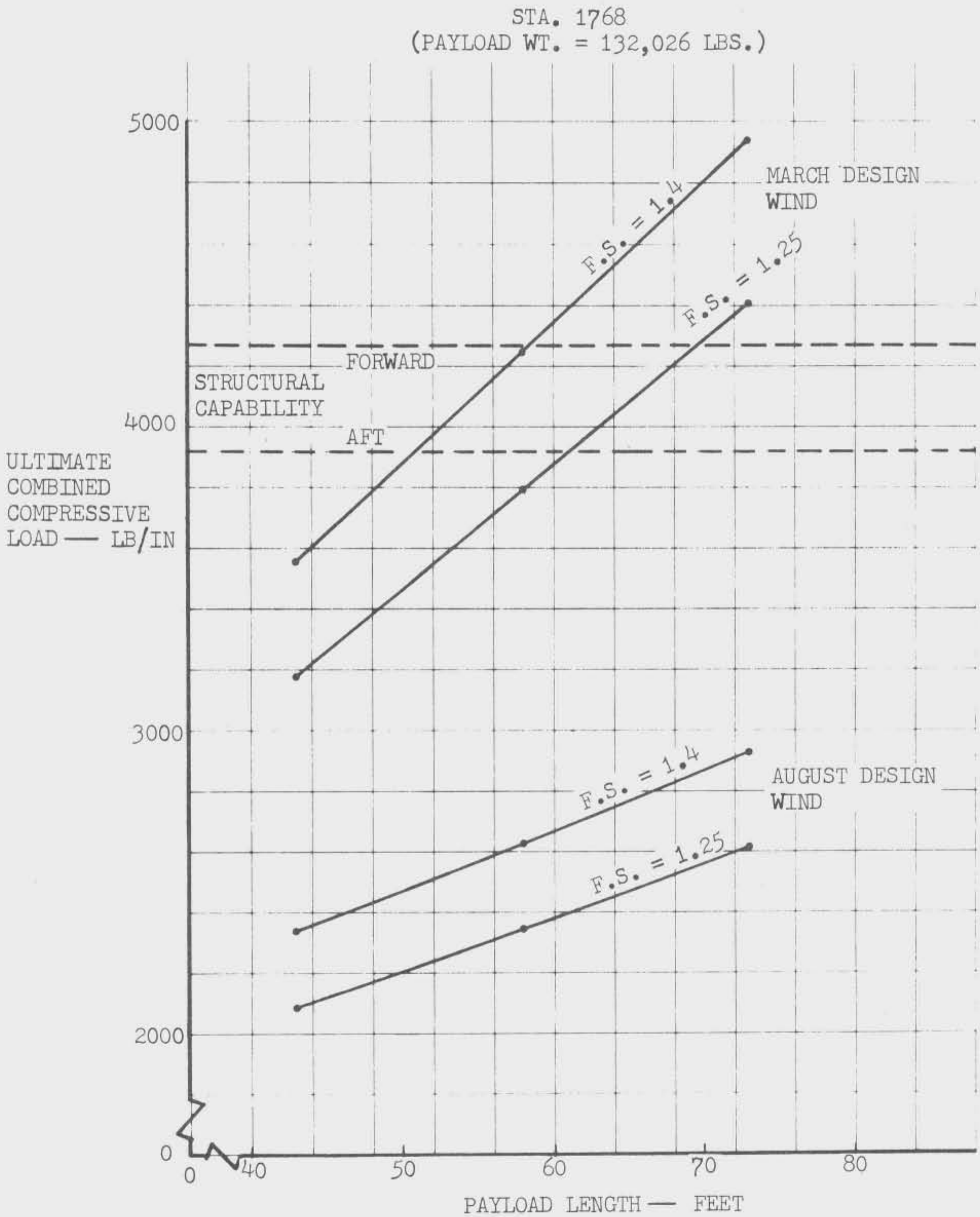


FIGURE 4.3.1.3-8 MAX (Qα) N_C ULTIMATE @ STA 1768 VERSUS TOTAL PAYLOAD LENGTH FOR THE INT-20 VEHICLE WITH A 132,026 LB.PPAYLOAD

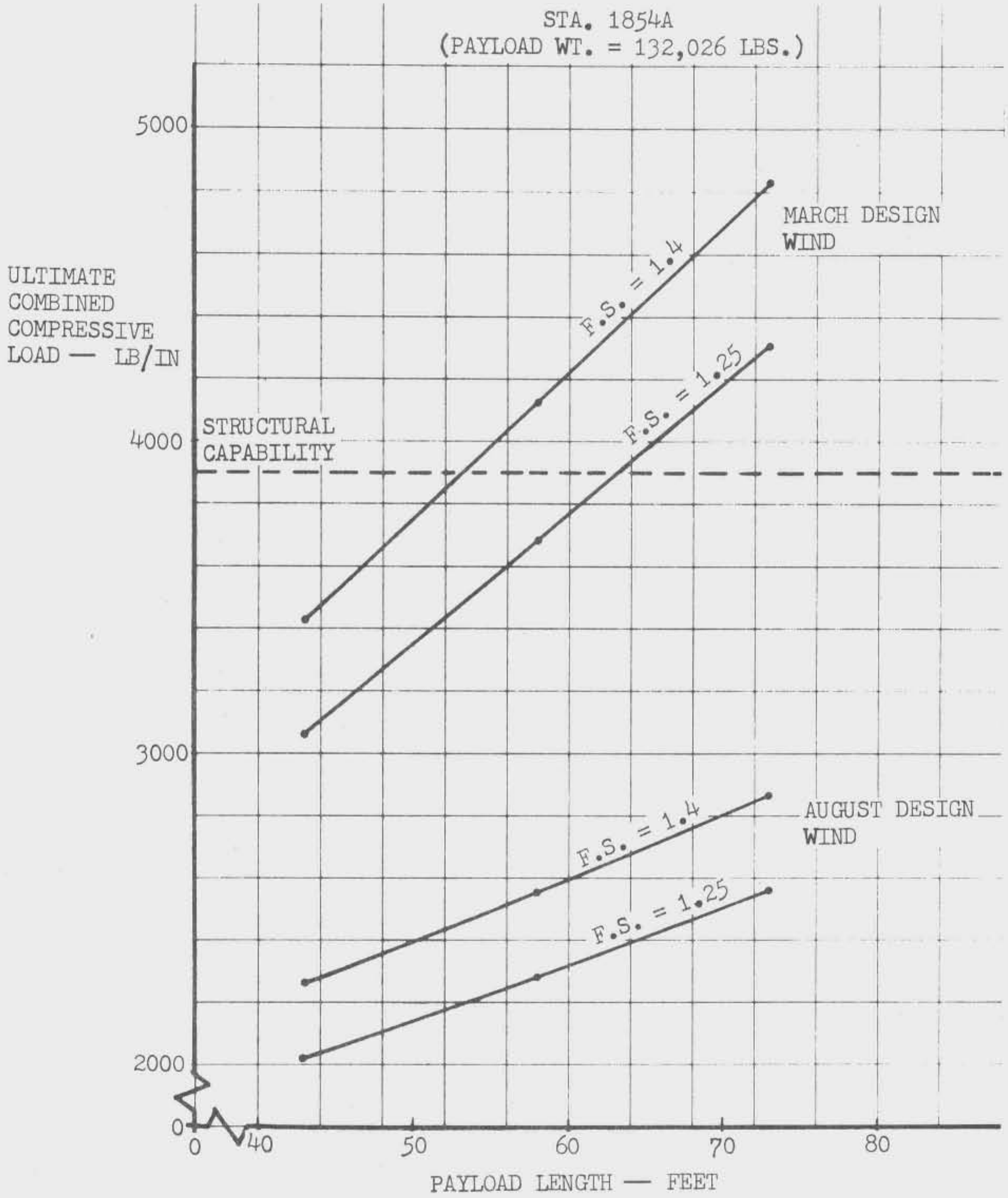


FIGURE 4.3.1.3-9 MAX $(Q\alpha)N_C$ ULTIMATE @ STA 1854A VERSUS TOTAL PAYLOAD LENGTH FOR THE INT-20 VEHICLE WITH A 132,026 LB. PAYLOAD

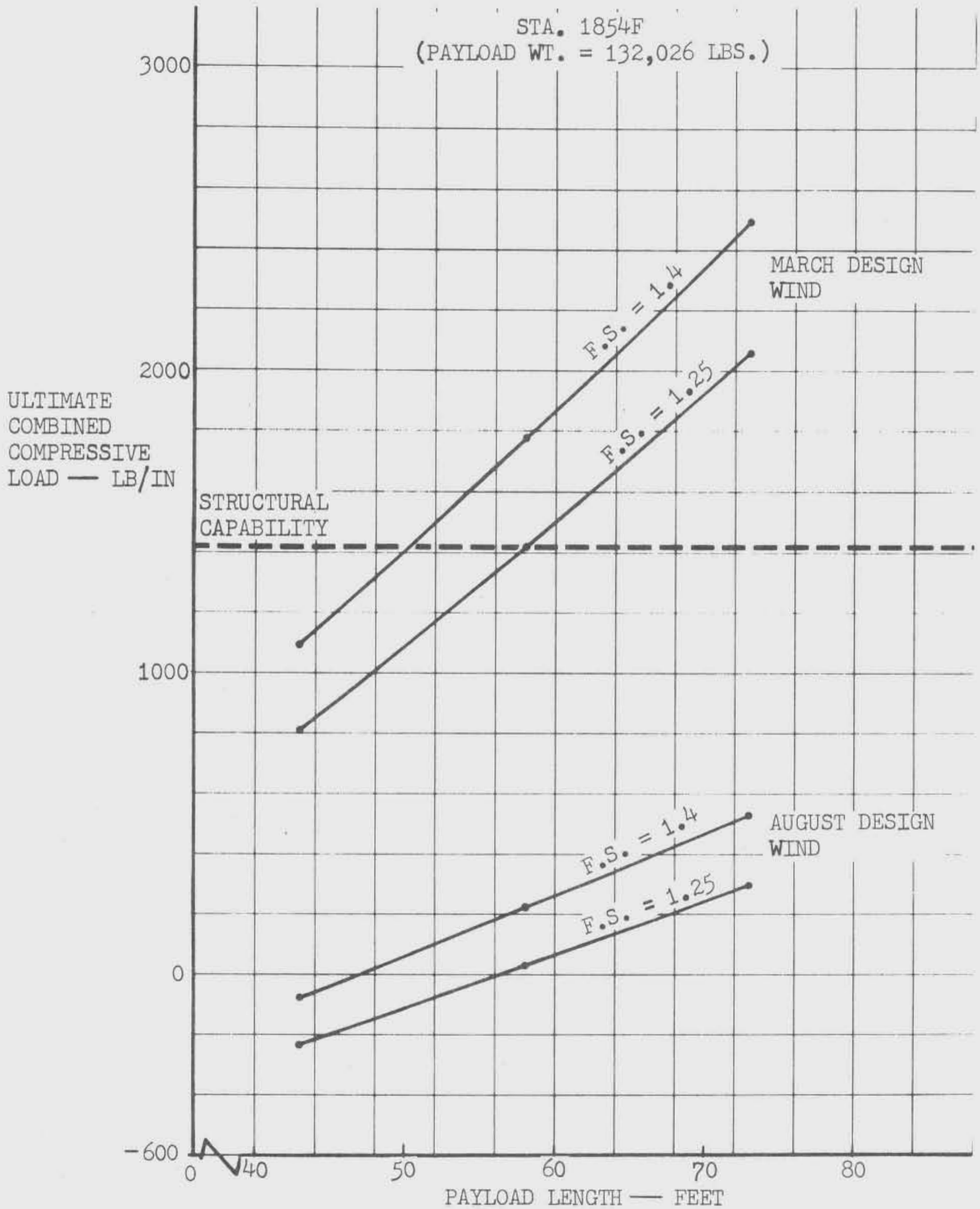


FIGURE 4.3.1.3-10 MAX (Qα) N_C ULTIMATE @ STA 1854F VERSUS TOTAL PAYLOAD LENGTH FOR THE INT-20 VEHICLE WITH A 132,026 LB. PAYLOAD

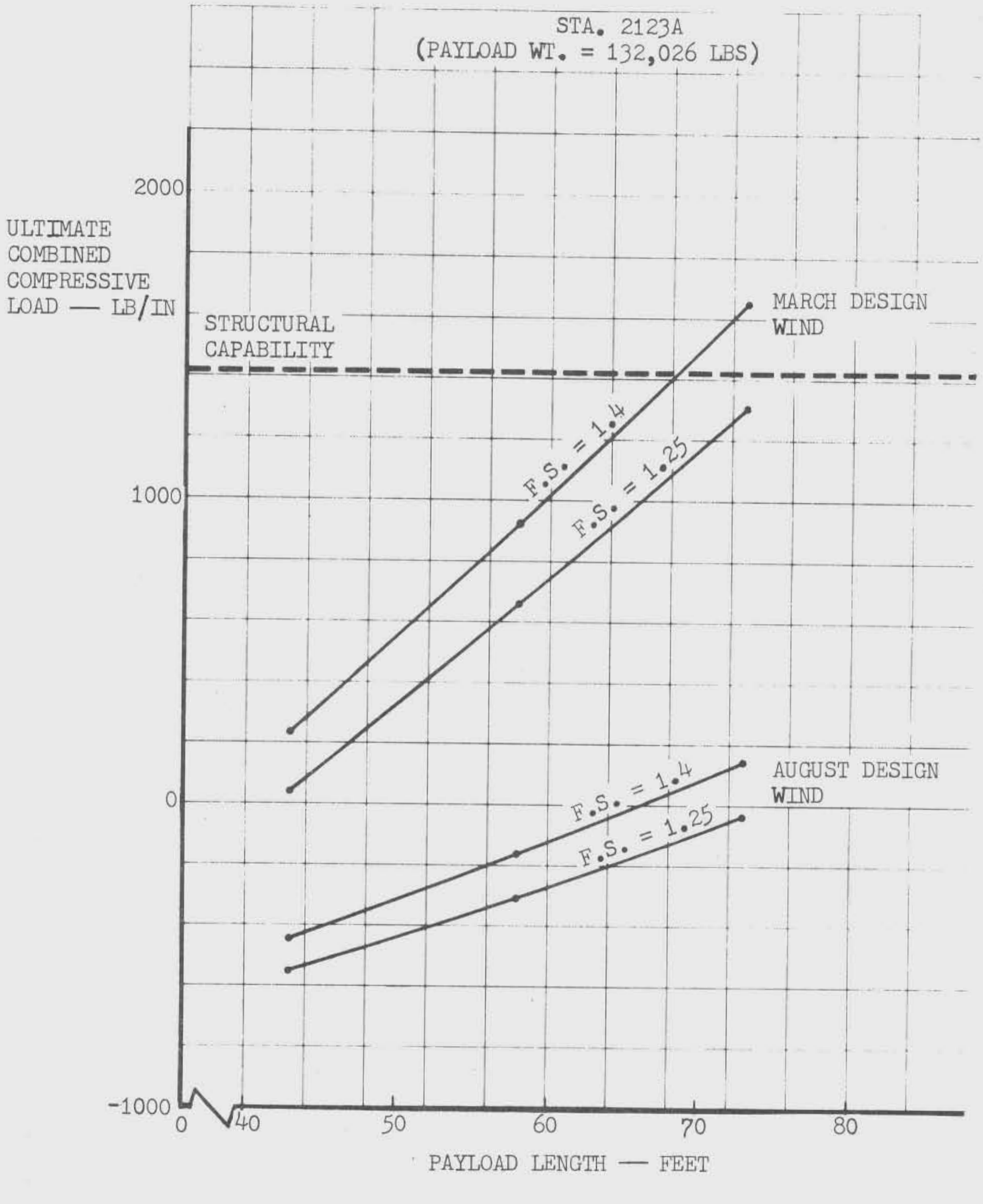


FIGURE 4.3.1.3-11 MAX ($Q\alpha$) N_C ULTIMATE @ STA 2123A VERSUS TOTAL PAYLOAD LENGTH FOR THE INT-20 VEHICLE WITH A 132,026 LB. PAYLOAD

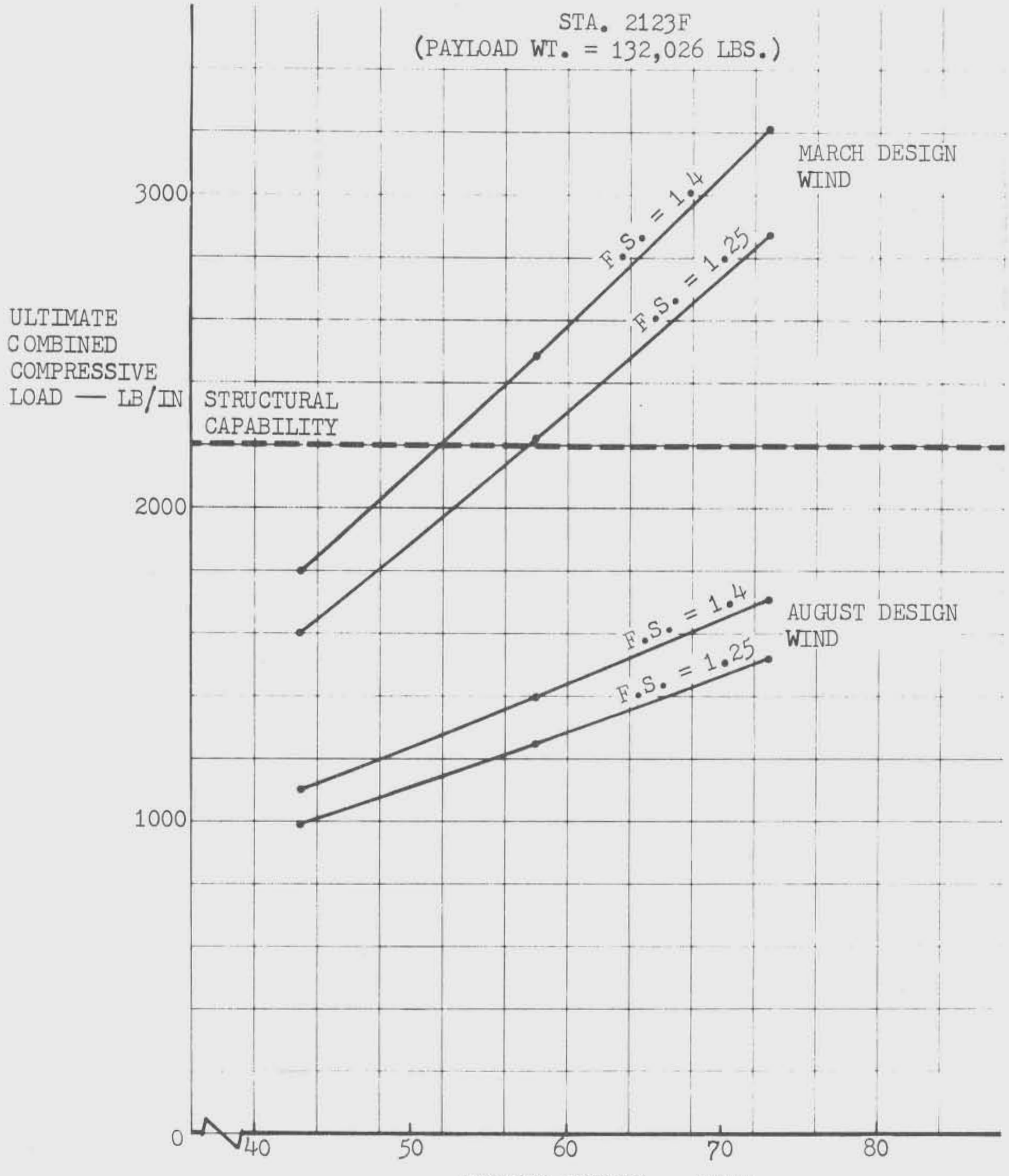


FIGURE 4.3.1.3-12 MAX (Q∝) N_C ULTIMATE @ STA 2123F VERSUS TOTAL PAYLOAD LENGTH FOR THE INT-20 VEHICLE WITH A 132,026 LB. PAYLOAD

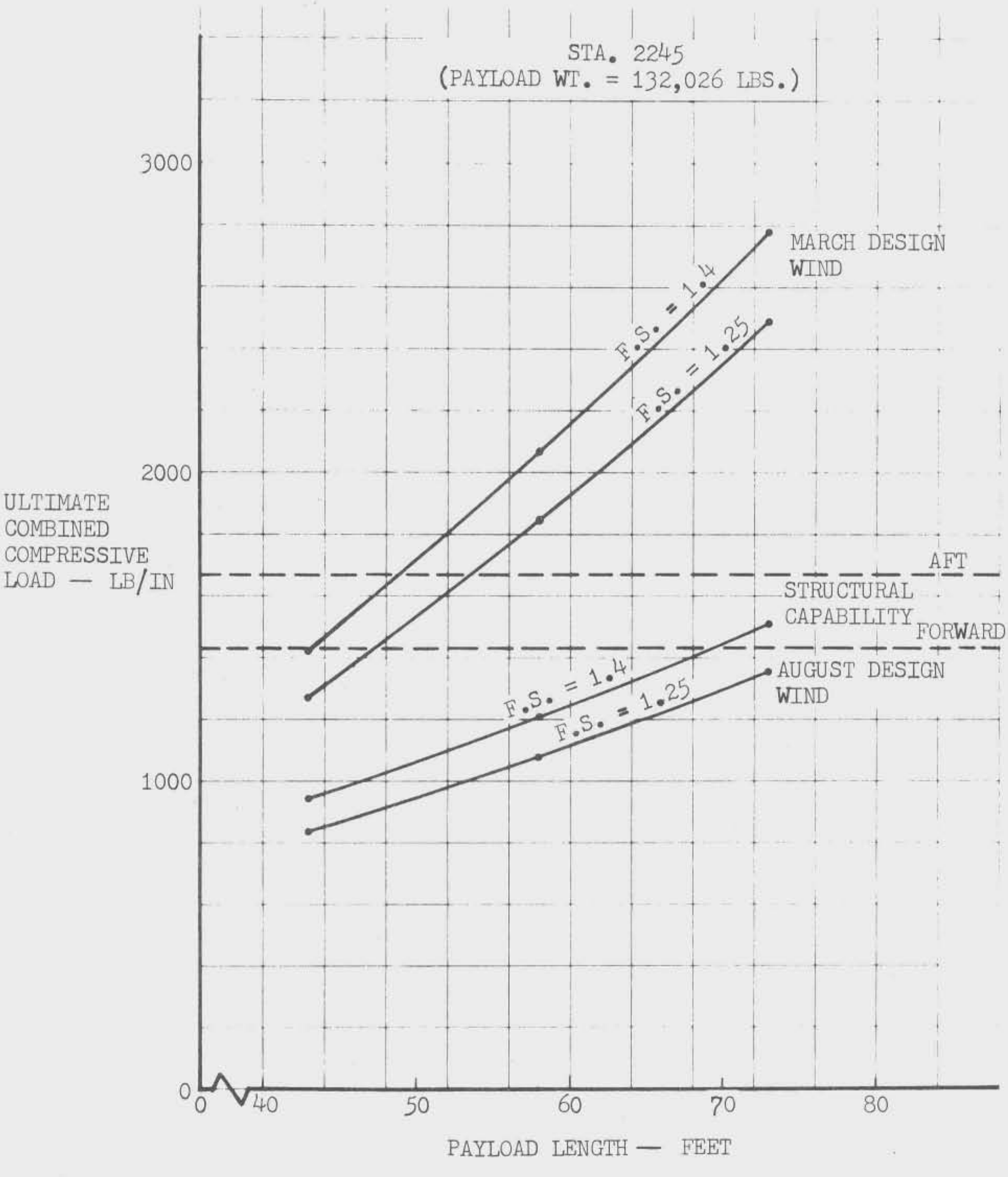


FIGURE 4.3.1.3-13 MAX (Qα) N_C ULTIMATE @ STA 2245 VERSUS TOTAL PAYLOAD LENGTH FOR THE INT-20 VEHICLE WITH A 132,026 LB. PAYLOAD

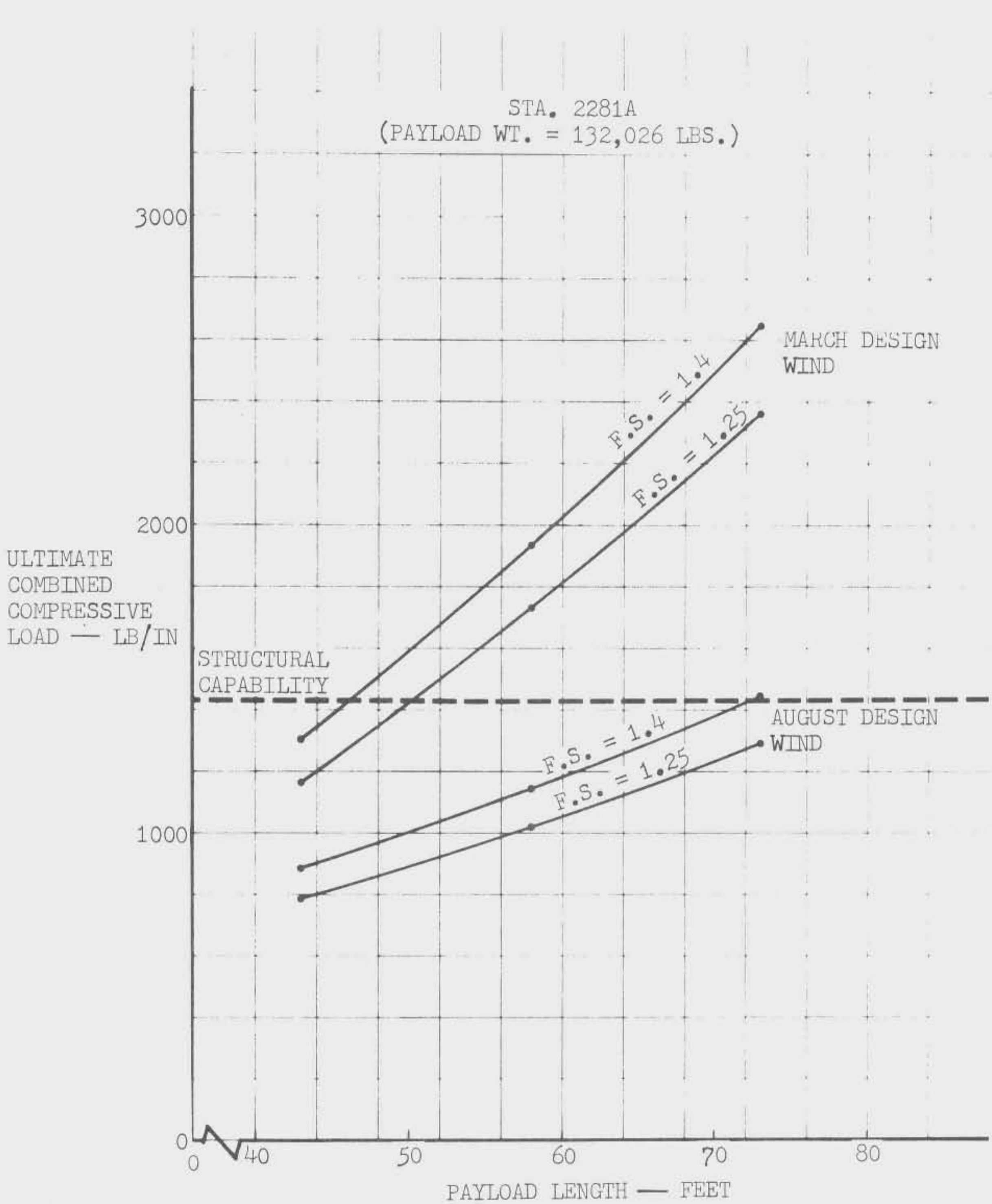


FIGURE 4.3.1.3-14 MAX ($Q\alpha$) N_C ULTIMATE @ STA 2281A VERSUS TOTAL PAYLOAD LENGTH FOR THE INT-20 VEHICLE WITH A 132,026 LB. PAYLOAD

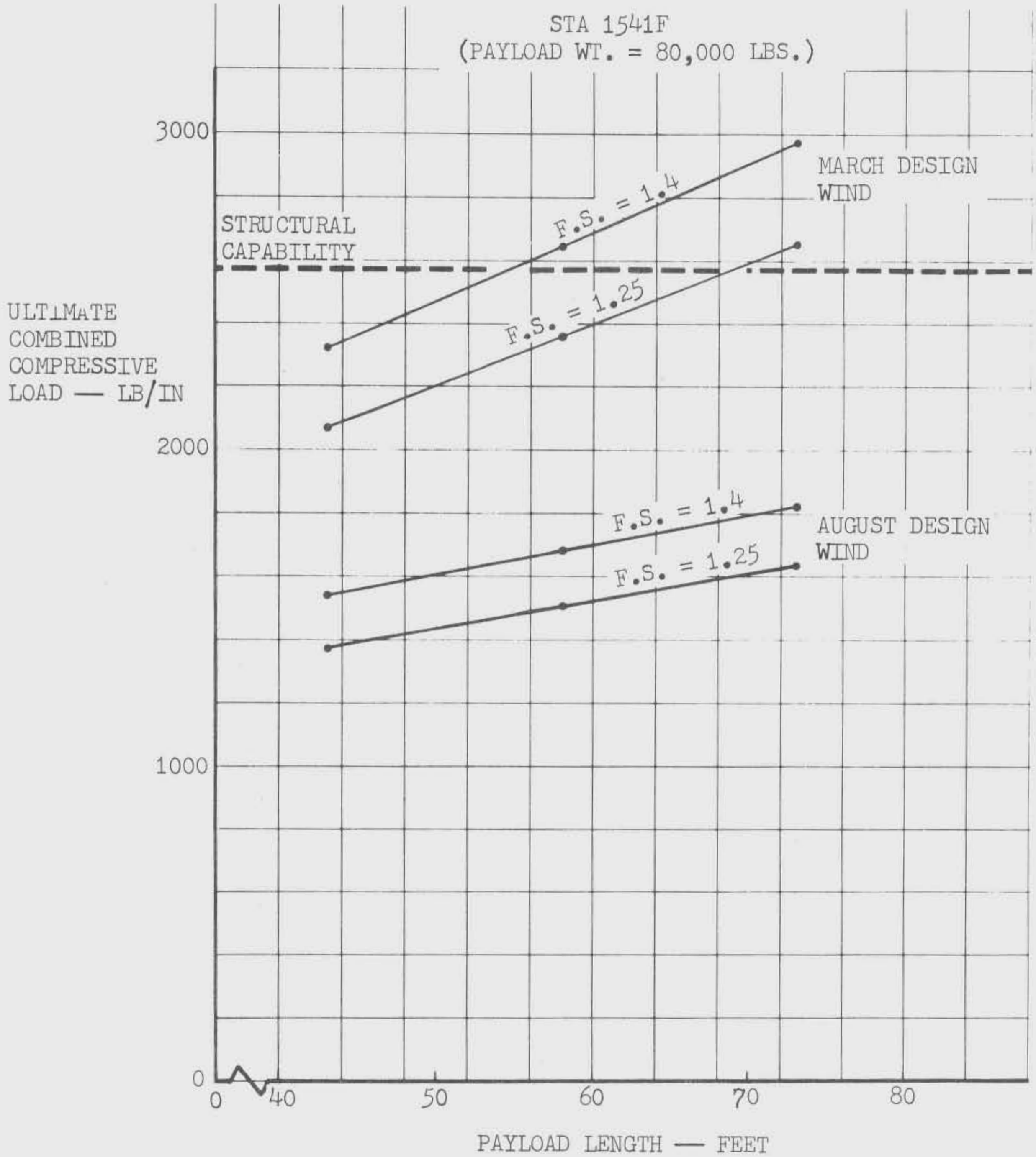


FIGURE 4.3.1.3-15 MAX (Qα) N_C ULTIMATE @ STA 1541F VERSUS TOTAL PAYLOAD LENGTH FOR THE INT-20 VEHICLE WITH A 80,000 LB. PAYLOAD

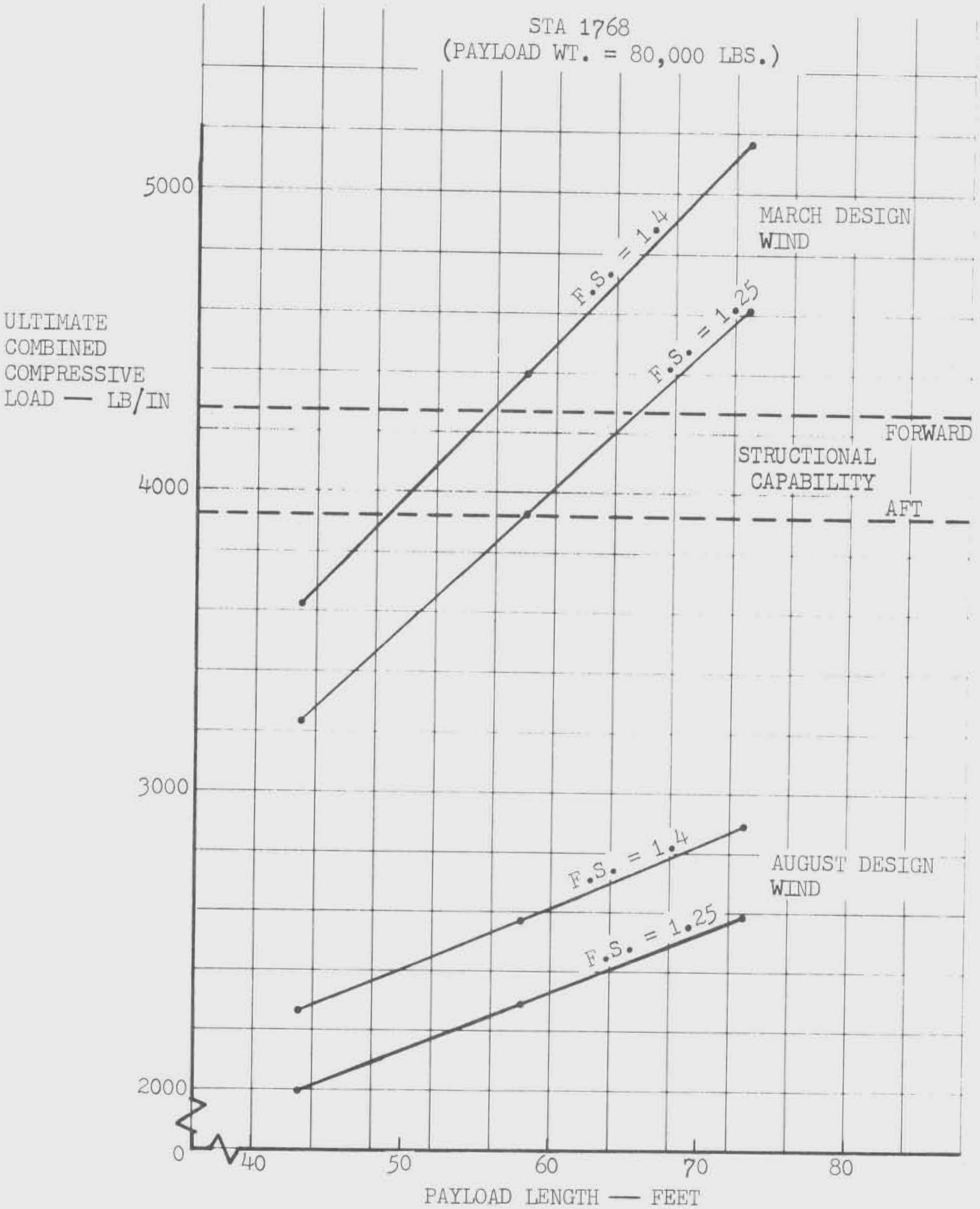


FIGURE 4.3.1.3-16 MAX ($Q\alpha$) N_C ULTIMATE @ STA 1768 VERSUS TOTAL PAYLOAD LENGTH FOR THE INT-20 VEHICLE WITH A 80,000 LB. PAYLOAD

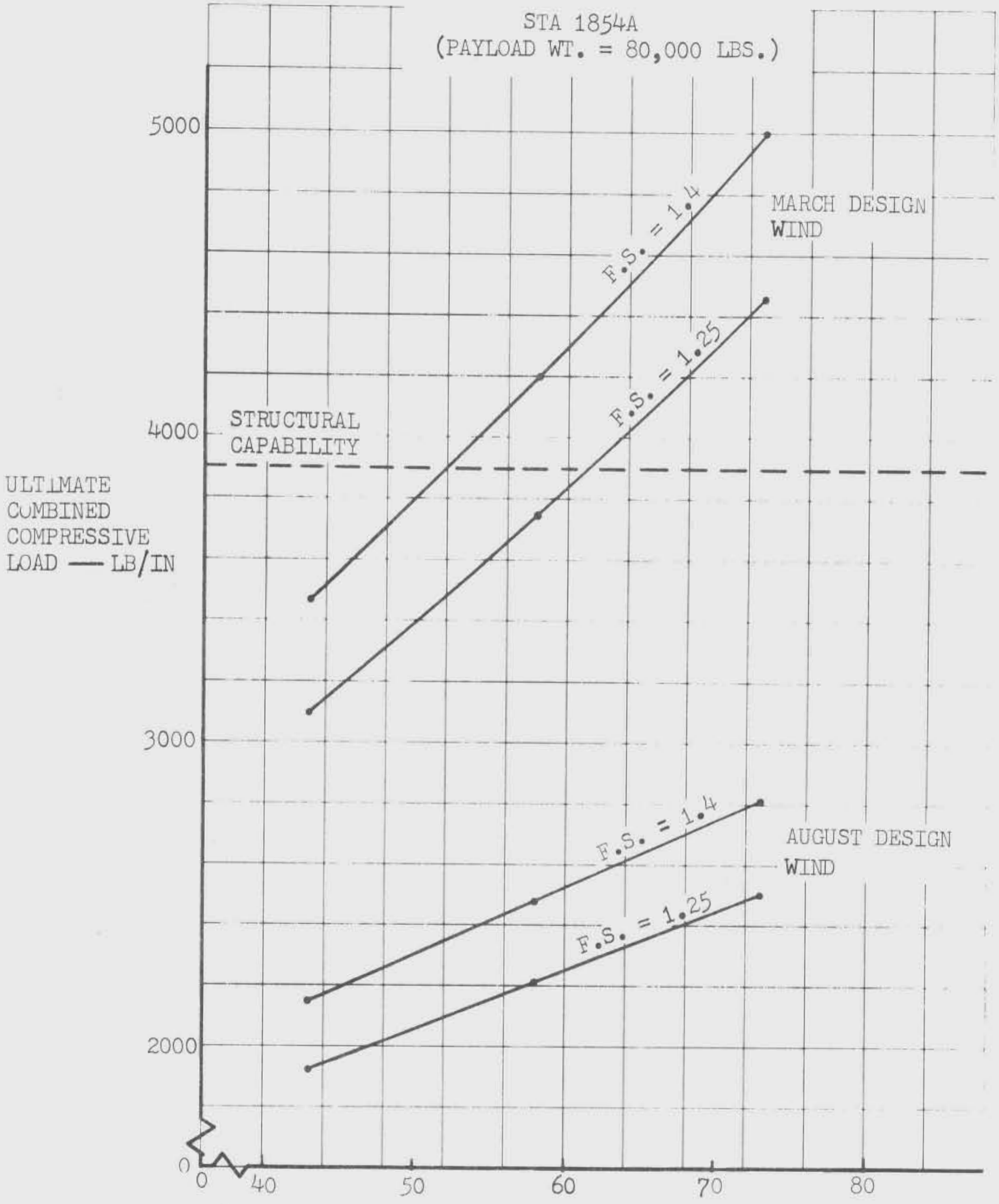


FIGURE 4.3.1.3-17 MAX $(Q\alpha)$ N_C ULTIMATE @ STA 1854A VERSUS TOTAL PAYLOAD LENGTH FOR THE INT-20 VEHICLE WITH A 80,000 LB. PAYLOAD

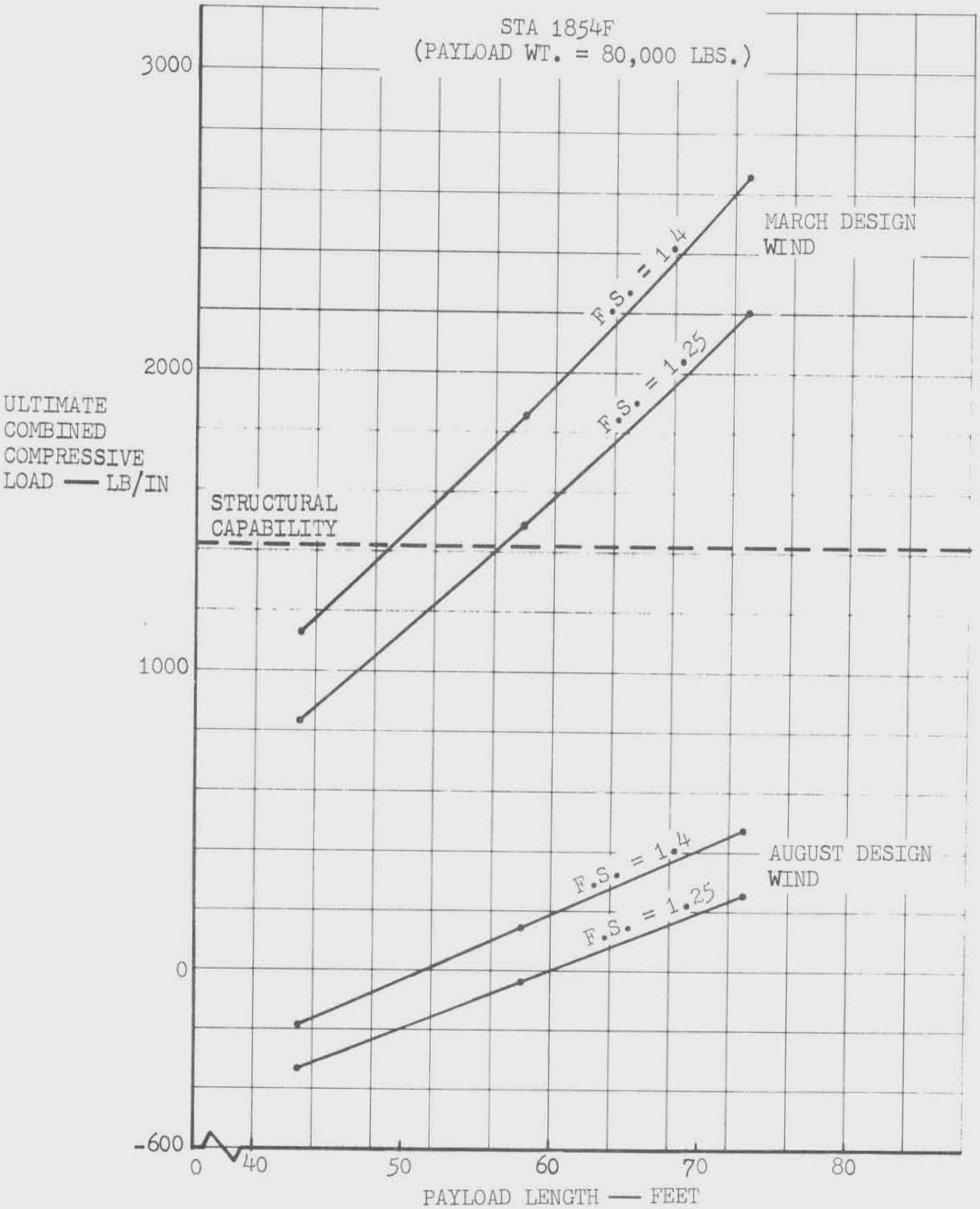


FIGURE 4.3.1.3-18 MAX (Qα) N_C ULTIMATE @ STA 1854F VERSUS TOTAL PAYLOAD LENGTH FOR THE INT-20 VEHICLE WITH A 80,000 LB. PAYLOAD

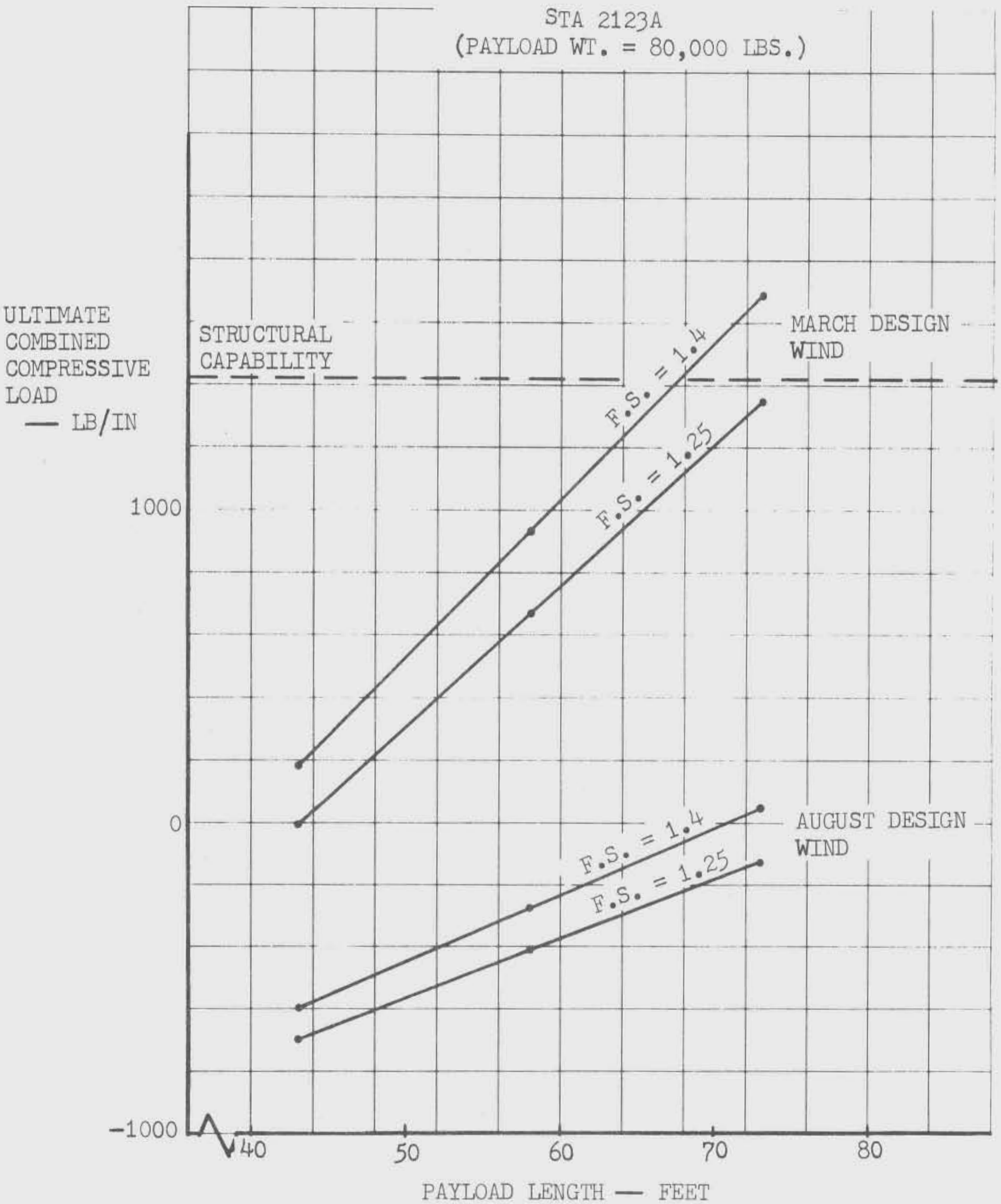


FIGURE 4.3.1.3-19 MAX (Qα) N_C ULTIMATE @ STA 2123A VERSUS TOTAL PAYLOAD LENGTH FOR THE INT-20 VEHICLE WITH A 80,000 LB. PAYLOAD

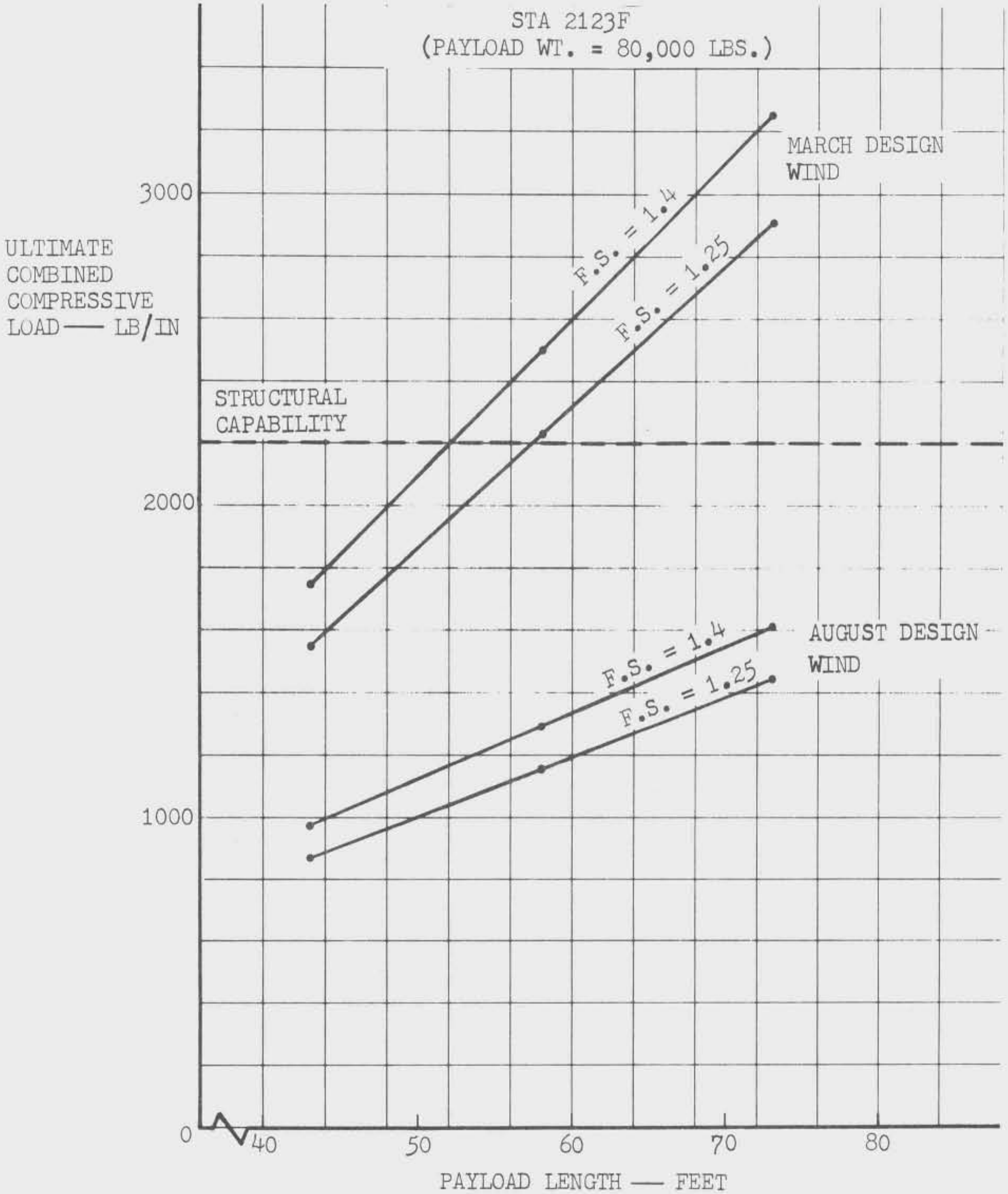


FIGURE 4.3.1.3-20 MAX ($Q\alpha$) N_C ULTIMATE @ STA 2123F VERSUS TOTAL PAYLOAD LENGTH FOR THE INT-20 VEHICLE WITH A 80,000 LB. PAYLOAD

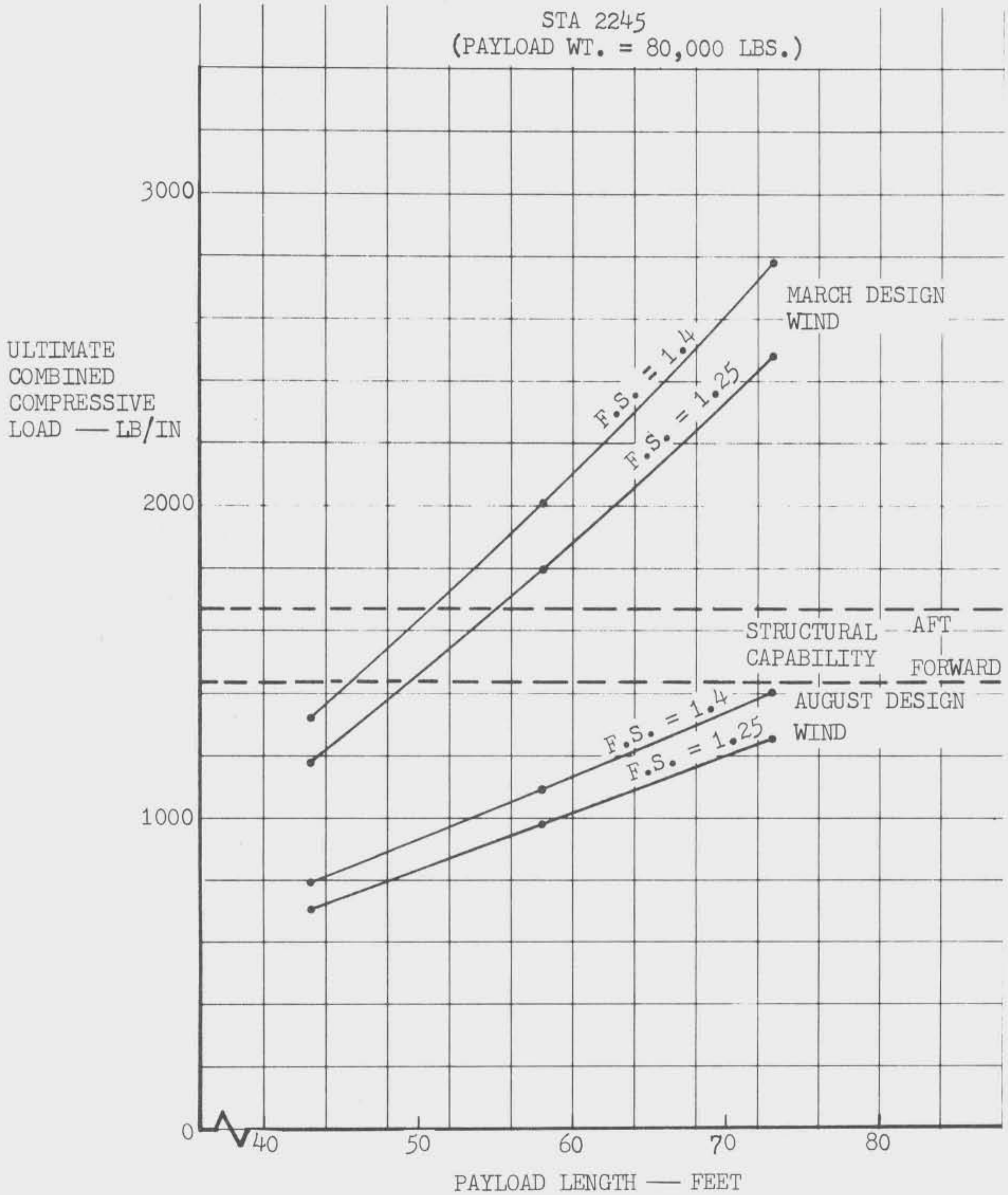


FIGURE 4.3.1.3-21 MAX ($Q\alpha$) N_C ULTIMATE @ STA 2245 VERSUS TOTAL PAYLOAD LENGTH FOR THE INT-20 VEHICLE WITH A 80,000 LB. PAYLOAD

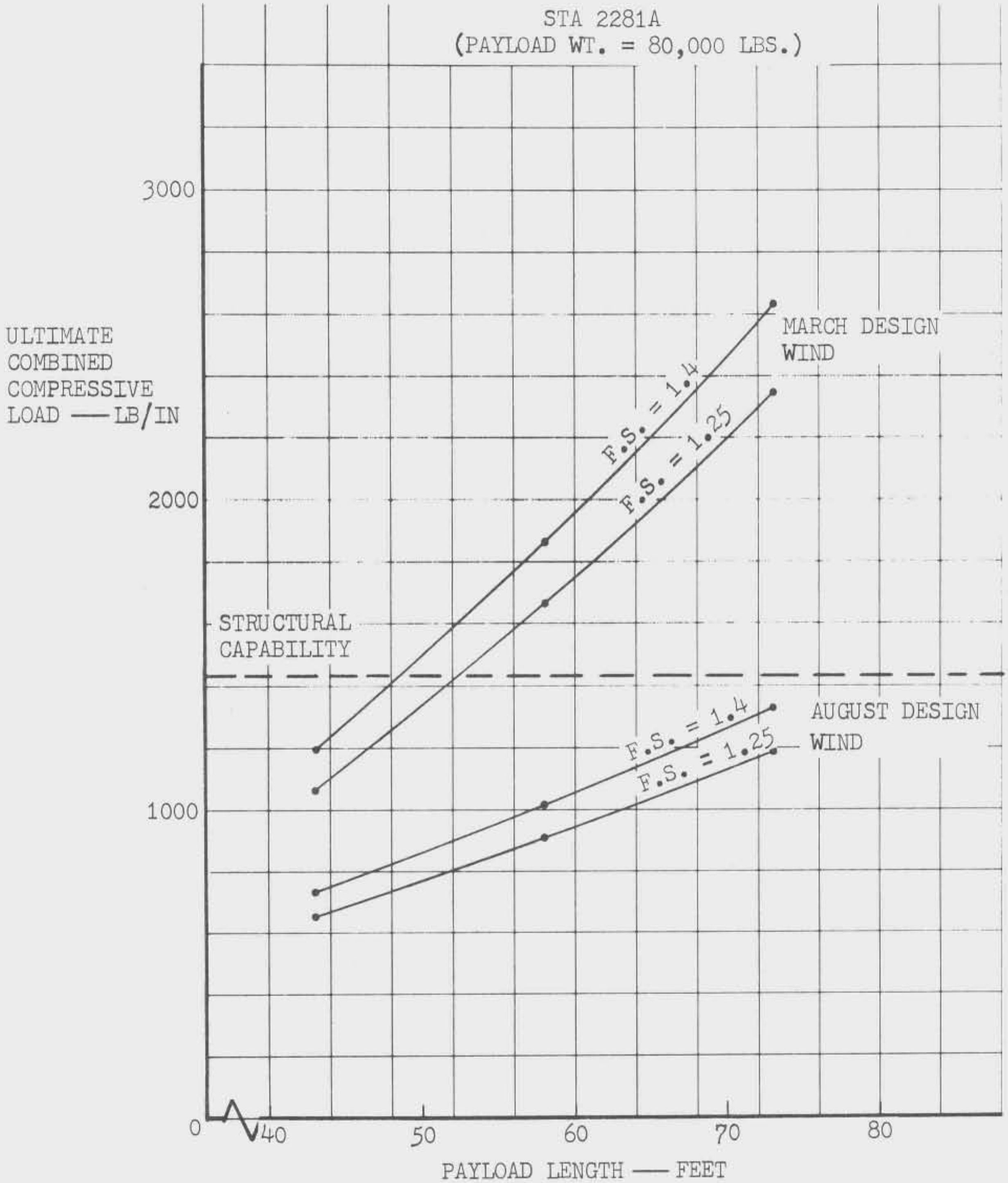


FIGURE 4.3.1.3-22 MAX ($Q\alpha$) N_C ULTIMATE @ STA 2281A VERSUS TOTAL PAYLOAD LENGTH FOR THE INT-20 VEHICLE WITH A 80,000 LB. PAYLOAD

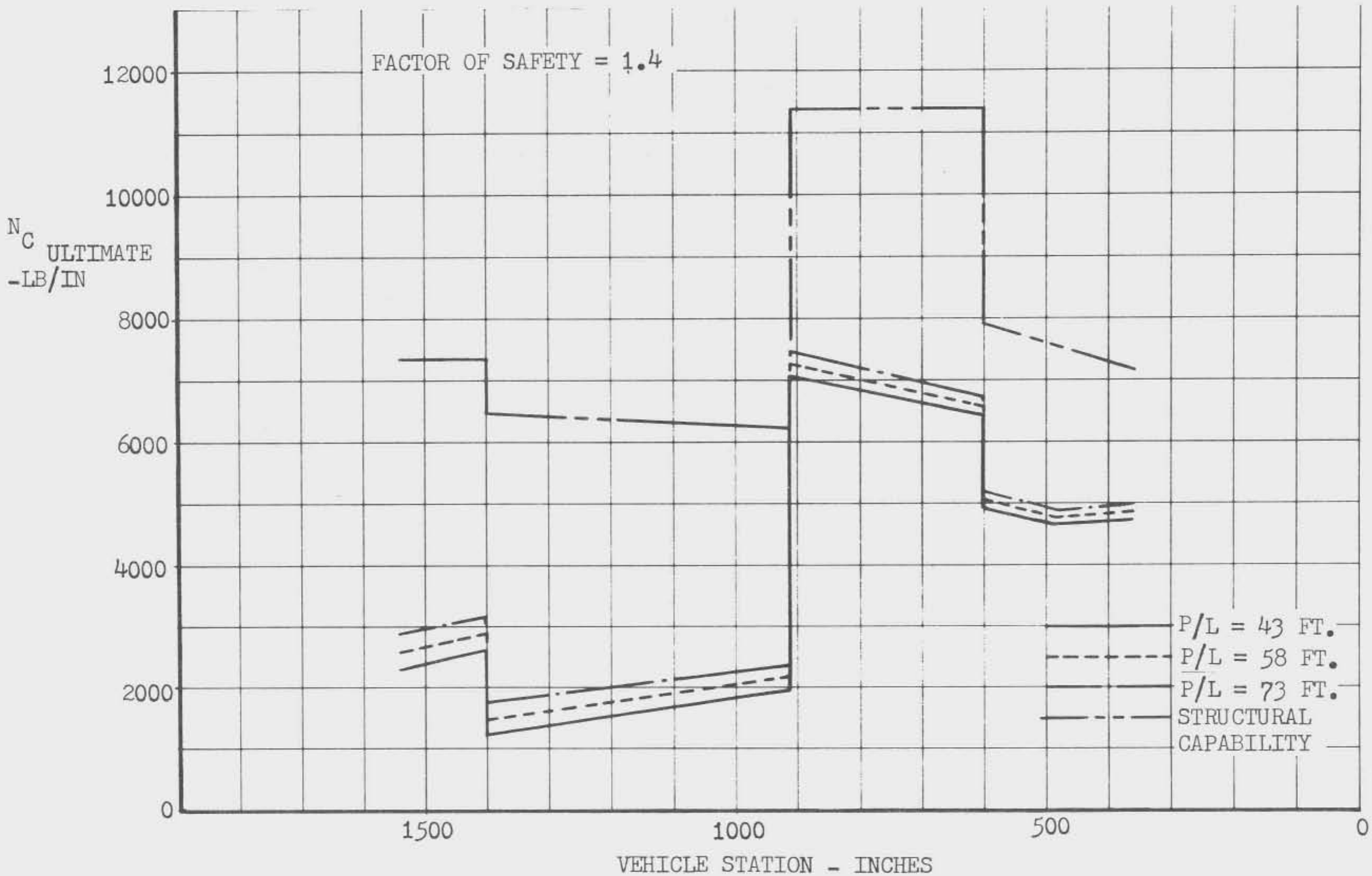


FIGURE 4.3.1.3-23 S-IC N_C ULTIMATE VERSUS VEHICLE STATION FOR THE INT-20 VEHICLE WITH A 132,026 LB. PAYLOAD (MARCH 95% DESIGN WIND)

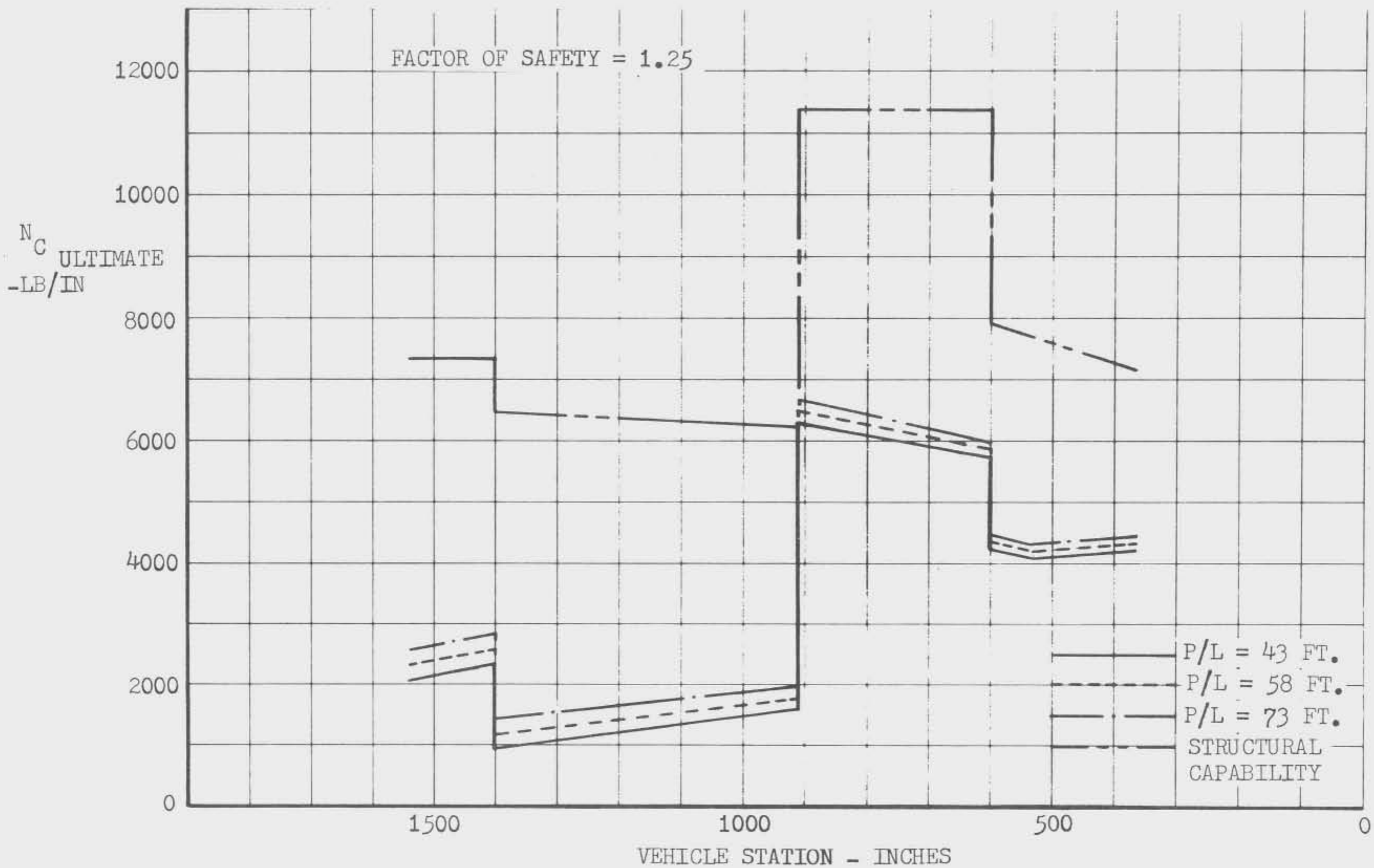


FIGURE 4.3.1.3-24 S-IC N_C ULTIMATE VERSUS VEHICLE STATION FOR THE INT-20 VEHICLE WITH A 132,026 LB. PAYLOAD (MARCH 95% DESIGN WIND)

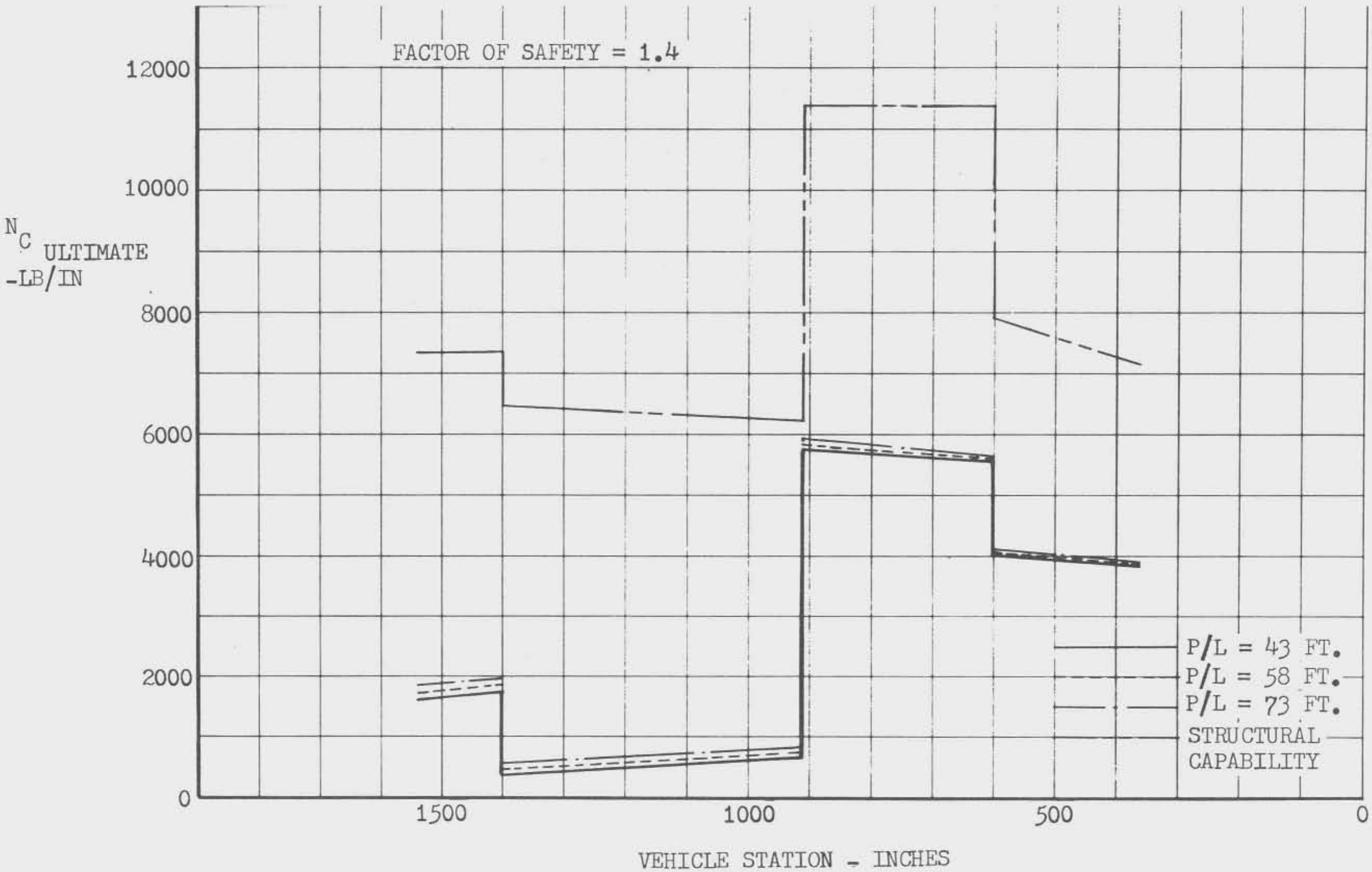


FIGURE 4.3.1.3-25 S-IC N_C ULTIMATE VERSUS VEHICLE STATION FOR THE INT-20 VEHICLE WITH A 132,026 LB. PAYLOAD (AUGUST 95% DESIGN WIND)

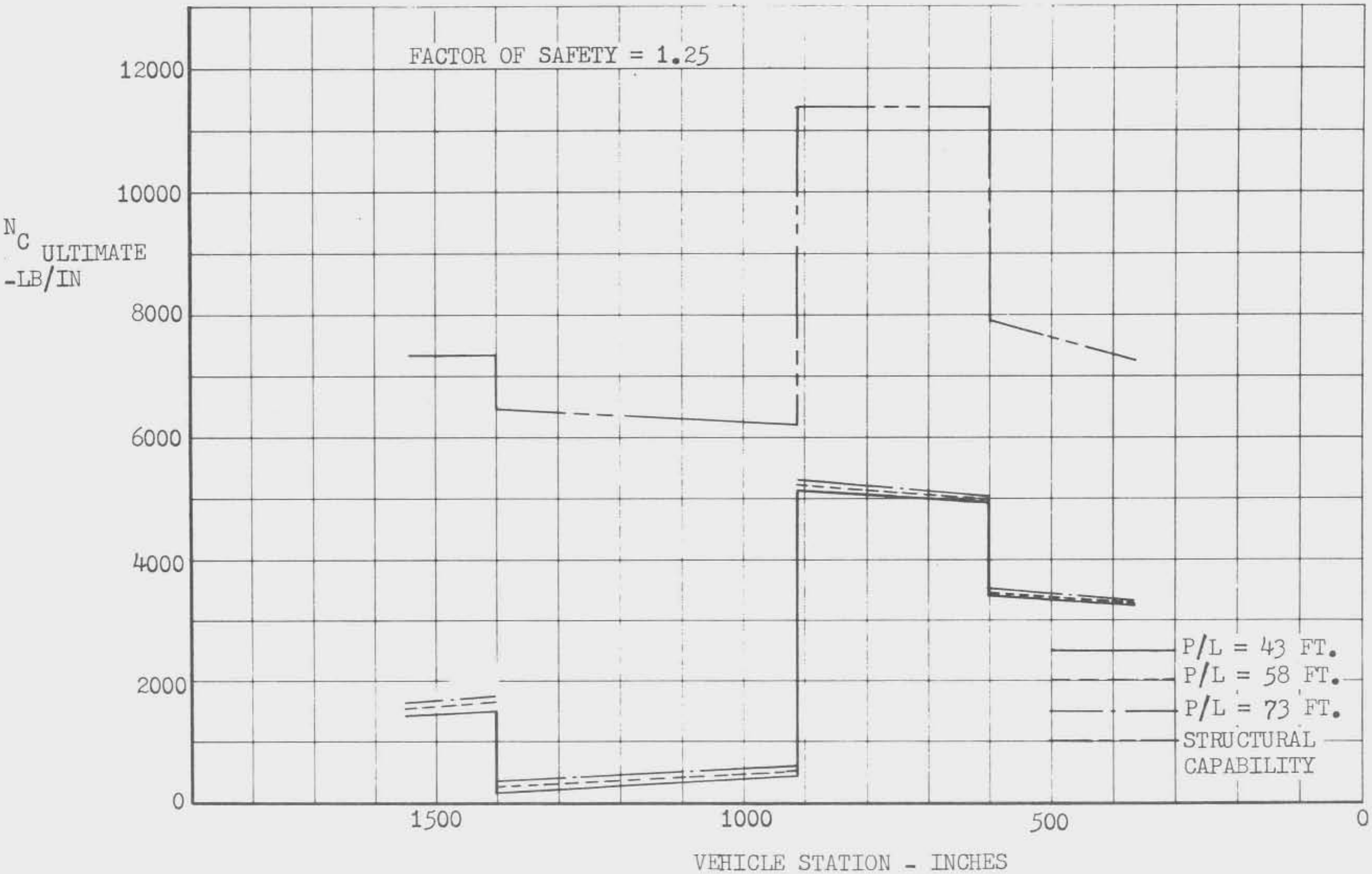


FIGURE 4.3.1.3-26 S-IC N_C ULTIMATE VERSUS VEHICLE STATION FOR THE INT-20 VEHICLE WITH A 132,026 LB. PAYLOAD (AUGUST 95% DESIGN WIND)

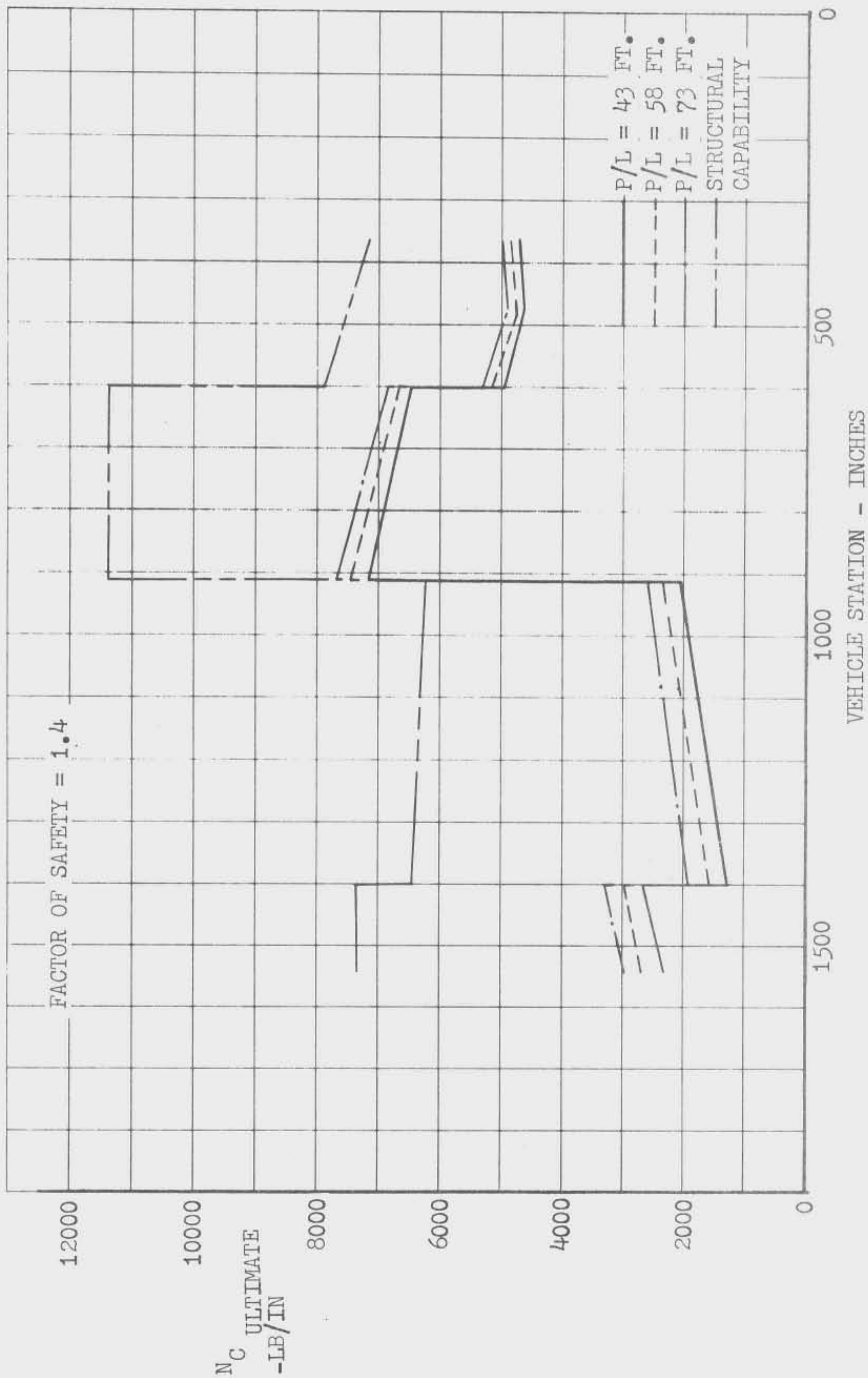


FIGURE 4.3.1.3-27 S-IC N_C ULTIMATE VERSUS VEHICLE STATION FOR THE INT-20 VEHICLE WITH A 80,000 LB. PAYLOAD (MARCH 95% DESIGN WIND)

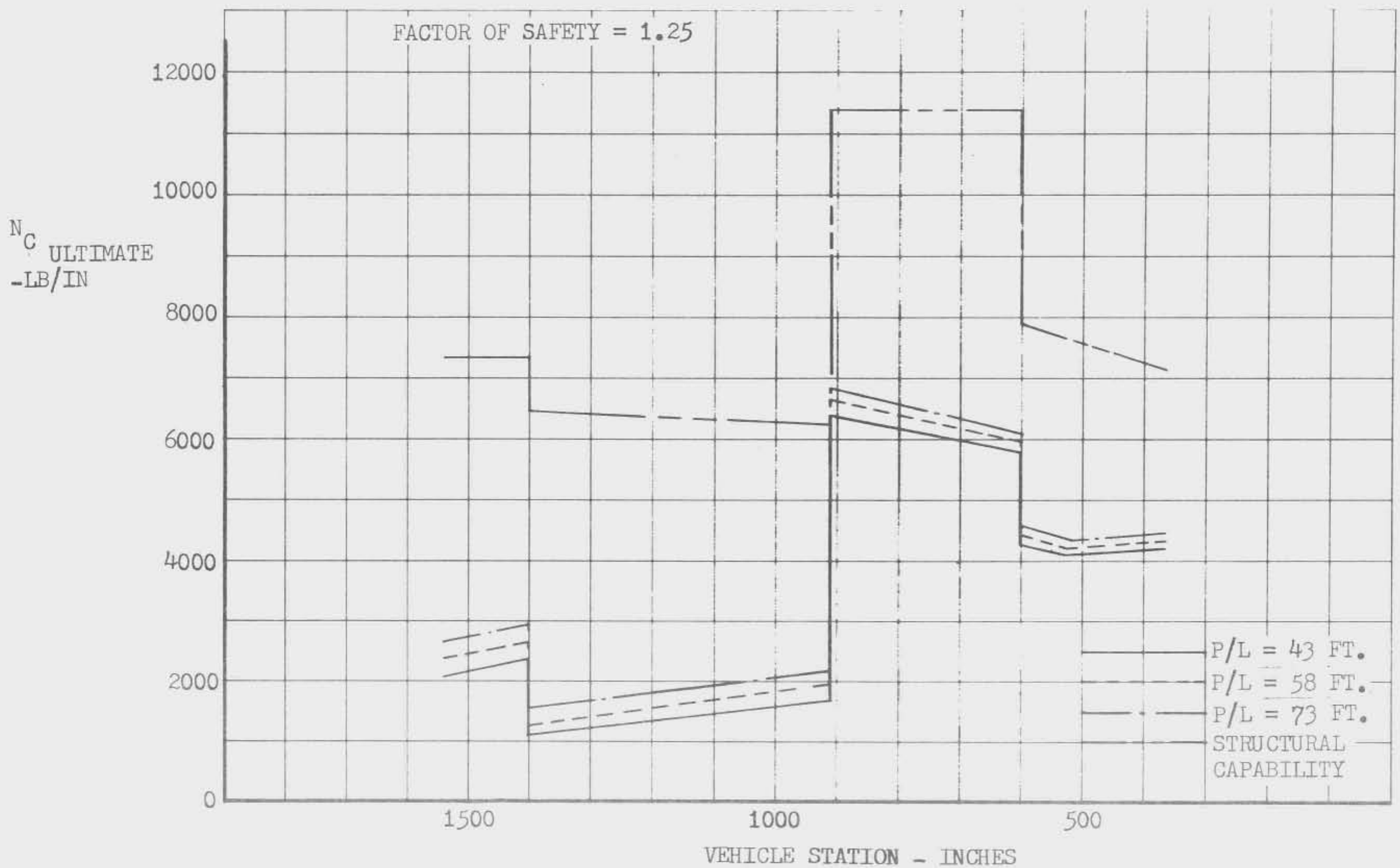


FIGURE 4.3.1.3-28 S-IC N_C ULTIMATE VERSUS VEHICLE STATION FOR THE INT-20 VEHICLE WITH A 80,000 LB. PAYLOAD (MARCH 95% DESIGN WIND)

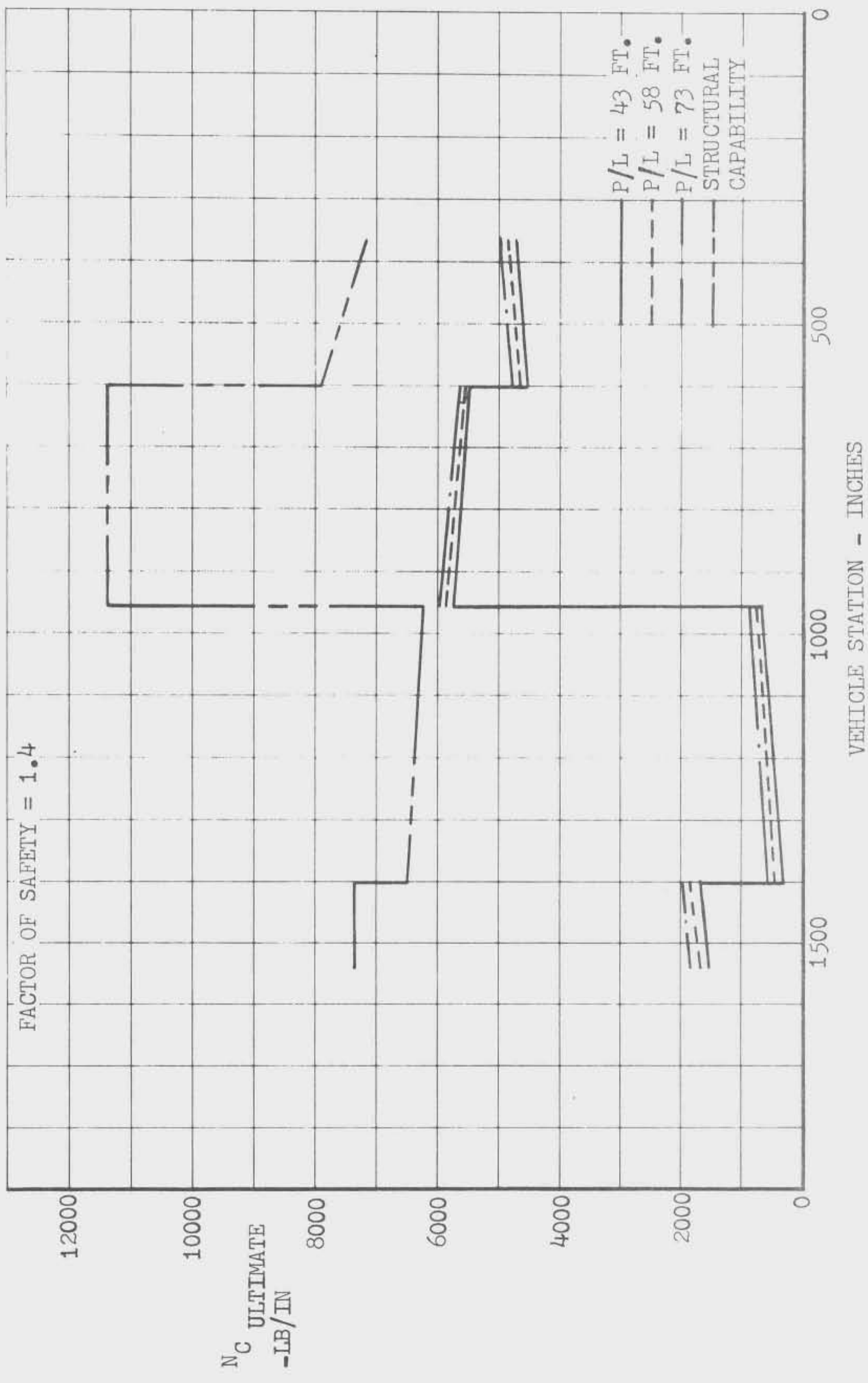


FIGURE 4.3.1.3-29 S-IC N_C ULTIMATE VERSUS VEHICLE STATION FOR THE INT-20 VEHICLE WITH A 80,000 LB. PAYLOAD (AUGUST 95% DESIGN WIND)

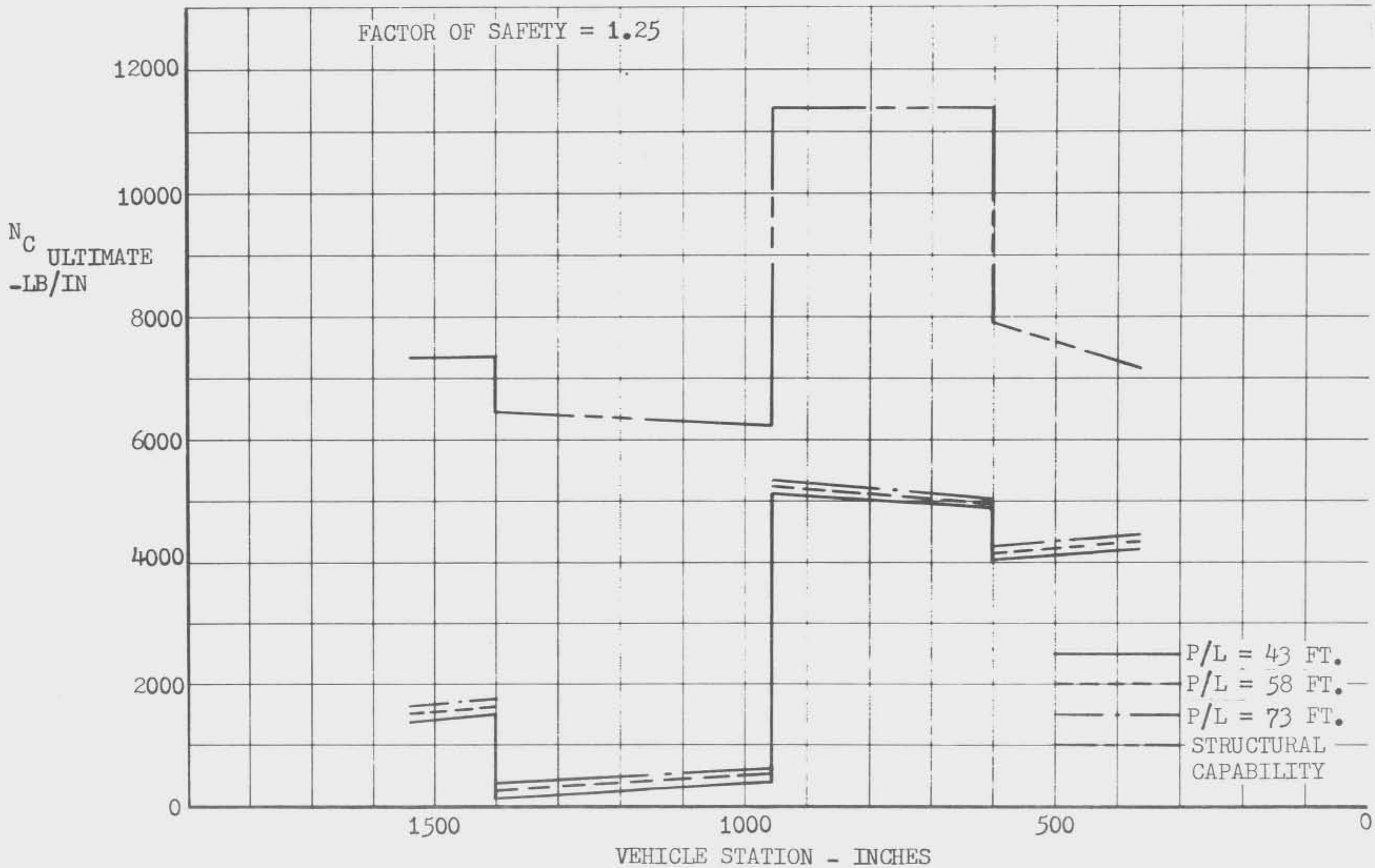


FIGURE 4.3.1.3-30 S-IC N_C ULTIMATE VERSUS VEHICLE STATION FOR THE INT-20 VEHICLE WITH A 80,000 LB. PAYLOAD (AUGUST 95% DESIGN WIND)

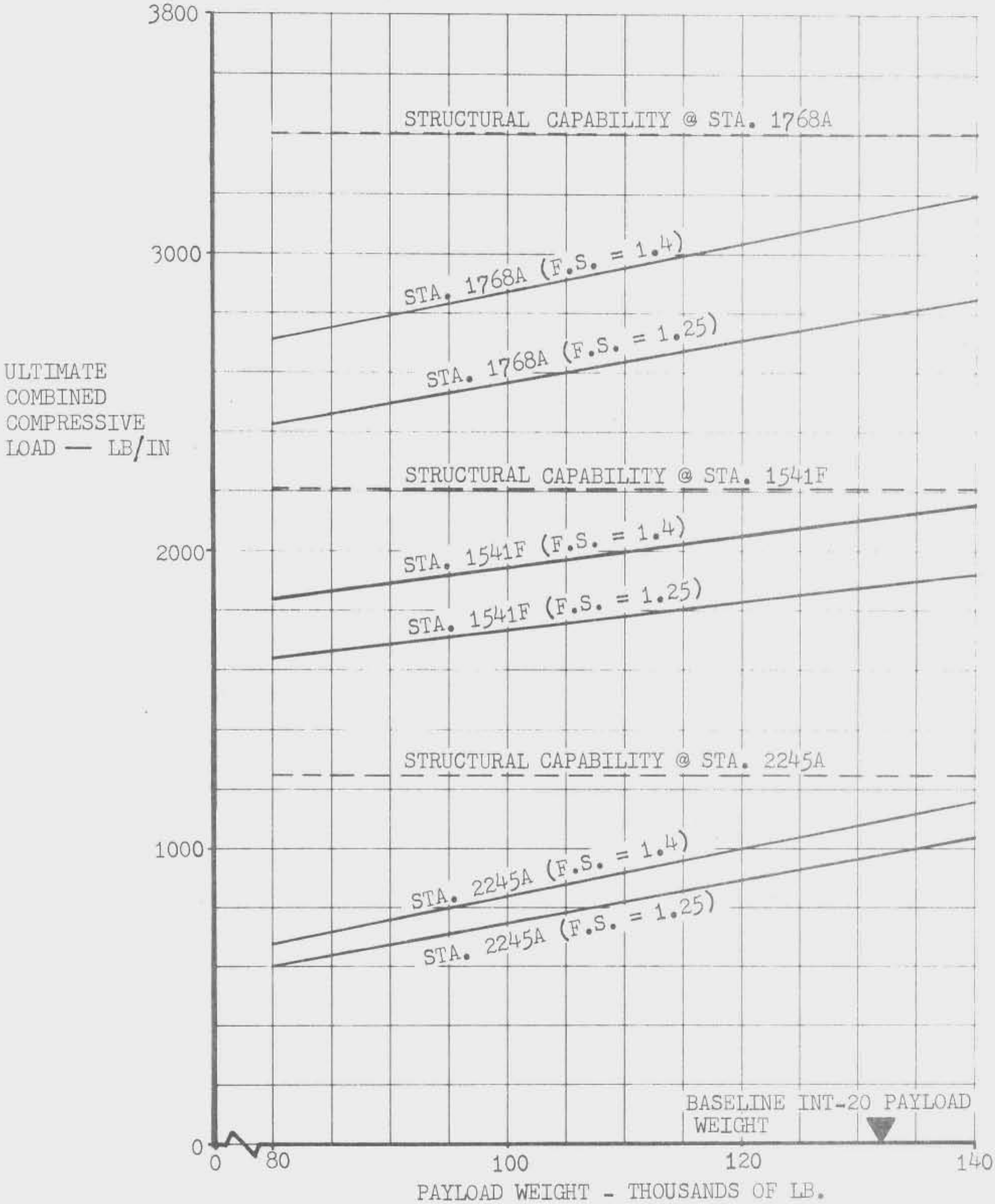


FIGURE 4.3.1.3-31 MAXIMUM ACCELERATION N_C ULTIMATE VERSUS PAYLOAD WEIGHT FOR CRITICAL VEHICLE STATIONS

TABLE 4.3.1.3-I

ALLOWABLE PAYLOAD LENGTHS FOR THE INT-20 VEHICLE WITH A 132,026 LB. PAYLOAD

VEHICLE STATION (IN)	FACTOR OF SAFETY	ALLOWABLE PAYLOAD LENGTH FT.	
		MARCH 75M/S WIND	AUGUST 22M/S WIND
2281 AFT	1.4	46.0	72.4
2245 FWD	1.4	43.2	69.6
2245 AFT	1.4	48.9	> 73.0
2123 FWD	1.4	51.8	> 73.0
2123 AFT	1.4	68.4	> 73.0
1854 FWD	1.4	50.3	> 73.0
1854 AFT	1.4	53.2	> 73.0
1768 FWD	1.4	58.4	> 73.0
1768 AFT	1.4	51.0	> 73.0
1541 FWD	1.4	57.8	> 73.0
2281 AFT	1.25	50.2	> 73.0
2245 FWD	1.25	47.6	> 73.0
2245 AFT	1.25	53.6	> 73.0
2123 FWD	1.25	57.4	> 73.0
2123 AFT	1.25	> 73.0	> 73.0
1854 FWD	1.25	58.0	> 73.0
1854 AFT	1.25	63.2	> 73.0
1768 FWD	1.25	69.8	> 73.0
1768 AFT	1.25	61.2	> 73.0
1541 FWD	1.25	> 73.0	> 73.0

NOTE: > 73.0 means that the allowable payload length is greater than 73.0 ft.

TABLE 4.3.1.3-II

ALLOWABLE PAYLOAD LENGTHS FOR THE INT-20 VEHICLE WITH AN 80,000 LB. PAYLOAD

VEHICLE STATION (IN)	FACTOR OF SAFETY	ALLOWABLE PAYLOAD LENGTH-FT.	
		MARCH 75 M/S WIND	AUGUST 22 M/S WIND
2281 AFT	1.4	48.7	> 73.0
2245 FWD	1.4	45.6	> 73.0
2245 AFT	1.4	50.8	> 73.0
2123 FWD	1.4	52.0	> 73.0
2123 AFT	1.4	67.6	> 73.0
1854 FWD	1.4	49.2	> 73.0
1854 AFT	1.4	52.0	> 73.0
1768 FWD	1.4	55.5	> 73.0
1768 AFT	1.4	48.7	> 73.0
1541 FWD	1.4	54.6	> 73.0
2281 AFT	1.25	52.5	> 73.0
2245 FWD	1.25	49.4	> 73.0
2245 AFT	1.25	55.0	> 73.0
2123 FWD	1.25	57.3	> 73.0
2123 AFT	1.25	> 73.0	> 73.0
1854 FWD	1.25	56.5	> 73.0
1854 AFT	1.25	61.4	> 73.0
1768 FWD	1.25	65.5	> 73.0
1768 AFT	1.25	57.8	> 73.0
1541 FWD	1.25	68.9	> 73.0

NOTE: >73.0 means that the allowable payload length is greater than 73.0 ft.

TABLE 4.3.1.3-III

SUMMARY OF ALLOWABLE PAYLOAD LENGTHS FOR THE 132,026 LB. PAYLOAD VEHICLE AND THE 80,000 LB. PAYLOAD VEHICLE

PAYLOAD WT. (LB)	FACTOR OF SAFETY	ALLOWABLE PAYLOAD LENGTH ~ FT. (NO STRUCTURAL MODIFICATIONS)		ALLOWABLE PAYLOAD LENGTH ~ FT. (STRUCTURAL MODIFICATIONS TO I.U.)	
		MARCH 75 M/S WIND	AUGUST 22M/S WIND	MARCH 75M/S WIND	AUGUST 22M/S WIND
132,026	1.4	43.2	69.6	48.9	>73.0
132,026	1.25	47.6	>73.0	53.6	>73.0
80,000	1.4	45.6	>73.0	48.7	>73.0
80,000	1.25	49.4	>73.0	55.0	>73.0

TABLE 4.3.1.3-IV

LIMITING ACCELERATIONS AND FACTORS OF SAFETY AT CRITICAL VEHICLE STATIONS

CRITICAL VEHICLE STATIONS (IN)	MAXIMUM OBTAINABLE FACTOR OF SAFETY FOR 4.68 g's AND 132,026 LB. PAYLOAD	MAXIMUM ALLOWABLE PAYLOAD WT. FOR F.S.=1.4, AND 4.68 g's (LB)	LIMITING ACCELERATION FOR F.S. = 1.4 AND 132,026 LB. PAYLOAD g's
1541F	1.464	150,474	4.895
1768A	1.522	166,012	5.088
2245A	1.602	151,685	5.355
2245F	1.794	170,390	5.998

4.3.2 INT-20/Big G Analysis

An alternate configuration of the INT-20, with a Big Gemini (Big G) logistics payload (defined in Reference 4.3.2-1) was studied.

a. Trajectory

The mission flown was direct injection with the INT-20 booster into a 100 x 270 nautical mile, 50^{o} inclined elliptical orbit; launch was assumed to be from the AMR with a launch azimuth of 44.5^{o} and a liftoff thrust to weight ratio of 1.25. The vehicle configuration is shown in Figure 4.3.2-1. The trajectory assumed a northerly, coplanar boost, resulting in a vehicle impact trace as presented in Figure 4.3.2-2. Assumed aerodynamic characteristics are presented in Figure 4.3.2-3 and in Appendix D.3.2. LES weights and the ballast required to remain within the 4.68 g acceleration limit are assumed to be staged with the S-IC. As shown in Table 4.3.2-I, this INT-20/Big G vehicle has a net payload capability of 117,300 pounds (53,206 kg). The trajectory print-out for this mission is contained in Appendix D.2.

For comparative purposes, the INT-20/Big G configuration was flown with a southerly launch, employing optimum boost turning and launch azimuth to avoid the South America land mass. A vehicle definition of this comparative vehicle is presented in Table 4.3.2-II, and its associated impact trace is shown in Figure 4.3.2-4. As is shown in the yaw history presented in 4.3.2-5, the majority of the yaw was accomplished after calculus of variations (COV) was initiated in the trajectory; this corresponded to a dynamic pressure of less than 50 psf. The resulting payload for this southerly launch, boost turning INT-20/ Big G configuration is 64,600 pounds (29,300 kg).

b. Weights

The INT-20/Big G vehicle distributed and accumulative weights are shown in Appendix D.4.

c. Ground and flight loads data for the INT-20/Big G configuration are as shown below:

1. The ground and inflight wind environments which were used in the calculation of the respective bending moment distributions were obtained by using MSFC design wind criteria and the methods given in Reference 4.1.4.5-1. The inflight wind profile was obtained from a 99 percent shear build-up reduced 15 percent to a 95 percent peak wind speed at 10,000 meters altitude. An embedded jet gust, reduced in magnitude 15 percent, was imposed upon the peak of the wind profile. The inflight wind profile is shown in Section 4.1.6.1.

4.3.2 (Continued)

2. Bending Moment Distributions

The ground wind bending moment distributions for a 99.9 percent prelaunch wind and a 99 percent launch wind are shownin Figure 4.3.2-6.

The maximum inflight bending moment distribution was determined from a flight simulation of the INT-20/Big G vehicle during first stage boost using MSFC design wind criteria in the yaw plane. Included in the flight simulation were rigid body translation and rotation in the yaw plane, one free-free bending mode and two nozzle degrees of freedom. Figure 4.3.2-7 presents the maximum inflight bending moment envelope.

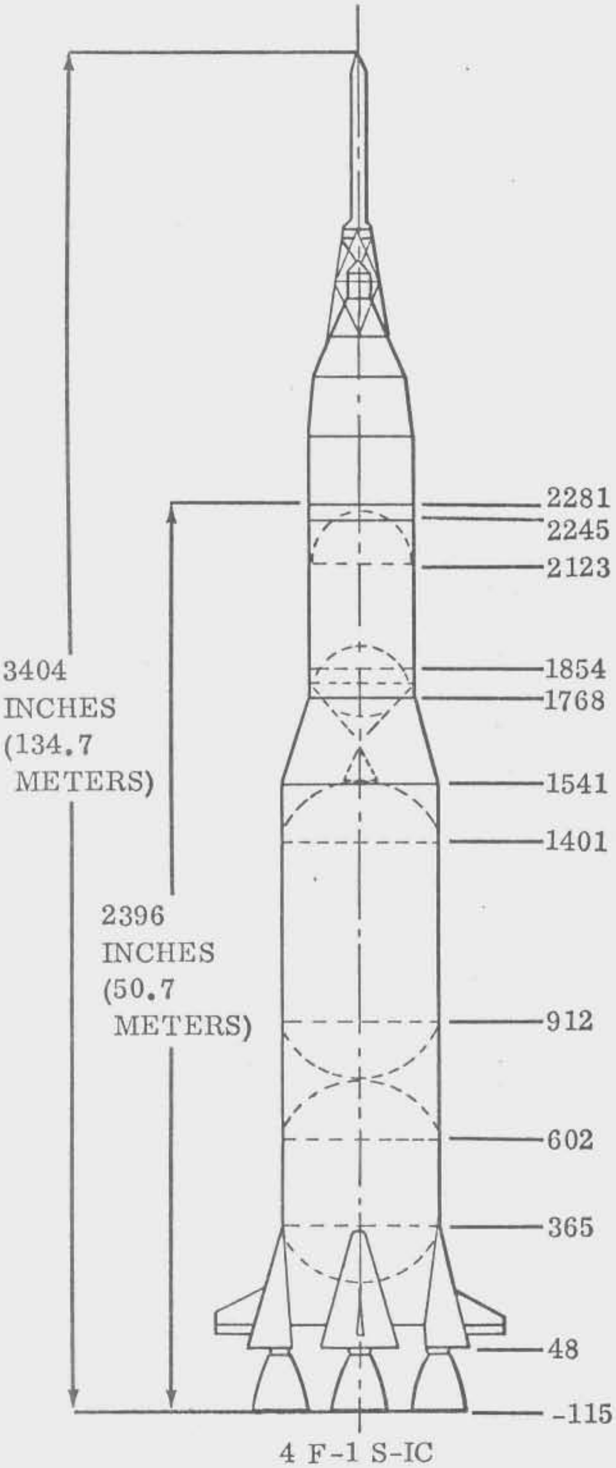
3. Longitudinal Force Distribution

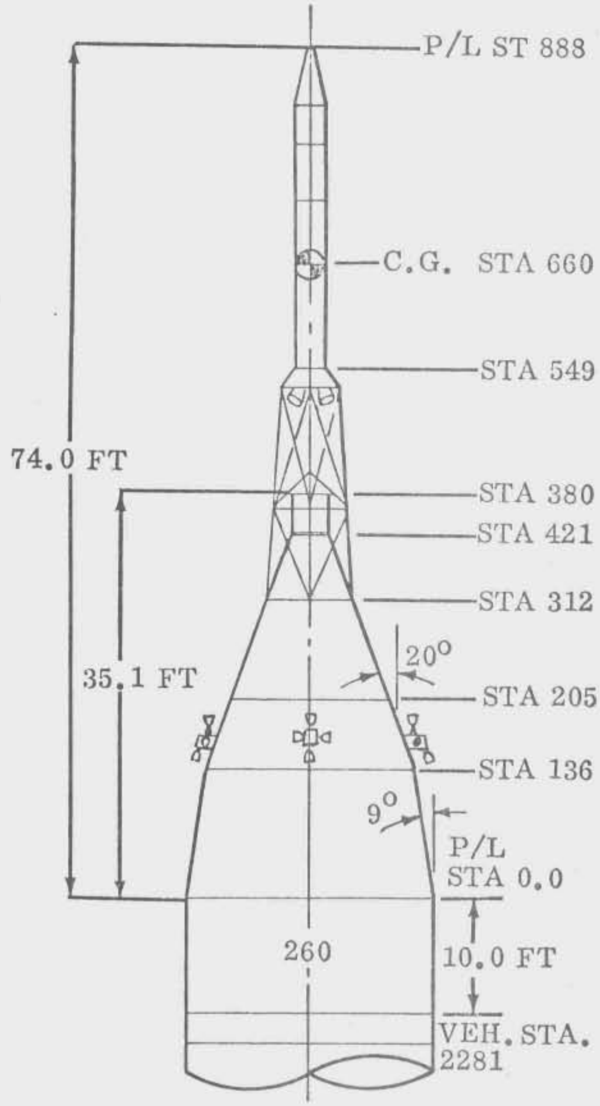
Longitudinal force distributions for the critical design conditions were calculated using the method shown in Section 4.1.6 and are shown in Figures 4.3.2-8 through 4.3.2-11.

4. Combined Loads

Ultimate compressive and tensile loads were calculated using the method shownin Section 4.1.6. Plots of ultimate combined compressive loading for the S-IC stage and the S-IVB stage and IU are given in Figure 4.3.2-12 and Figure 4.3.2-13, respectively. Tabulations of these loads are included in Tables 4.3.2-IV through 4.3.2-VII.

Figures 4.3.2-14 and 4.3.2-15 show the ultimate tensile combined loads for the S-IC stage and the S-IVB stage and IU and tabulations of these loads are given in Tables 4.3.2-VIII through 4.3.2-XI.





BIG "G" CONFIGURATION

NOTES:

1. ALL DIMENSIONS IN INCHES EXCEPT AS NOTED.
2. ALL DIMENSIONS ARE APPROXIMATE

FIGURE 4.3.2-1 INT-20/BIG "G" LOGISTICS VEHICLE CONFIGURATION

TABLE 4.3.2-I

30506 - INT-20/BIG G
100 x 270 N.M., 50° ORBIT
COPLANAR, NORTHERLY LAUNCH

FIRST STAGE OPERATION		
Lift-Off Weight	lbs	4,870,400
T/W Ratio		1.25
Sea Level Thrust	lbs	6,088,000
Sea Level I_{SP}	sec	263.58
Liquid Propellant Consumed	lbs	4,122,325
Stage Weight @ Staging	lbs	332,635
Ballast & LES	lbs	36,433
Max. Dynamic Pressure	lbs/ft^2	715
Lift-Off Azimuth Angle	degs	44.5
SECOND STAGE OPERATION		
Thrust (VAC)	lbs	205,000
I_{SP} (Nominal)	sec	426
Weight @ Ignition	lbs	379,007
Propellant Capacity	lbs	230,000
Propellant Consumed	lbs	227,500
Stage Separation Weight	lbs	27,504
Gross Payload	lbs	124,003
Astrionics Equipment	lbs	4,183
Flight Performance Reserves	lbs	2,500
Net Payload	lbs	117,320

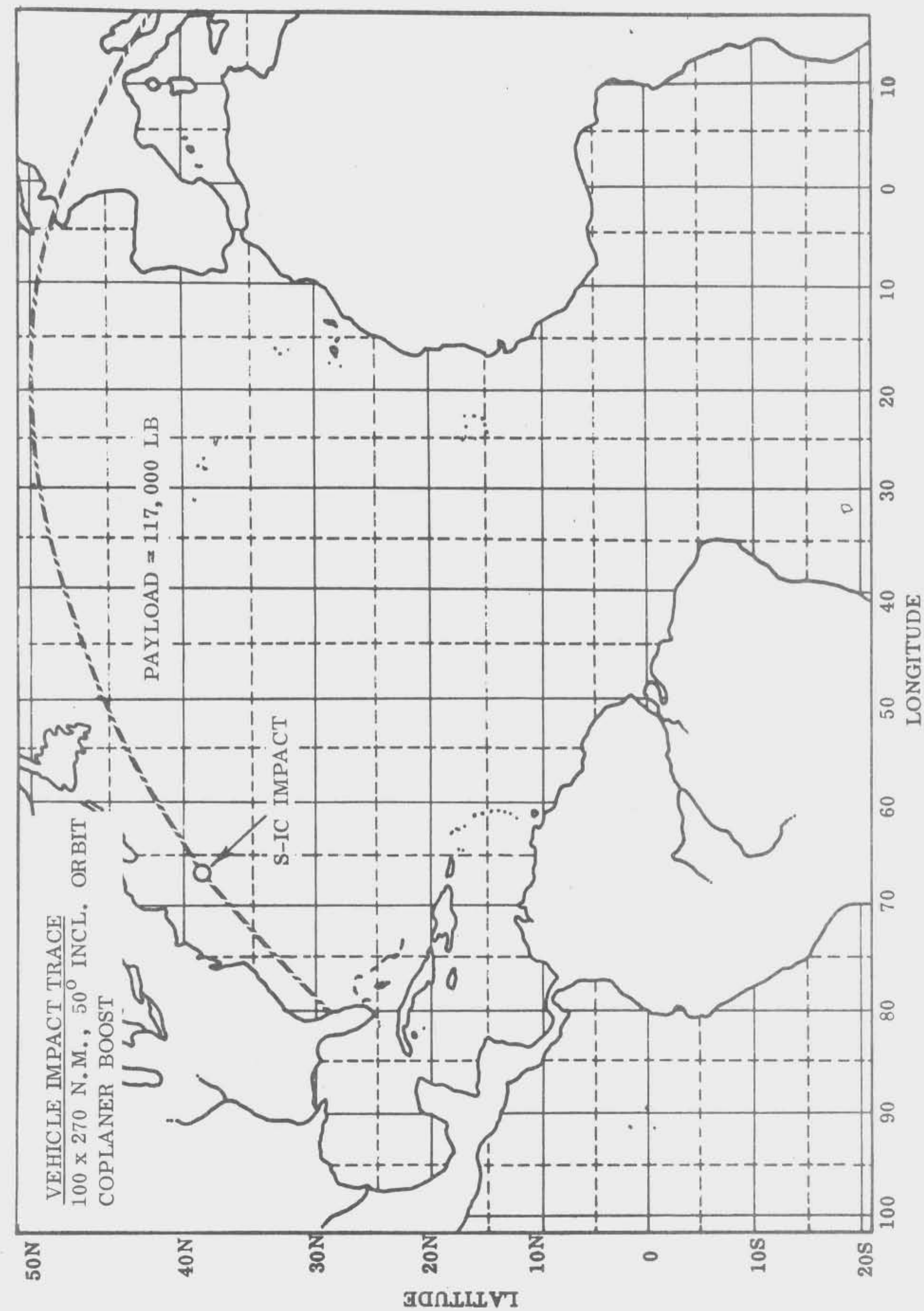


FIGURE 4.3.2-2 INT-20/BIG G IMPACT TRACE - NORTHERLY LAUNCH

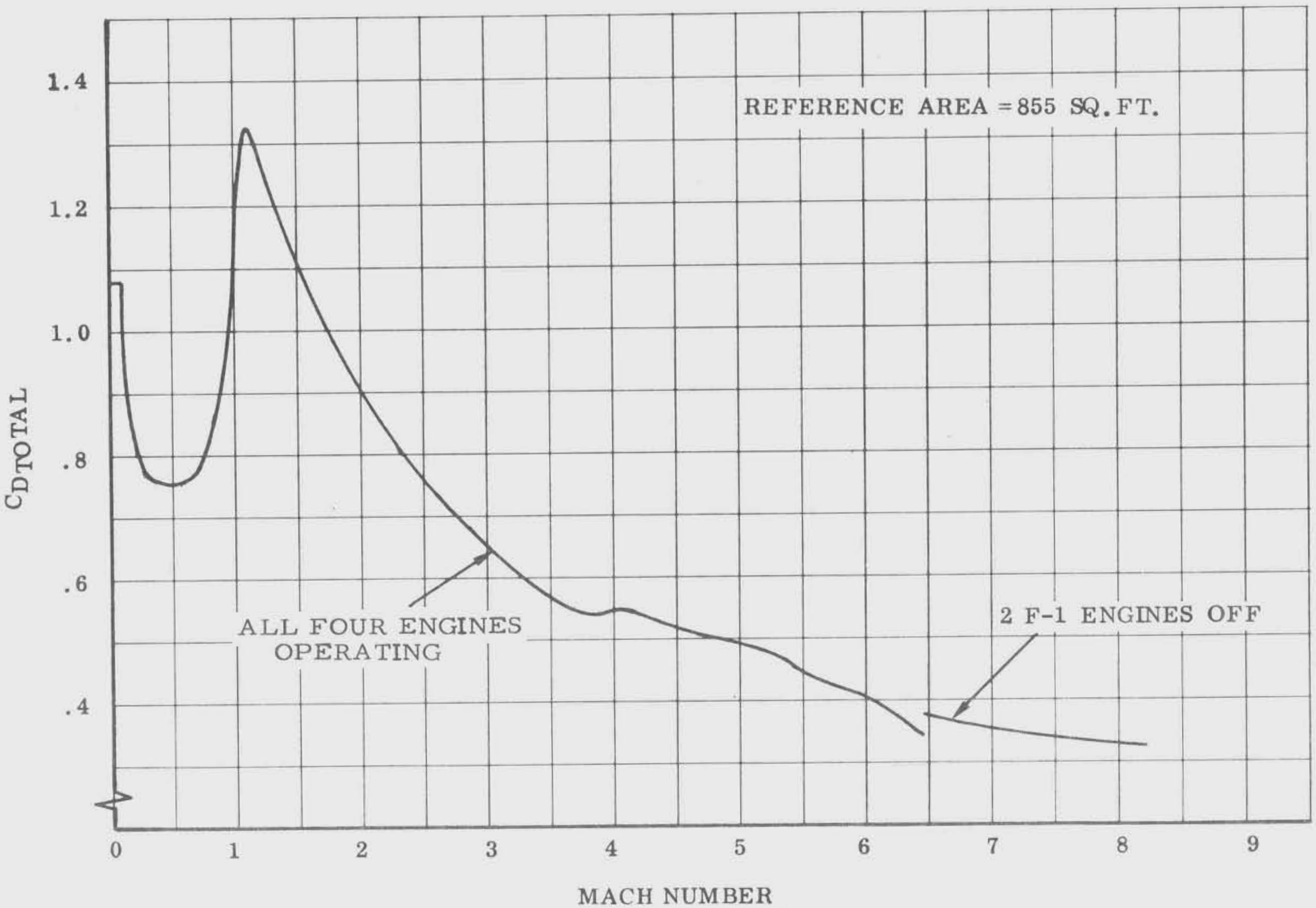


FIGURE 4.3.2-3 INT-20/BIG TOTAL DRAG COEFFICIENT

TABLE 4.3.2-II

30506 - INT-20/BIG G
100 x 270 N.M., 50° ORBIT
OPTIMUM BOOST TURN TO AVOID LAND
MASS IMPACT
(SOUTHERLY LAUNCH)

FIRST STAGE OPERATION		
Lift-Off Weight	lbs	4,870,400
T/W Ratio		1.25
Sea Level Thrust	lbs	6,088,000
Sea Level I_{SP}	sec	263.58
Liquid Propellant Consumed	lbs	4,122,325
Stage Weight @ Staging	lbs	332,635
LES & Ballast	lbs	89,094
Max. Dynamic Pressure	lbs/ft^2	776
Lift-Off Azimuth Angle	Deg	95.11
SECOND STAGE OPERATION		
Thrust (VAC)	lbs	205,000
I_{SP} (Nominal)	sec	426
Weight @ Ignition	lbs	326,346
Propellant Capacity	lbs	230,000
Propellant Consumed	lbs	228,174
Stage Separation Weight	lbs	27,504
Gross Payload	lbs	70,668
Astrionics Equipment	lbs	4,183
Flight Perf. Reserves	lbs	1,826
Net Payload	lbs	64,659

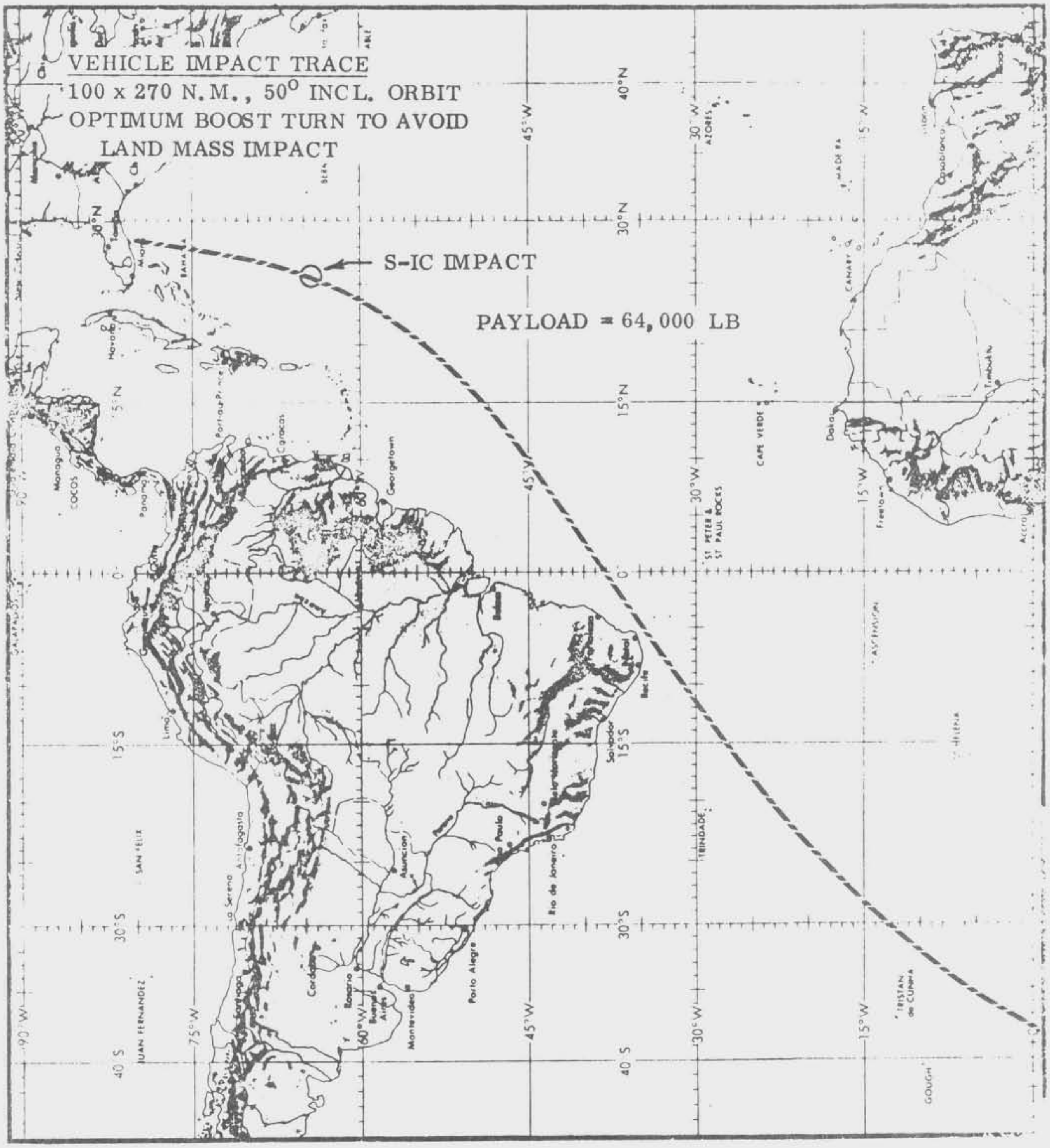


FIGURE 4.3.2-4 INT-20/BIG G IMPACT TRACE - SOUTHERLY LAUNCH

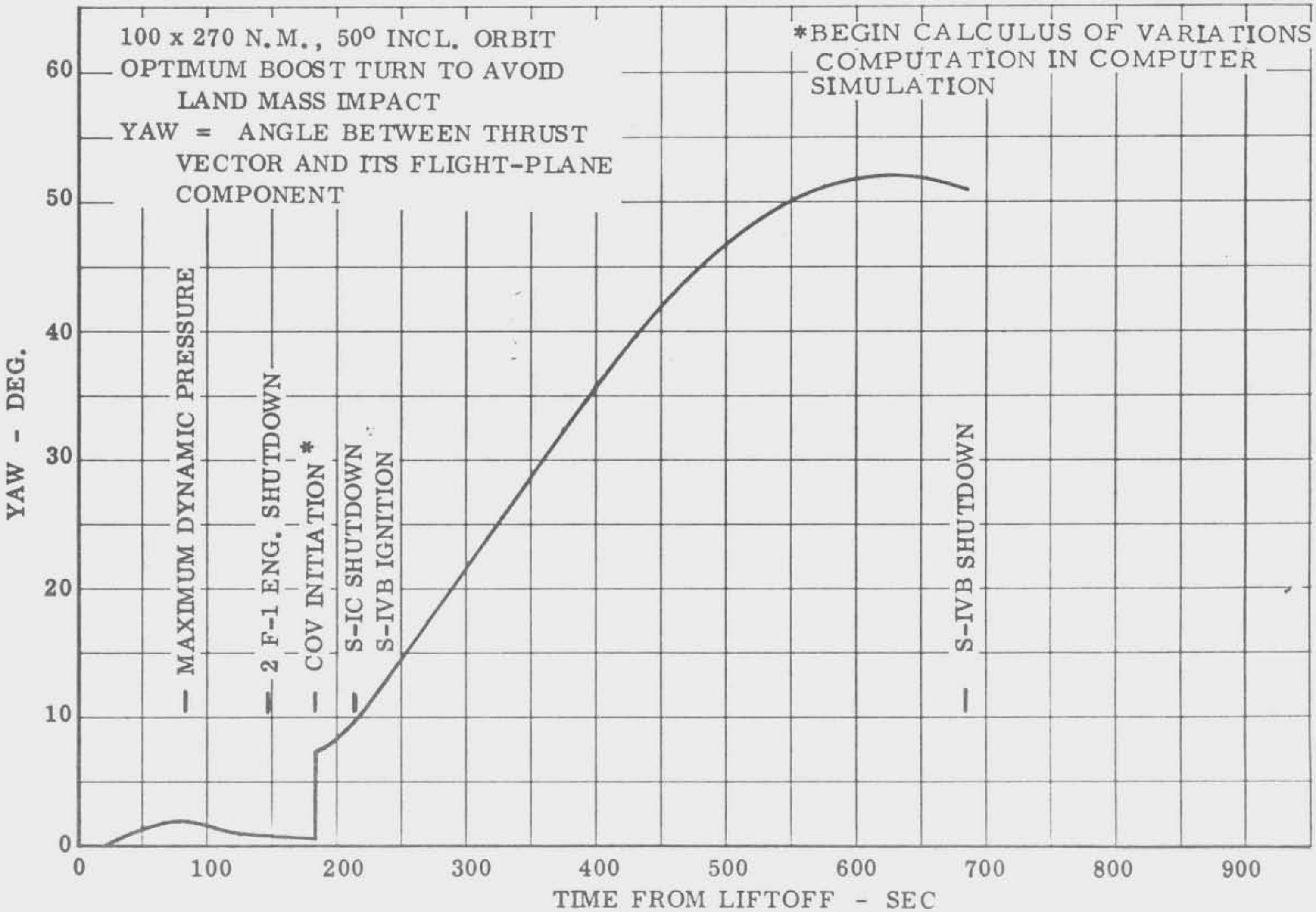


FIGURE 4.3.2-5 INT-20/BIG G OPTIMUM BOOST TURN - YAW

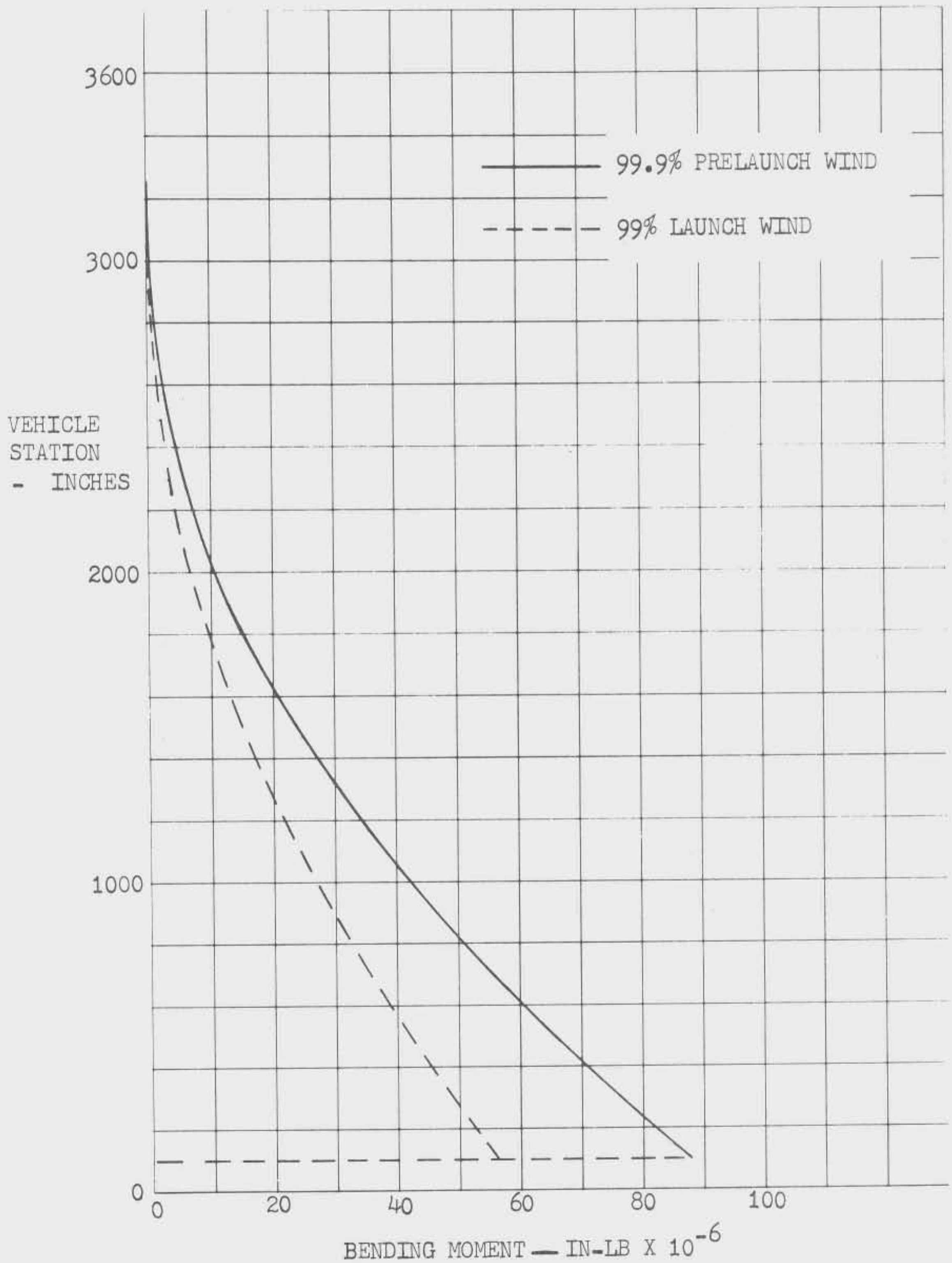


FIGURE 4.3.2-6 INT-20/BIG G VEHICLE GROUND WIND BENDING MOMENT DISTRIBUTION

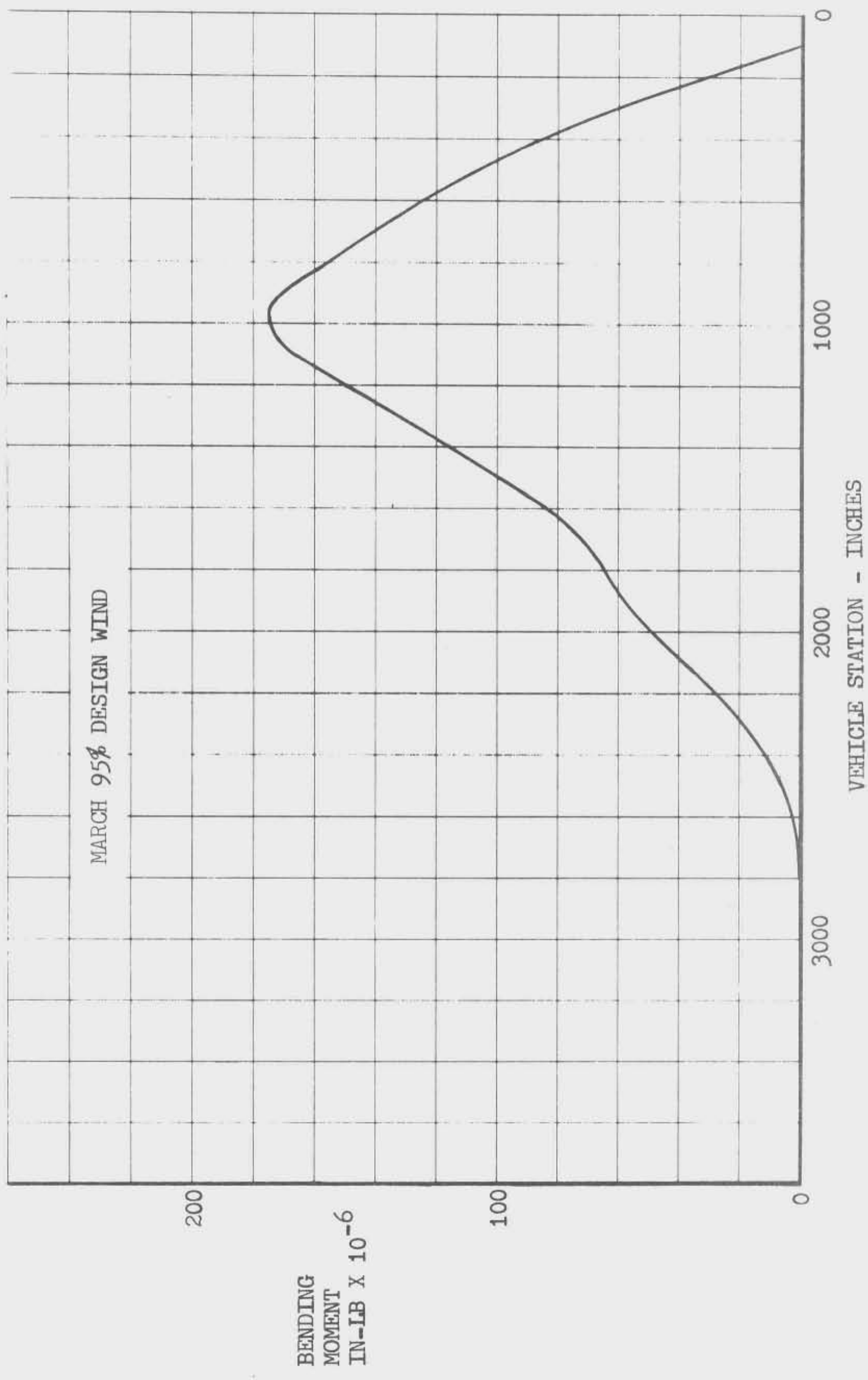


FIGURE 4.3.2-7 INT-20/BIG G INFLIGHT BENDING MOMENT @ MAX (Q α)

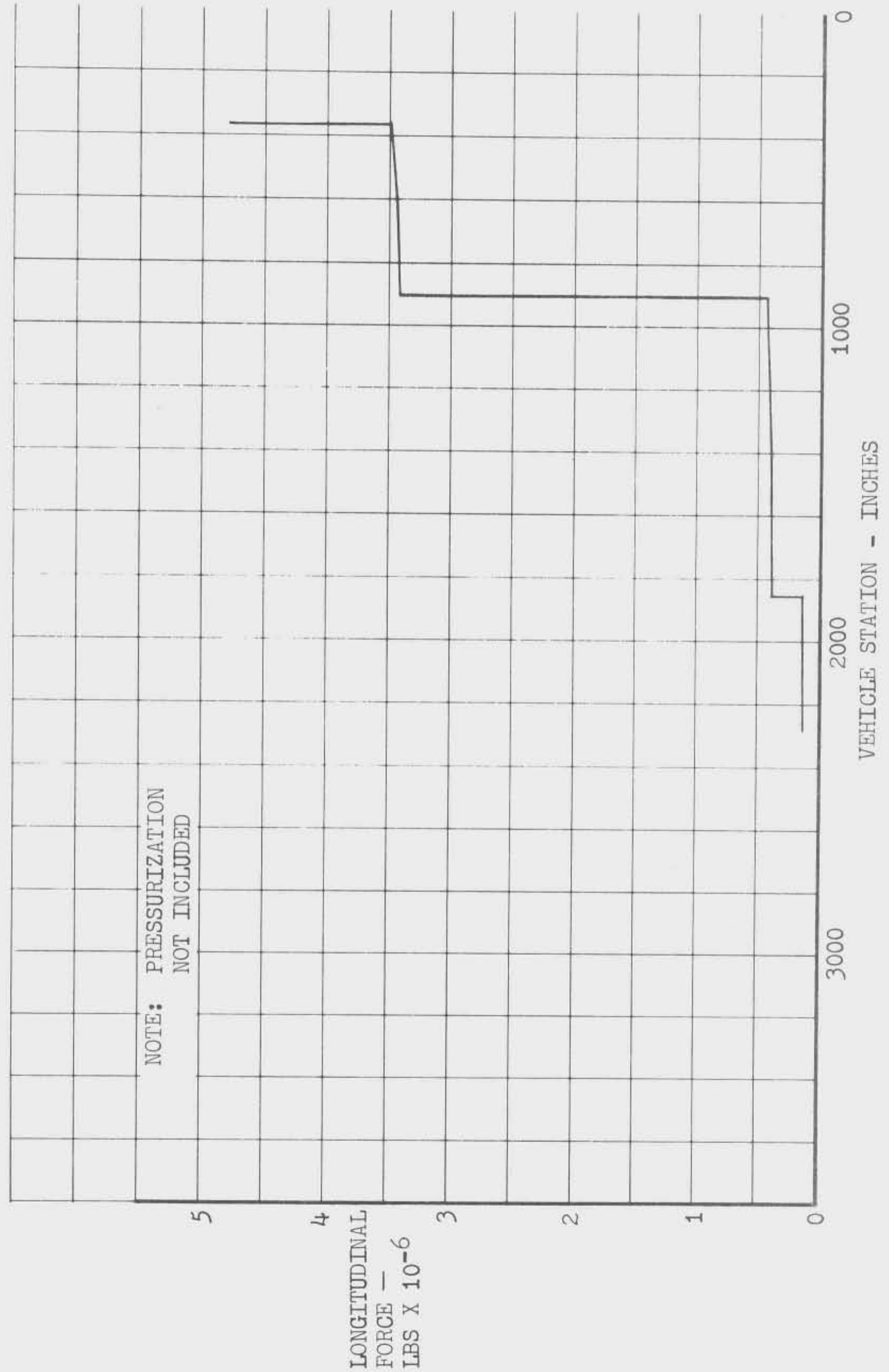


FIGURE 4.3.2-8 INT-20/BIG G VEHICLE LONGITUDINAL FORCE DISTRIBUTION FOR ON-PAD, FUELED, UNPRESSURIZED CONDITION

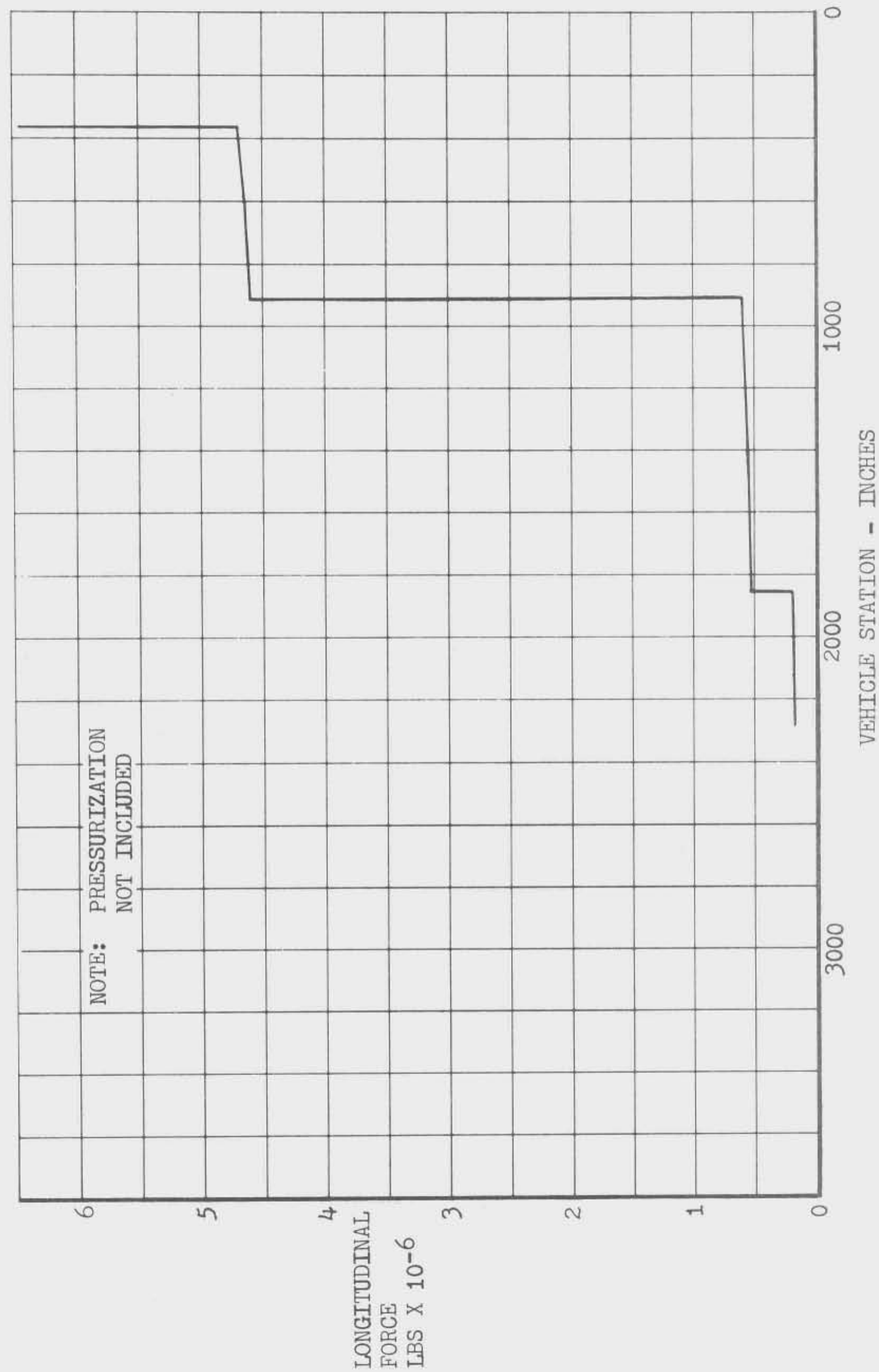


FIGURE 4.3.2-9 INT-20/BIG G VEHICLE LONGITUDINAL FORCE DISTRIBUTION FOR REBOUND (EMERGENCY SHUTDOWN)

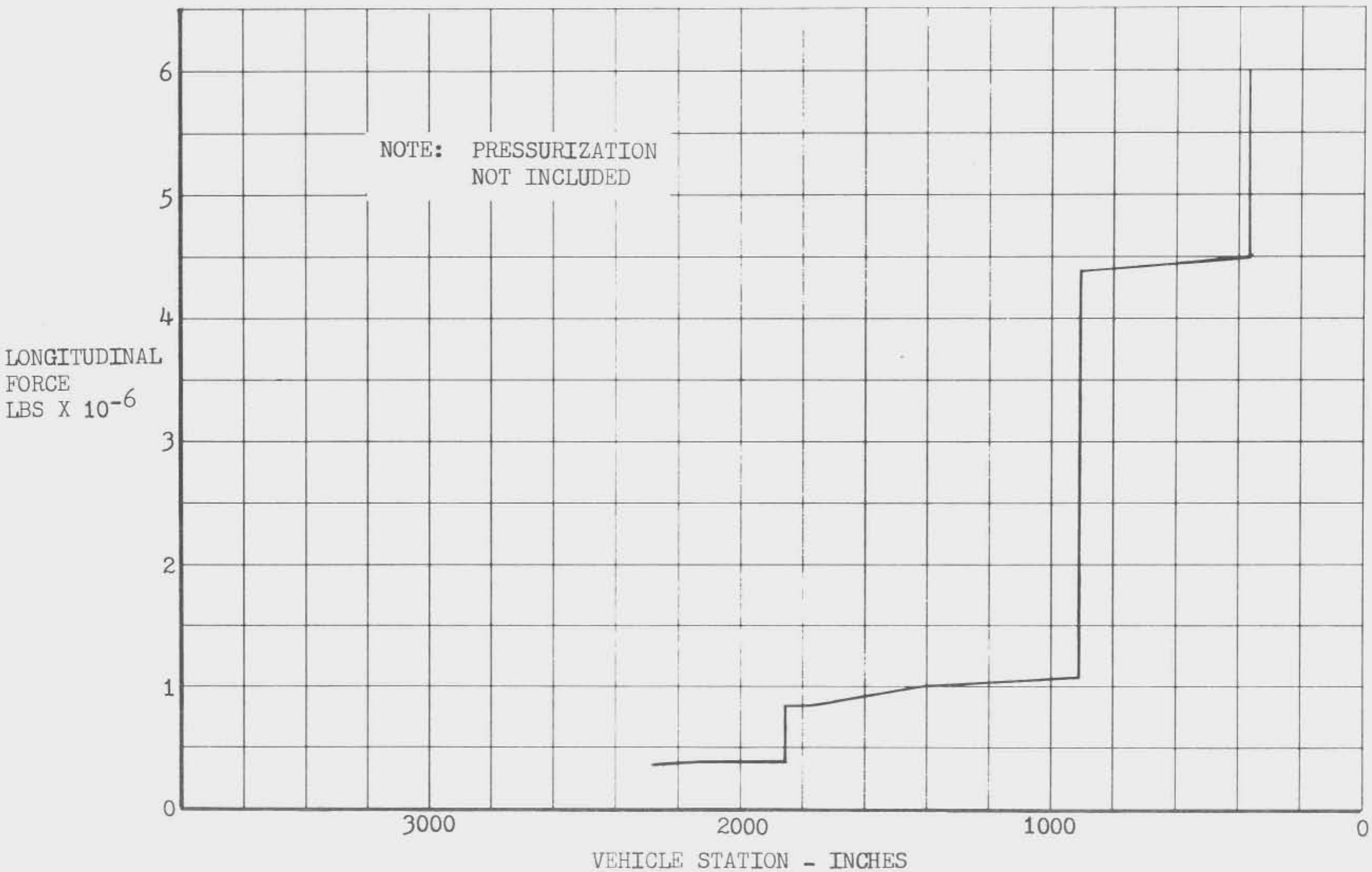


FIGURE 4.3.2-10 INT-20/BIG G VEHICLE LONGITUDINAL FORCE DISTRIBUTION AT MAX ($Q\alpha$)

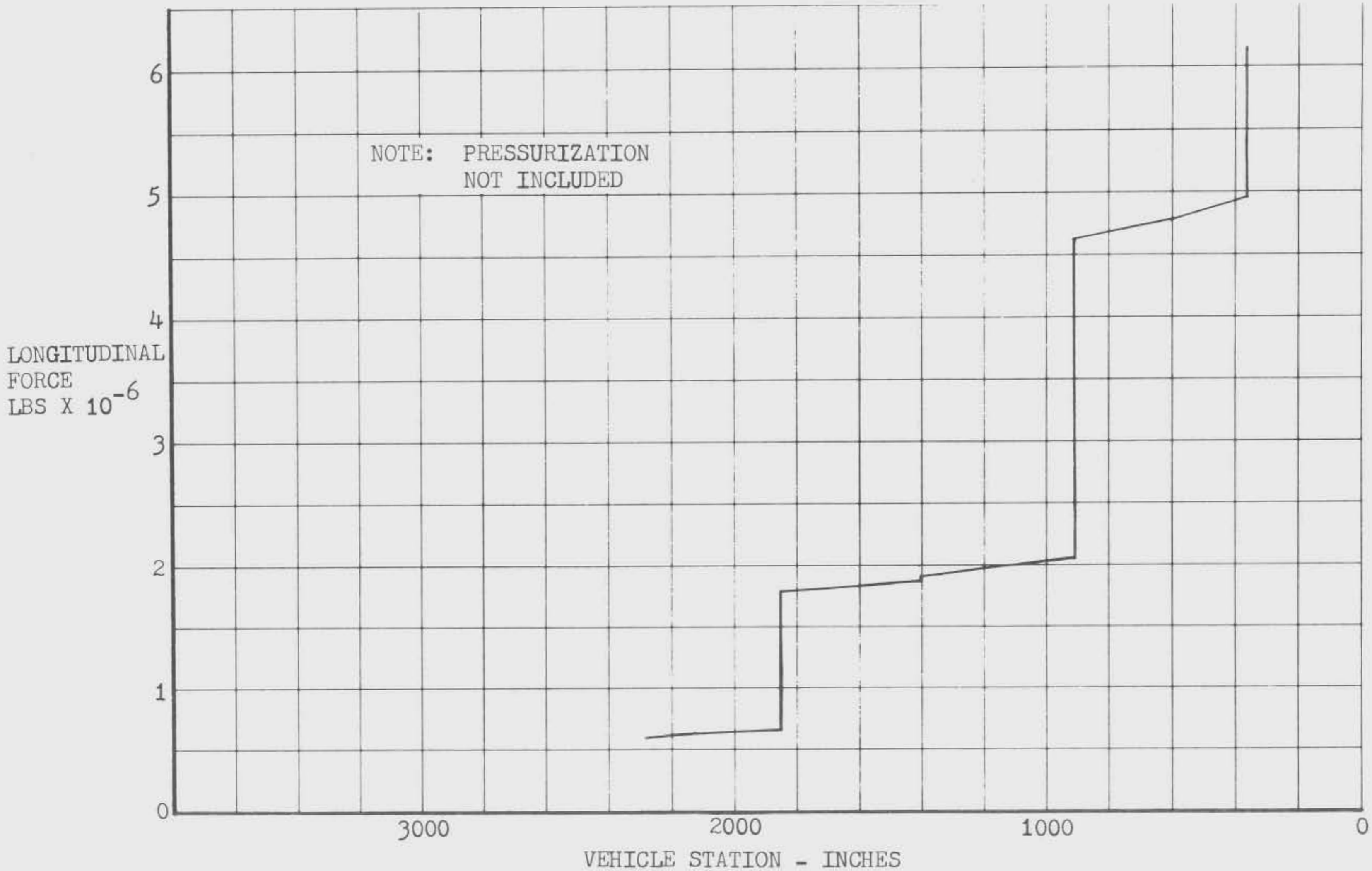


FIGURE 4.3.2-11 INT-20/BIG G VEHICLE LONGITUDINAL FORCE DISTRIBUTION AT PEAK ACCELERATION (t = 146 SEC.)

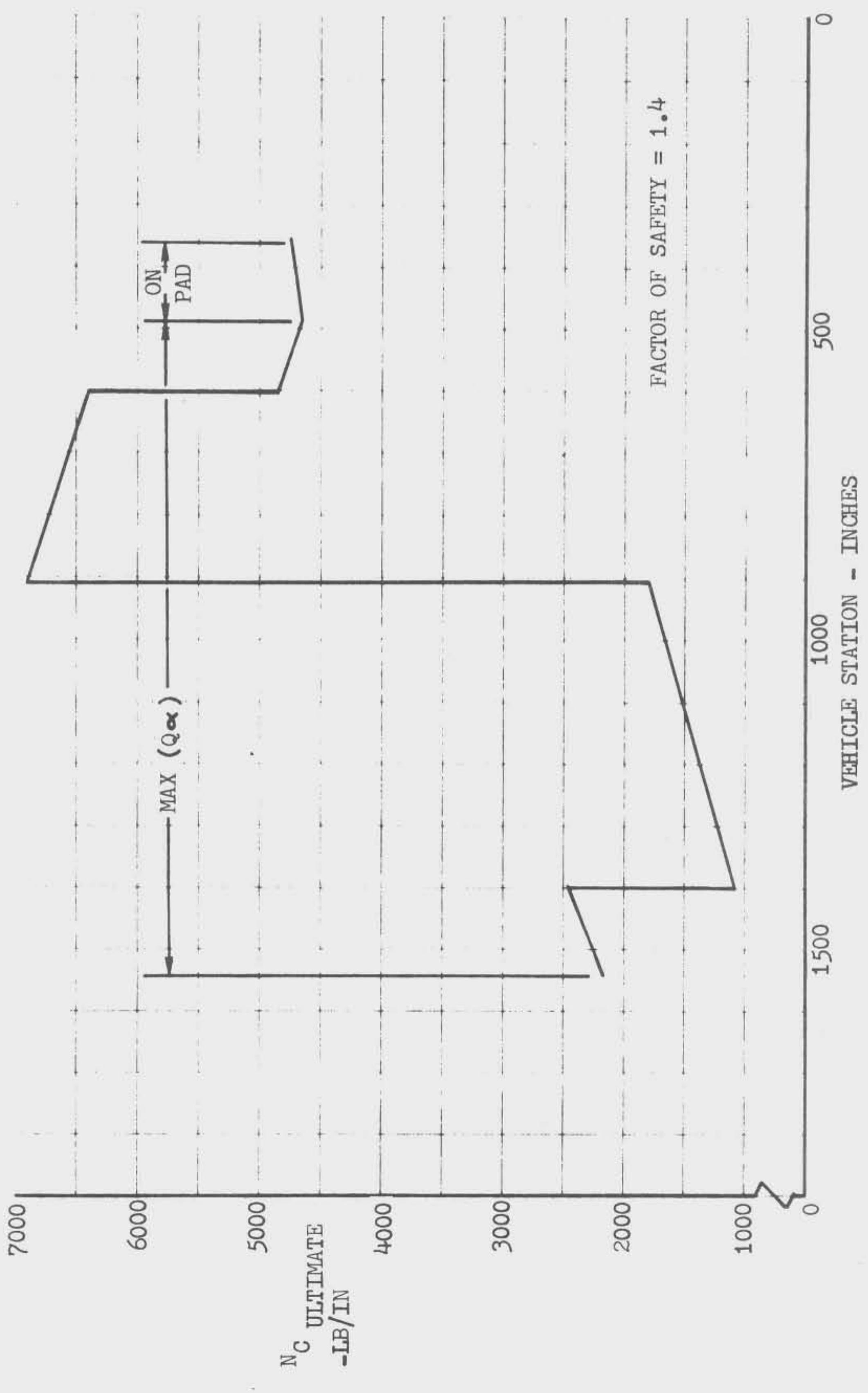


FIGURE 4.3.2-12 INT-20/BIG G S-IC ULTIMATE COMPRESSIVE COMBINED LOAD

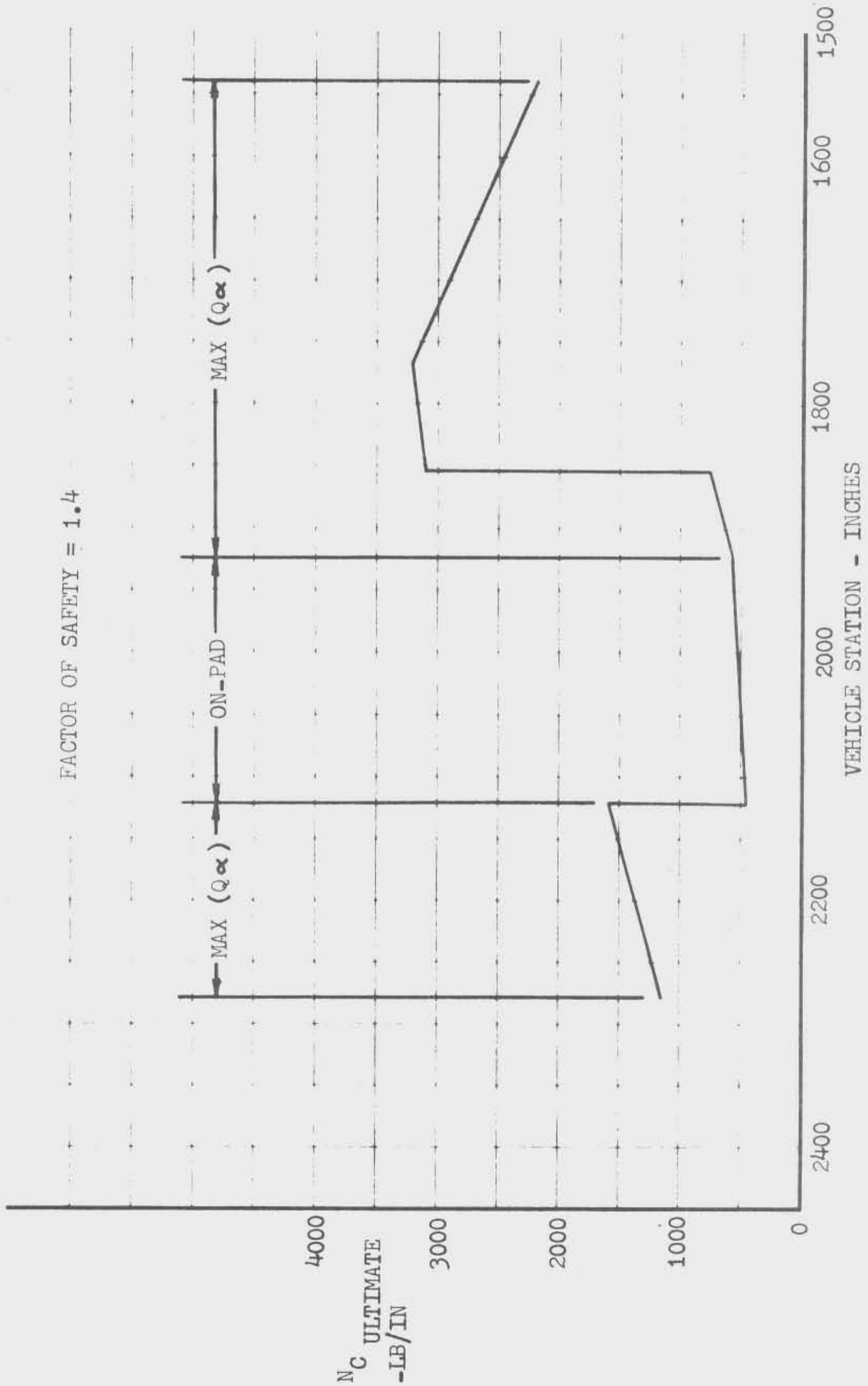


FIGURE 4.3.2-13 INT-20/BIG G S-IVB AND I.U. ULTIMATE COMPRESSIVE COMBINED LOAD

TABLE 4.3.2-IV INT-20/BIG G N_C-ULTIMATE FOR ON-PAD, FUELED, UNPRESSURIZED CONDITION

STATION (IN)	$M \times 10^{-6}$ (IN-LB)	$M/\pi R^2$ (LB/IN)	$P \times 10^{-6}$ (LB)	$P/2\pi R$ (LB/IN)	N_C LIMIT (LB/IN	1.4 N_C LIMIT (LB/IN)	P_{ULLAGE} (PSIG)	$\frac{R}{2} P_U$ (LB/IN)	N_C ULT (LB/IN)
365A	73.4	595.96	4.7840	3845	4441.4	6218.0	0	0	6218.0
365F	73.4	595.96	3.4870	2803	3398.9	4758.4	0	0	4758.4
602A	61.3	497.71	3.4448	2769	3266.7	4573.4	0	0	4573.4
602F	61.3	497.71	3.4405	2766	3263.2	4568.5	0	0	4568.5
912A	46.5	377.55	3.4164	2746	3123.7	4373.2	0	0	4373.2
912F	46.5	377.55	.4394	353	730.7	1023.0	0	0	1023.0
1401A	27.3	221.66	.4101	330	551.3	771.8	0	0	771.8
1401F	27.3	221.66	.4053	326	547.4	766.4	0	0	766.4
1541	22.9	185.9	.3965	319	504.6	706.5	0	0	706.5
1768	16.2	305.1	.3853	472	776.8	1087.6	0	0	1087.6
1854A	14.0	263.7	.3839	470	733.7	1027.2	0	0	1027.2
1854F	14.0	263.7	.1420	174	437.5	612.5	0	0	612.5
2123A	8.7	163.9	.1351	165	429.3	461.0	0	0	461.0
2123F	8.7	163.9	.1344	164	328.4	459.8	0	0	459.8
2245	6.8	128.1	.1316	161	289.2	404.9	0	0	404.9
2281	6.0	113.0	.1274	156	269.0	376.6	0	0	376.6

TABLE 4.3.2-V INT-20/BIG G N_C ULTIMATE FOR EMERGENCY SHUTDOWN CONDITION

STATION (IN)	M x 10^{-6} (IN-LB)	$M/\pi R^2$ (LB/IN)	P x 10^{-6} (LB)	$P/2\pi R$ (LB/IN)	N_C LIMIT (LB/IN	1.4 N_C LIMIT (LB/IN)	P_{ULLAGE} (PSIG)	$\frac{R}{2} P_U$ (LB/IN)	N_C ULT (LB/IN)
365A	47.1	382.4	6.4520	5186	5568.6	7796.1	0	0	7796.1
365F	47.1	382.4	4.6995	3778	4160.0	5823.9	12.8	1267	4556.7
602A	39.0	316.7	4.6414	3731	4047.5	5666.5	12.8	1267	4399.3
602F	39.0	316.7	4.6356	3726	4042.8	5659.9	0	0	5659.9
912A	29.5	239.5	4.6025	3700	3939.1	5514.7	0	0	5514.7
912F	29.5	239.5	.6041	486	725.1	1015.5	9.3	921	94.4
1401A	17.2	139.7	.5639	453	592.9	830.1	9.3	921	-90.6
1401F	17.2	139.7	.5573	448	587.6	822.7	0	0	822.7
1541	14.3	116.1	.5452	438	554.3	776.1	0	0	776.1
1768	10.3	194.0	.5298	649	842.6	1179.7	0	0	1179.7
1854A	9.1	171.4	.5278	646	817.6	1144.6	0	0	1144.6
1854F	9.1	171.4	.1952	239	410.4	574.5	13.3	865	-290.0
2123A	5.7	107.4	.1858	228	334.8	468.8	13.3	865	-395.7
2123F	5.7	107.4	.1848	226	333.6	467.0	0	0	467.0
2245	4.4	82.9	.1809	222	304.3	426.1	0	0	426.1
2281	4.0	75.3	.1751	214	289.7	405.6	0	0	405.6

TABLE 4.3.2-VI INT-20/BIG G N_C ULTIMATE FOR MAX $(Q\alpha)$ CONDITION

STATION (IN)	M x 10^{-6} (IN-LB)	$M/\pi R^2$ (LB/IN)	P x 10^{-6} (LB)	$P/2\pi R$ (LB/IN)	N_C LIMIT (LB/IN	1.4 N_C LIMIT (LB/IN)	P_{ULLAGE} (PSIG)	$\frac{R}{2} P_U$ (LB/IN)	N_C ULT (LB/IN)
365A	76.3	620	5.9823	4809	5429	7601	0	0	7601
365F	76.3	620	4.5269	3639	4259	5963	15.6	1544	4419
602A	123	999	4.4450	3573	4572	6401	15.6	1544	4857
602F	123	999	4.4371	3567	4566	6392	0	0	6392
912A	171.9	1396	4.3879	3527	4923	6892	0	0	6892
912F	171.9	1396	1.1005	885	2281	3193	14.1	1396	1797
1401A	115.4	937	1.0390	835	1772	2481	14.1	1396	1085
1401F	115.4	937	1.0300	828	1765	2471	0	0	2471
1541	91.6	744	1.0116	813	1557	2180	0	0	2180
1768	66.4	1251	.8486	1039	2290	3206	0	0	3206
1854A	62.5	1177	.8459	1036	2213	3098	0	0	3098
1854F	62.5	1177	.3962	485	1662	2327	24.1	1567	760
2123A	35.5	669	.3796	465	1134	1588	24.1	1567	21
2123F	35.5	669	.3783	463	1132	1585	0	0	1585
2245	23.7	446	.3713	455	901	1261	0	0	1261
2281	20.4	384	.3629	444	828	1159	0	0	1159

TABLE 4.3.2-VII INT-20/BIG G N_C ULTIMATE FOR MAXIMUM ACCELERATION CONDITION

STATION (IN)	M x 10^{-6} (IN-LB)	$M/\pi R^2$ (LB/IN)	P x 10^{-6} (LB)	$P/2\pi R$ (LB/IN)	N_C LIMIT (LB/IN	1.4 N_C LIMIT (LB/IN)	P_{ULLAGE} (PSIG)	$\frac{R}{2} P_U$ (LB/IN)	N_C ULT (LB/IN)
365A	0	0	6.1679	4958	Same as	6941	0	0	6941
365F	0	0	4.9627	3989	$(P/2\pi R)$	5585	19.5	1931	3654
602A	0	0	4.7656	3831		5363	19.5	1931	3432
602F	0	0	4.7458	3815		5341	0	0	5341
912A	0	0	4.6331	3724		5214	0	0	5214
912F	0	0	2.0527	1650		2310	18.0	1782	528
1401A	0	0	1.9162	1540		2156	18.0	1782	374
1401F	0	0	1.8938	1522		2131	0	0	2131
1541	0	0	1.8524	1489		2085	0	0	2085
1768	0	0	1.8002	2204		3086	0	0	3086
1854A	0	0	1.7935	2196		3074	0	0	3074
1854F	0	0	.6632	812		1137	28.0	1820	-683
2123A	0	0	.6314	773		1082	28.0	1820	-738
2123F	0	0	.6280	769		1077	0	0	1077
2245	0	0	.6146	752		1053	0	0	1053
2281	0	0	.5951	729		1021	0	0	1021

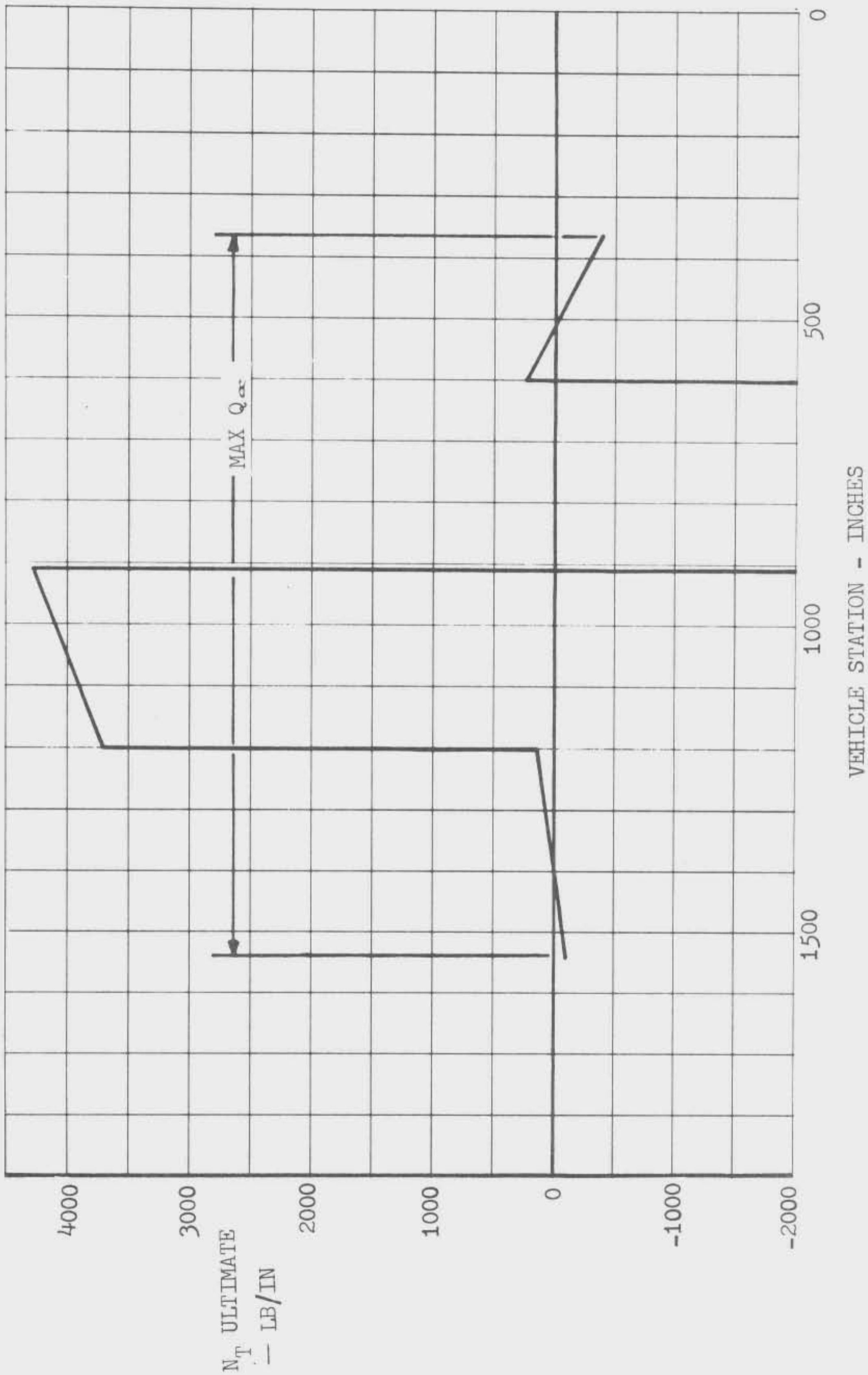


FIGURE 4.3.2-14 INT-20/BIG G S-IC COMBINED TENSION LOADS

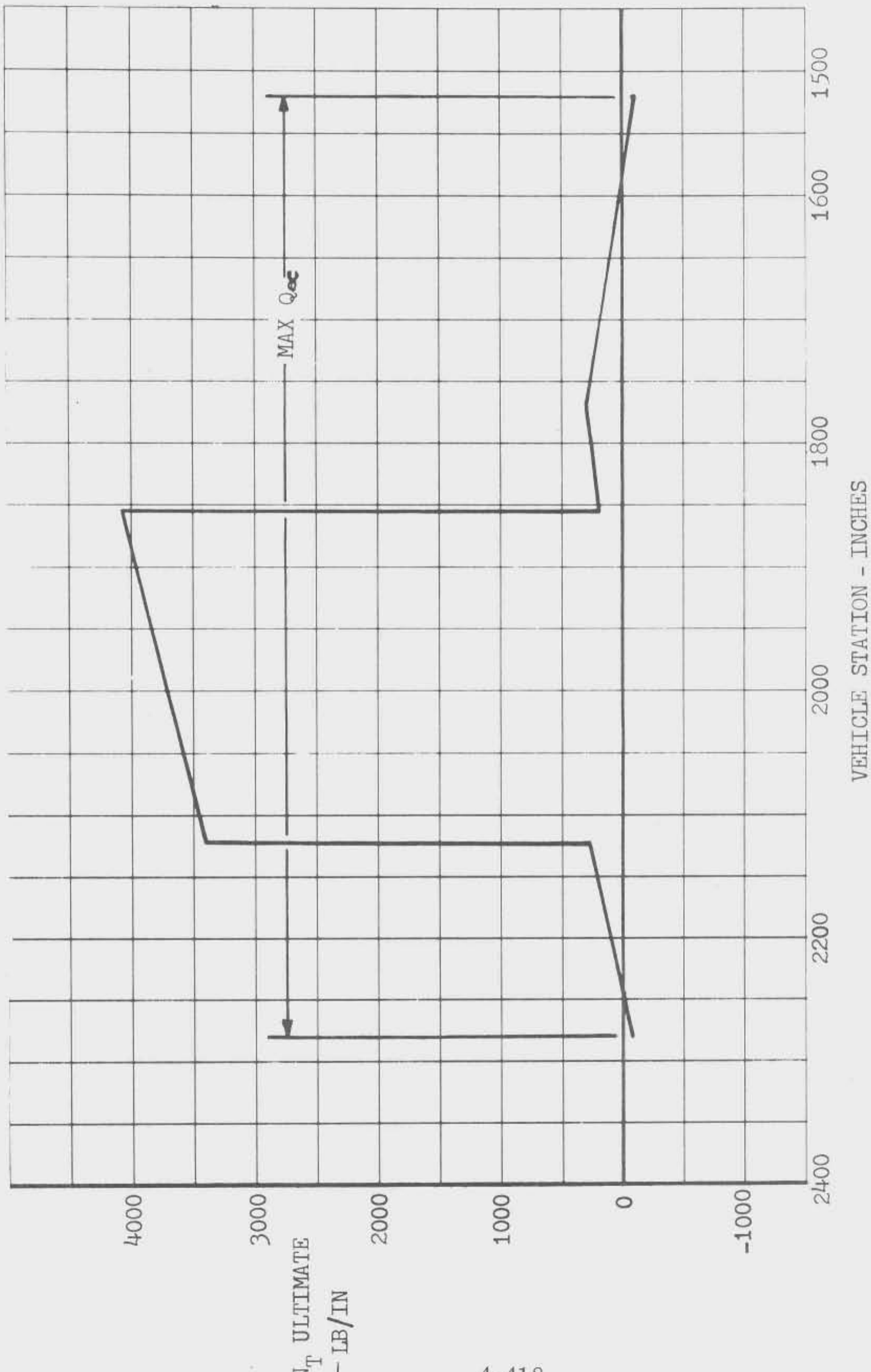


FIGURE 4.3.2-15 INT-20/BIG G S-IVB & I.U. COMBINED TENSION LOAD

TABLE 4.3.2-VIII INT-20/BIG G N_T ULTIMATE FOR ON-PAD, FUELED, UNPRESSURIZED CONDITION

STATION (IN)	M x 10^{-6} (IN-LB)	$M/\pi R^2$ (LB/IN)	P x 10^{-6} (LB)	$P/2\pi R$ (LB/IN)	$(M/\pi R^2 - P/2\pi R)$ (LB/IN)	P_U MAX (LB/IN)	P_U R/2 (PSIG)	N_T LIMIT (LB/IN)	N_T ULT. (LB/IN)
365A	73.4	596.0	4.7840	3845	-3249.0	0	0	-3249	-4549
365F	73.4	596.0	3.4870	2803	-2207.0	0	0	-2207	-3090
602A	61.3	497.7	3.4448	2769	-2271.3	0	0	-2271.3	-3180
602F	61.3	497.7	3.4405	2766	-2268.3	0	0	-2268.3	-3176
912A	46.5	377.6	3.4164	2746	-2368.4	0	0	-2368.3	-3316
912F	46.5	377.6	.4394	353	24.6	0	0	24.6	34
1401A	27.3	221.7	.4101	330	-108.3	0	0	-108.3	-152
1401F	27.3	221.7	.4053	326	-104.3	0	0	-104.3	-146
1541	22.9	185.9	.3965	319	-133.1	0	0	-133.1	-186
1768	16.2	305.1	.3853	472	-166.9	0	0	-166.9	-234
1854A	14.0	263.7	.3839	470	-206.3	0	0	-206.3	-289
1854F	14.0	263.7	.1420	174	89.7	0	0	89.7	126
2123A	8.7	163.9	.1351	165	-1.1	0	0	-1.1	-2
2123F	8.7	163.9	.1344	164	-0.1	0	0	-0.1	-.14
2245	6.8	128.1	.1316	161	-32.9	0	0	-32.9	-46
2281	6.0	113.0	.1274	156	-43.0	0	0	-43.0	-60

TABLE 4.3.2-IX INT-20/BIG G N_T ULTIMATE FOR EMERGENCY SHUTDOWN CONDITION

STATION (IN)	M x 10^{-6} (IN-LB)	$M/\pi R^2$ (LB/IN)	P x 10^{-6} (LB)	$P/2\pi R$ (LB/IN)	$(M/\pi R^2 - P/2\pi R)$ (LB/IN)	P_U MAX (LB/IN)	P_U R/2 (PSIG)	N_T LIMIT (LB/IN)	N_T ULT. (LB/IN)
365A	47.1	382.4	6.4520	5186	-4803.6	0	0	-4803.6	-6725
365F	47.1	382.4	4.6995	3778	-3395.6	16.8	1663.2	-1743.4	-2425
602A	39.0	316.7	4.6414	3731	-3414.3	16.8	1663.2	-1751.1	-2452
602F	39.0	316.7	4.6356	3726	-3409.3	0	0	-3409.3	-4773
912A	29.5	239.5	4.6025	3700	-3460.5	0	0	-3460.5	-4845
912F	29.5	239.5	.6041	486	-246.5	16.8	1663.2	1416.7	1983
1401A	17.2	139.7	.5639	453	-313.3	16.8	1663.2	1349.9	1890
1401F	17.2	139.7	.5573	448	-308.3	0	0	-308.3	-432
1541	14.3	116.1	.5452	438	-321.9	0	0	-321.9	-451
1768	10.3	194.0	.5298	649	-455.0	0	0	-455.0	-637
1854A	9.1	171.4	.5278	646	-474.6	0	0	-474.6	-664
1854F	9.1	171.4	.1952	239	- 67.6	23.3	1514.5	1446.9	2026
2123A	5.7	107.4	.1858	228	-120.6	23.3	1514.5	1393.9	1951
2123F	5.7	107.4	.1848	226	-118.6	0	0	-118.6	-166
2245	4.4	82.9	.1809	222	-139.1	0	0	-139.1	-195
2281	4.0	75.3	.1751	214	-138.7	0	0	-138.7	-194

TABLE 4.3.2-X INT-20/BIG G N_T ULTIMATE FOR MAX (Q_α) CONDITION

STATION (IN)	M x 10^{-6} (IN-LB)	$M/\pi R^2$ (LB/IN)	P x 10^{-6} (LB)	$P/2\pi R$ (LB/IN)	$(M/\pi R^2 - P/2\pi R)$ (LB/IN)	P_U MAX (LB/IN)	P_U R/2 (PSIG)	N_T LIMIT (LB/IN)	N_T ULT. (LB/IN)
365A	76.3	620	5.9823	4809	-4189	0	0	-4189.0	-5865
365F	76.3	620	4.5269	3639	-3019	27.65	2737.4	-281.6	-394
602A	123	999	4.4450	3573	-2574	27.65	2737.4	163.4	229
602F	123	999	4.4371	3567	-2568	0	0	-2568.0	-3595
912A	171.9	1396	4.3879	3527	-2131	0	0	-2131.0	-2983
912F	171.9	1396	1.2005	885	511	25.5	2549.7	3060.7	4285
1401A	115.4	937	1.0390	835	102	25.5	2549.7	2651.7	3712
1401F	115.4	937	1.0300	828	109	0	0	109.0	153
1541	91.6	744	1.0116	813	-69	0	0	-69.0	-97
1768	66.4	1251	.8486	1039	212	0	0	212.0	297
1854A	62.5	1177	.8459	1036	141	0	0	141.0	197
1854F	62.5	1177	.3962	485	692	34.15	2219.8	2911.8	4077
2123A	35.5	669	.3796	465	204	34.15	2219.8	2423.8	3393
2123F	35.5	669	.3783	463	206	0	0	206.0	288
2245	23.7	346	.3713	455	-11	0	0	-11.0	-15
2281	20.4	384	.3629	444	-60	0	0	-60.0	-84

TABLE 4.3.2-XI INT-20/BIG G N_T ULTIMATE FOR MAXIMUM ACCELERATION CONDITION

STATION (IN)	M x 10^{-6} (IN-LB)	$M/\pi R^2$ (LB/IN)	P x 10^{-6} (LB)	$P/2\pi R$ (LB/IN)	$(M/\pi R^2 - P/2\pi R)$ (LB/IN)	P_U MAX (LB/IN)	P_U R/2 (PSIG)	N_T LIMIT (LB/IN)	N_T ULT. (LB/IN)
365A	0	0	6.1679	4958	-4958	0	0	-4958	-6941
365F	0	0	4.9627	3989	-3989	31.5	3119	- 870	-1218
602A	0	0	4.7656	3831	-3831	31.5	3119	- 712	- 997
602F	0	0	4.7458	3815	-3815	0	0	-3815	-5341
912A	0	0	4.6331	3724	-3724	0	0	-3724	-5214
912F	0	0	2.0527	1650	-1650	25.5	2525	875	1225
1401A	0	0	1.9162	1540	-1540	25.5	2525	985	1379
1401F	0	0	1.8938	1522	-1522	0	0	-1522	-2131
1541	0	0	1.8524	1489	-1489	0	0	-1489	-2085
1768	0	0	1.8002	2204	-2204	0	0	-2204	-3086
1854A	0	0	1.7935	2196	-2196	0	0	-2196	-3074
1854F	0	0	.6632	812	-812	38.0	2470	1658	2321
2123A	0	0	.6314	773	-773	38.0	2470	1697	2376
2123F	0	0	.6280	769	-769	0	0	-769	-1077
2245	0	0	.6146	752	-752	0	0	-752	-1053
2281	0	0	.5951	729	-729	0	0	-729	-1021

4.3.3 Removal of S-IVB Restart Capability

4.3.3.1 Alternate Stage Configuration

The baseline INT-20/S-IVB stage, as described in Section 4.2.4.1, is a Saturn V/S-IVB stage with a number of changes and/or deletions associated with removing the stage restart capability. These changes would be somewhat minor, however, designed more to render affected subsystems non-functional rather than remove them entirely. Thus, flexibility to return to a re-startable configuration was maintained.

As an INT-20 alternate configuration, a Saturn V/S-IVB stage with more substantial changes and/or deletions was investigated. In this case, entire subsystems were removed, so the aforementioned flexibility (to add restart) was not maintained. Some of the baseline stage deletions apply also to the alternate, i.e., the ambient repressurization system, the APS ullaging engines, the retrorocket plume impingement curtain and the instrumentation associated with those systems. In addition, the following Saturn V stage systems or installations will be deleted.

a. Cryogenic repressurization system. Not only will the O_2H_2 burner be deleted, but also the associated plumbing, burner ignition exciters, burner supports and other associated hardware. In conjunction with the burner system removal, three cold helium spheres, strap assemblies and attaching parts will be removed. A new, shortened manifold would be employed for sphere No. 6.

b. Continuous Vent System. The entire continuous vent system - bellows, modules, ducts, flanges, nozzles, etc. - will be removed and the forward skirt openings covered.

c. Fuel Tank Baffles. The baffle and deflector assembly located in the forward end of the fuel tank will be removed. The propellant tank wall studs used for attachment are left in place.

d. Electrical/Instrumentation. Instrumentation associated with the deleted systems will be deleted, and wire harnesses will be reworked and wires and connectors removed rather than coiled and stowed with the stage.

4.3.3.2 Alternate Interstage Configuration

The interstage for the alternate configuration will be the same as that for the baseline configuration. All retrorocket provisions will be deleted and an adapter ring will be employed for S-IC/S-IVB stage mating.

4.3.3.3 Propulsion System, Alternate Configuration

The propulsion system changes/deletions outlined for the baseline INT-20/S-IVB stage (Section 4.2.4.3) also apply to the alternate configuration. In addition, three cold helium spheres associated with the cryogenic repressurization system are deleted, and the propulsive vent (or continuous vent) system is removed entirely. These changes are reflected in the propulsion system schematic for the alternate stage, Figure 4.3.3-1.

The removal of three cold helium spheres from the baseline configuration results in a storage capability equal to that of the normal Saturn IB/S-IVB stage. This configuration will afford sufficient mass for failure contingencies and the Saturn IB-type mission duration requirements.

The physical removal of the LH_2 continuous vent system (as opposed to isolating it, as was the case for the baseline configuration) does not impair stage performance. Neither do these deletions have any effect on the remaining propulsion systems.

4.3.3.4 Electrical System, Alternate Configuration

Whereas the general approach for the baseline configuration was to coil and stow unused connectors, that for the alternate configuration is to rework wire harnesses and delete wiring for the deleted subsystems. Instrumentation deletions are essentially the same for both configurations, so will not be discussed further in this section.

The following list itemizes the electrical system changes that would take place due to the various systems' deletions on the alternate INT-20/S-IVB stage. These changes would in place of those shown for the baseline (Table 4.2.4.1-I) for the same subsystems.

a. Cryogenic Repressurization System (O_2H_2 burner system).

   Four wire harnesses would be deleted entirely. In addition, another 8 wire harnesses would be reworked with the resulting deletion of 230 wires.

b. APS - Ullage Engines. Rework four wire harnesses and delete 25 wires.

c. Continuous Vent System. A total of 44 wires would be deleted in reworking five wire harnesses.

Since the INT-20 mission imposes considerably less electrical load requirements on the stage than that available on Saturn V/S-IVB stages, a brief investigation was made of replacing the Saturn V stage batteries with the smaller Saturn IB type batteries. It was determined that the design effort required to design a new panel for the installation of these batteries would be considerable, such that the concept would not be cost effective.

4.3.3.5 Alternate Stage Weight

a. Weight Breakdown

   A detailed dry stage weight breakdown for the INT-20/S-IVB alternate stage configuration is presented in Table 4.3.3-I. The reference S-IVB

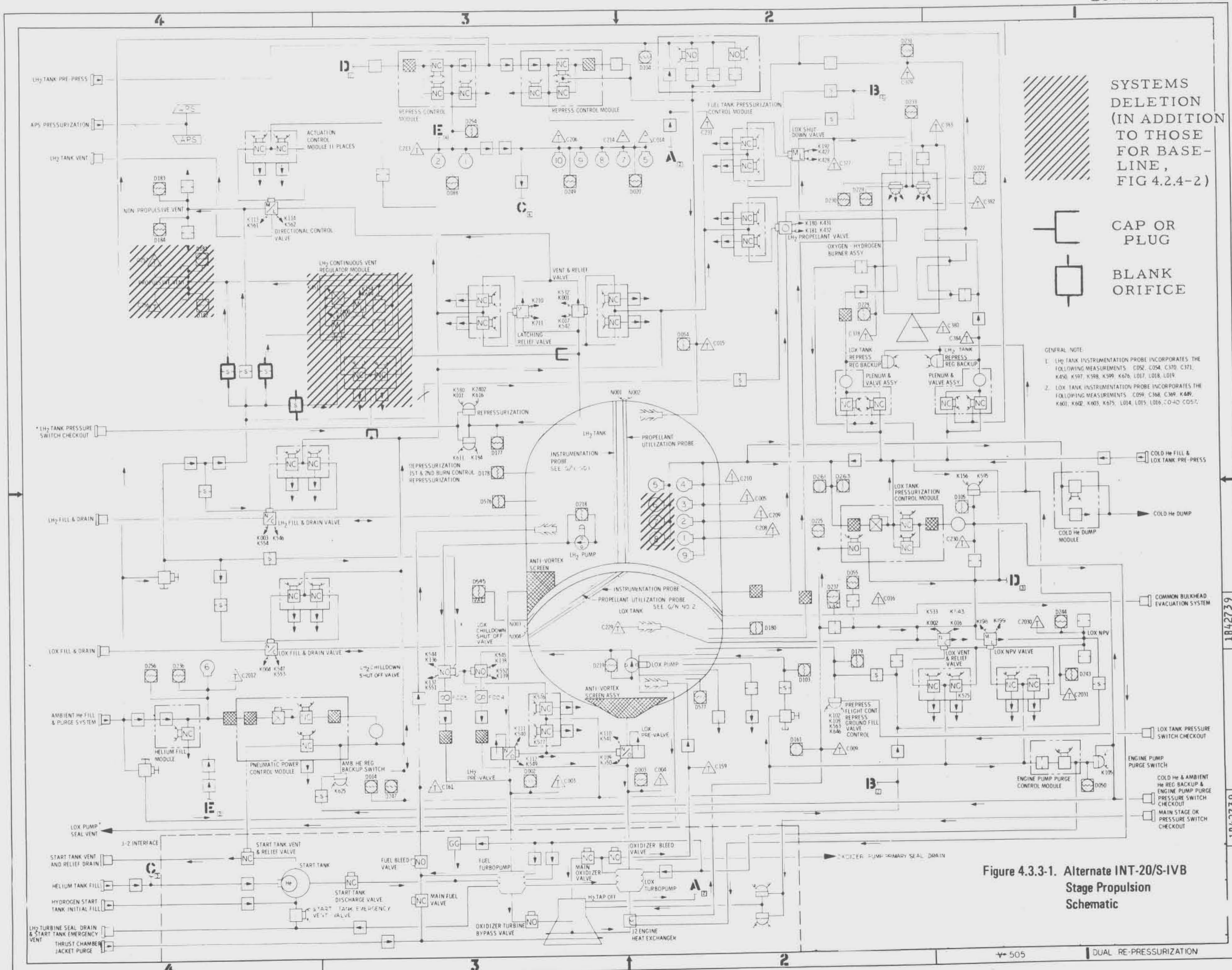


Figure 4.3.3-1. Alternate INT-20/S-IVB Stage Propulsion Schematic

stage is -511, and the resulting weight decrease indicated for the INT-20 alternate stage is 1880 lbs.

The interstage/adapter ring weight summary is the same as that for the baseline configuration (Section 4.2.4.8), so is not repeated here.

b. Weight Substantiation

The substantiation for the weight changes reflected in Table 4.3.3-I is presented below.

W3.3 Propellant Container

Delete existing LH_2 tank baffle and deflector Assembly	-198 Lb
Change to W3.3	-198 Lb

W3.18 Heat & Flame Protection

Delete retrorocket plume impingement curtain installation	-115 Lb
Change to W3.18	-115 Lb

W4.7 Fuel System

Delete (5) ambient helium bottles and plumbing	-665 Lb
Delete Continuous Vent System	-116 Lb
Change to W4.7	-781 Lb

W4.8 Oxidizer System

Delete (2) ambient helium bottles and plumbing	-266 Lb
Delete (1) cold helium bottle and plumbing	- 57 Lb
Change to W4.8	-323 Lb

W4.9 Cryogenic Repress System

Delete O_2H_2 burner and plumbing, supports	-196 Lb
Delete (2) cold helium bottles and plumbing	-114 Lb
Change to W4.9	-310 Lb

W6. 8 Telemetry and Measuring System

Delete telemetry measurements and wiring	-130 Lb
Change to W6. 8	-130 Lb

W6. 16 Auxiliary Propulsion System

Delete (2) ullage engines	- 23 Lb
Change to W6. 16	- 23 Lb

Table 4.3.3-I

INT-20/S-IVB ALTERNATE STAGE DRY WEIGHT SUMMARY

NASA Second Generation Breakdown		S-IVB-511 Reference Stage (lbs)	INT-20/S-IVB Alternate Configuration (lbs)
W3.3	Propellant Container	8,933	8,735
W3.6	Forward of Tanks	1,242	1,242
W3.8	Aft of Tanks	1,816	1,816
W3.9	Thrust Structure	774	774
W3.10	Fairings and Associated Structure	197	197
W3.15	Paint and Sealer	104	104
W3.18	Heat and Flame Protection	182	67
W3.0	Structure	13,248	12,935
W4.1	Engine and Accessories	3,572	3,572
W4.6	Purge System for Chilldown	272	272
W4.7	Fuel System	1,573	792
W4.8	Oxidizer System	1,264	941
W4.9	Cryogenic Repressurization System	310	0
W4.10	Stage Control System Hardware	284	284
W4.0	Propulsion System	7,275	5,861
W6.1	Equipment and Instrumentation Structure	430	430
W6.2	Environmental Control System	231	231
W6.5	Control System Electronics	116	116
W6.8	Telemetry and Measuring System	1,165	1,035
W6.10	P.U. System	175	175
W6.11	Electrical System	829	829
W6.12	Range Safety System	69	69
W6.15	Pneumatic System	298	298
W6.16	Auxiliary Propulsion System	855	832
W6.17	Separation System	117	117
W6.18	Ullage System	212	212
W6.20	Systems for Total Vehicle	91	91
W6.0	Equipment and Instrumentation	4,588	4,435
WAD	Stage Dry Weight	25,111	23,231
Change from S-IVB-511 Baseline		0	-1,880

4.3.4 Improved Flight Control System

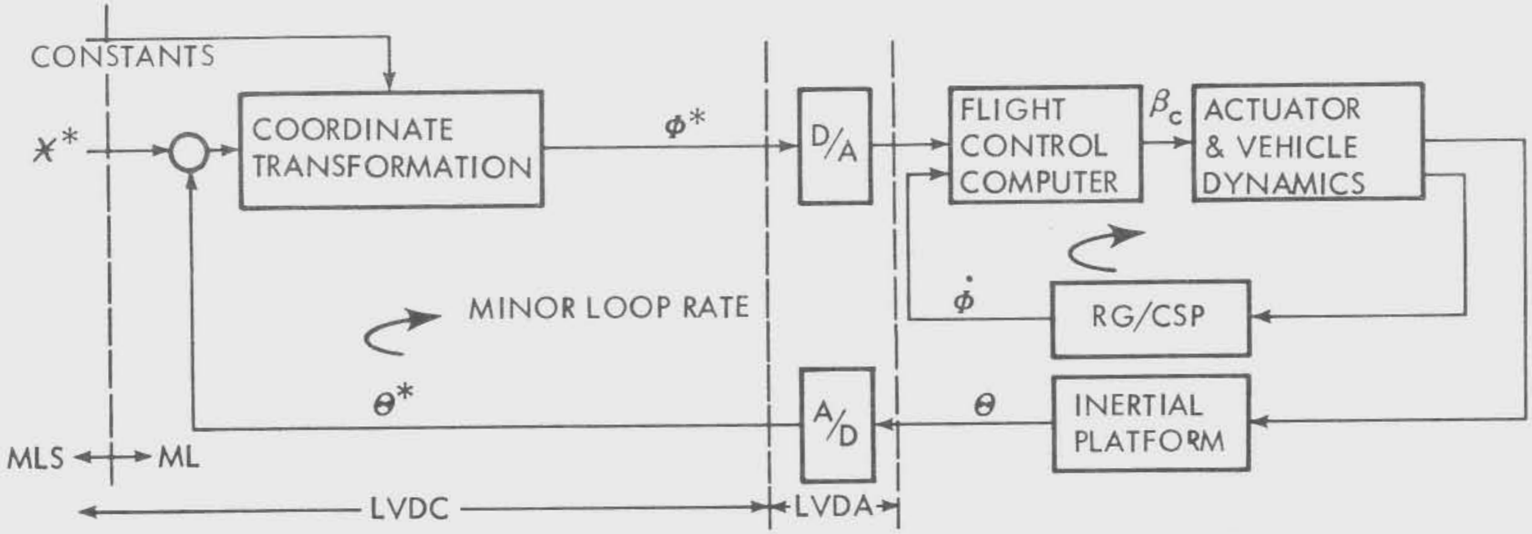
4.3.4.1 Introduction

With the philosophy of minimum-modification to the existing Saturn IU, the concept of digital control for the INT-20 vehicle results in the absorption, into the LVDC, of the control gain program and control loop stabilization functions for the S-IC stage, (pitch, yaw, roll) and S-IVB boost stage (pitch-yaw) control planes. Roll control during boost for the S-IVB stage as well as orbital attitude control will remain as presently implemented. For the control planes affected, the basic attitude control loop functional flow will be altered as shown in Figure 4.3.4.1-1. Whereas the present system solves the control law differential equations in the FCC via various combinations of resistors, capacitors and inductors, the alternate system will utilize difference equations implemented in the LVDC to serve a similar function (commonly referred to as digital filtering). The control gain changes which are presently implemented by relay switches in the FCC would be accomplished by simply changing the static gain of the digital filter. In addition the digital system would now become an all attitude loop with the elimination of the output of the rate gyro/CSP package as a control parameter. The rate damping required for stability would be obtained within the LVDC either by lead compensation or by deriving rate explicitly.

The hardware impact of this change lies in the fact that the removal of the stabilization filters and gain program relays from the FCC results in its sole function being confined to the non-mission dependent role of signal mixing to obtain engine deflection commands. This would allow the FCC to become a standardized piece of hardware for all payload/vehicle configurations of this class of vehicle. Thus it can be seen that a reduction in hardware will be effected through simplification of the system.

As indicated previously, the digital control system will be represented in the LVDC flight program by difference equations. The fact that these equations must be solved at the minor loop rate coupled with limited computation time in the present minor loop, leads to the development of a new LVDC software concept called the split minor loop. This technique allows difference equation computation without causing a major software impact — that of having to change the minor loop cycle time. The split minor loop would sequence the processing so that for one nominal minor loop, a particular control plane attitude signal would be processed and the next minor loop the remaining planes (two for S-IC, one for S-IVB) would be handled. While the minor loop cycle time is unaffected, the sampling period would be changed from 0.04 second to the specified 0.08 second. The 0.08 second sampling period does not hinder stability as will be shown in the response studies section.

PRESENT SYSTEM (TWO-LOOP ANALOG)



PROPOSED SYSTEM (ONE-LOOP DIGITAL)

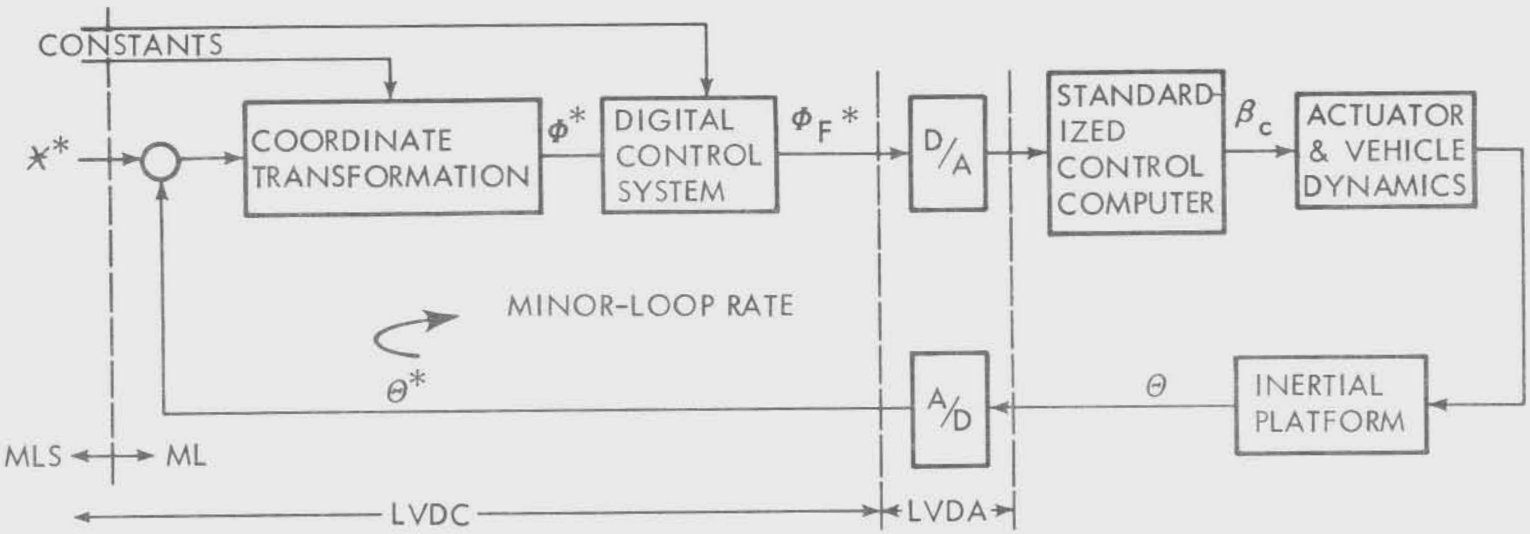


FIGURE 4.3.4.1-1. ATTITUDE CONTROL SYSTEMS

4.3.4.1 (Continued)

It is therefore theoretically possible to change the gain of the digital filter and/or change its dynamic characteristics via difference equation coefficient changes every 0.08 second. This feature allows both the gain program and the filter transfer function to be essentially time variable. This capability does not exist within the present analog system where the gains are discrete levels which are switched at critical times of flight.

This flexibility also leads to what may be the most important feature of the digital control system. As has already been indicated, the formerly mission dependent hardware has been standardized to the point where redefinition of the control system dynamics resulting from a mission/payload change, may be implemented with no hardware modifications. Software changes historically require less impact on the manufacturing and retest cycles.

Due to the attractiveness of such a digital system, this study addresses the feasibility of digital control for the INT-20 vehicle. The study task centers on the following three areas:

Theoretical digital filter design.

Evaluation of control system and vehicle performance via simulation.

Determining facility impact.

The theoretical filter design was accomplished via frozen point W-plane Nyquist responses. Due to the preliminary nature of the design data, the work shown in this area will necessarily supply information concerning only the degree of complexity of the design and the approximate order of the digital filters. Refinement of the digital filter coefficients will be necessary if final vehicle data is significantly different from this preliminary data. However, the feasibility of being able to synthesize a control design for the INT-20 vehicle is adequately shown.

The simulation studies had two main objectives. The first was to verify the frozen point design by observing vehicle performance in the presence of 95 percent profile design winds. The second was to evaluate the system performance using various techniques of software implementation. Minimizing the effects of quantization and guidance non-linearities was the desired objective.

Facility impact centers on modifications to the philosophy of qualification testing of the deliverable flight program, dynamic analysis of the deliverable flight program, dynamic analysis of the guidance and control implementation, acceptance testing of the standardized control computer, and requirements for flight qualification of the digital flight control system.

4.2.4.2 Analytical Design

a. Technical Approach

1. W-Plane Nyquist

Due to the discrete nature of the stabilization equations required for the digital attitude control system, the Nyquist criteria as defined for continuous systems is not directly applicable. Modifications must be made so that the dynamics of the sampling process are included and that the compensator may be considered conveniently in a discrete form. To serve this function a w-plane digital Nyquist program that was previously developed for designing digital filters for the Saturn launch vehicles was used. This program is directly analogous to the s-plane Nyquist program presently used to design analog filters.

The w-plane is the result of two transformations. The first relates the discrete variable, z, to the Laplace operation, s, by

$$z = e^{sT} \tag{1}$$

where T is the sampling interval. The Laplace transform of a sampled signal may be written in the form

$$F^*(s) = \sum_{n=o}^{\infty} f(nT)e^{-nsT} \tag{2}$$

combining Equations (1) and (2) defines the z-transform operation as:

$$F(z) = \sum_{n=o}^{\infty} f(nT)z^{-n} \tag{3}$$

A property of the change of variable is that the primary strip in the s-plane (bounded by $s=\pm j\omega_s/2$) maps into the entire z-plane with the left-half primary strip mapping into the interior of the z-plane unit circle. While the system expressed in z has the desired dynamic properties, no convenient techniques are currently available to evaluate relative stability in this plane. The w-transform was therefore proposed as a means to allow the use of all the well defined continuous-data stability criteria for sampled-data synthesis. It is defined by the change of variable $z = \frac{1+w}{1-w}$. Thus,

$$w = \frac{z-1}{z+1} = \frac{e^{sT/2} - e^{-sT/2}}{e^{sT/2} + e^{-sT/2}} = \tanh\ sT/2 \tag{4}$$

4.3.4.2 (Continued)

This bilinear transformation maps the interior of the z-plane unit circle into the entire left-half w-plane. This fact allows application of linear, continuous-data analysis techniques to the corresponding sampled-data problem; i.e., Routh-Hurwitz, Nyquist, etc. Further, the z to w transformation produces transfer functions which are rational fractions in the variable of interest (frequency in the case of Bode and Nyquist methods application) thereby allowing use of asymptotic plotting techniques. For $s = j\omega$, the complex variable $w = u + jv$ of Equation (4) becomes $u = 0$ and

$$v = \tan wT/2 \tag{5}$$

Equation (5) defines the scaling between the "real" frequency ω and the "fictitious" frequency v.

Therefore, the mechanics of the w-plane Nyquist program involve obtaining the plant dynamics as a function of w (accomplished by the s to z and z to w transformations); combining with the control system dynamics specified in w, and computing the response as a function of v. Having obtained the w-plane response, the design of the digital compensator for the linear sampled system is exactly analogous to design of a continuous compensator of a linear continuous system with the "fictitious" frequency v playing the role of the real frequency ω, where these frequencies are related by Equation (5).

The program has the capability for including in the continuous plant dynamics linearized models of:

rigid body dynamics

flexible body dynamics

propellant slosh.

actuation device.

sensor dynamics

sample and hold phenomena

4.3.4.2 (Continued)

with the choice of stage and/or control plane determining the type and complexity of each model. The equations of motion for the S-IC stage pitch-yaw plane are summarized in Table 4.3.4.2-I. The S-IC stage, roll plane equations are given in Table 4.3.4.2-II. The S-IVB stage, pitch-yaw axis model contains equations similar to those in Table 4.3.4.2-I.

2. Design Criteria

As previously mentioned, the transformation of the linear discrete system to the w-plane allows the application of continuous system Nyquist criteria. Therefore, the following list of design objectives based on previous Saturn experience was formulated:

6 db aerodynamic gain margin

6 db rigid body gain margin

minimum of 30^{o} rigid body phase margin

phase stabilization of lowest frequency bending mode (S-IC stage only) with minimum phase margins of $\pm$ 60 degrees

gain stabilization of higher bending modes with a minimum of 6 db attenuation

Definition of these stability margins is shown in Figure 4.3.4.2-1.

b. Data Reduction

1. S-IC Stage, Pitch-Yaw

The basic design data for these control planes were obtained from Reference 3.1.3.6-1. Mass distribution data, aerodynamic data, bending data for three flight times, and slosh data for four flight times were available from this source. Since achieving the slosh stability objectives proved to be difficult, slosh data at additional flight times were improvised using the propellant loading and longitudinal acceleration information from Reference 3.1.3.6-1 and the slosh parameter curves in Reference 3.1.3.6-2.

Tables 4.3.4.2-III through 4.3.4.2-VI summarize the rigid body, bending and slosh coefficients and the actuator transfer function used in the analysis. Comparison of these with similar data for the Saturn V (Reference 3.1.3.6-3) reveals the following facts pertinent to the control system design.

TABLE 4.3.4.2-I. S-IC AND S-IVB STAGE PITCH-YAW EQUATIONS OF MOTION

Rigid Body Moment Equation

$$\ddot{\varphi}_R = -C_1\alpha - C_2\beta_e$$

Bending Equations

$$\ddot{\eta}_i + 2\zeta_i\omega_i\dot{\eta}_i + \omega_i^2\eta_i = \frac{1}{m_i}\left[(\Sigma_E Y_{i\beta} + \theta_E Y'_{i\beta})\ddot{\beta}_e + R'Y_{i\beta}\beta_e\right.$$

$$\dot{\varphi}_B = -\sum_{i=1}^{n} Y'_{iRG}\dot{\eta}_i$$

$$\varphi_B = -\sum_{i=1}^{n} Y'_{i\varphi}\eta_i$$

Fuel Slosh Equations

$$\ddot{\xi}_{sj} + 2\zeta_{sj}\omega_{sj}\dot{\xi}_{sj} + \omega_{sj}^2\xi_{sj} = \overline{X}_{sj}(\ddot{\varphi}_R + \ddot{\varphi}_s) + K_3\varphi_T - \ddot{Z}$$

$$\ddot{\varphi}_s = \sum_{j=1}^{n} \frac{m_{sj}}{I_{XX}}(\overline{X}_{sj}\ddot{\xi}_{sj} + K_3\xi_{sj})$$

Normal Acceleration of Vehicle c.g.

$$\ddot{Z} = K_4\beta_e + K_3\varphi_T + K_7\alpha - \sum_{j=1}^{n}\frac{m_{sj}}{m}\ddot{\xi}_{sj}$$

Miscellaneous Equations

$$\dot{\varphi}_T = \dot{\varphi}_R + \dot{\varphi}_s + \dot{\varphi}_B$$

$$\varphi_T = \varphi_R + \varphi_s + \varphi_B$$

$$\alpha = \varphi_T$$

NOTE: See Appendix C for definitions of control parameters used in the equations.

TABLE 4.3.4.2-II. S-IC STAGE ROLL EQUATIONS OF MOTION

Rigid Body Moment Equation

$$\ddot{\varphi}_R = -C_{2R}\beta_e$$

Torsional Bending Equation

$$\ddot{\tau}_i + 2\zeta_{ri}\omega_{ri}\dot{\tau}_i + \omega_{ri}^2\tau_i = \frac{1}{m_{ri}}\left(\Sigma_E\, \ell_{\beta R}\,\theta_{ib}\ddot{\beta}_e + R'\,\ell'_{\beta R}\,\theta_{iB}\beta_e\right)$$

$$\dot{\varphi}_\tau = -\sum_{i=1}^{n}\theta_{iRG}\dot{\tau}_i$$

$$\varphi_\tau = -\sum_{i=1}^{n}\theta_{i\varphi}\tau_i$$

Fuel Slosh Equation

$$\ddot{\xi}_{sj} + 2\zeta_{sj}\omega_{sj}\dot{\xi}_{sj} + \omega_{sj}^2\xi_{sj} = d_{Ej}(\ddot{\varphi}_R + \ddot{\varphi}_s)$$

$$\ddot{\varphi}_s = \sum_{j=1}^{m}\frac{m_{sj}d_{Ej}}{I_{RR}}\ddot{\xi}_{sj}$$

NOTE: See Appendix C for definitions of control parameters used in the equations.

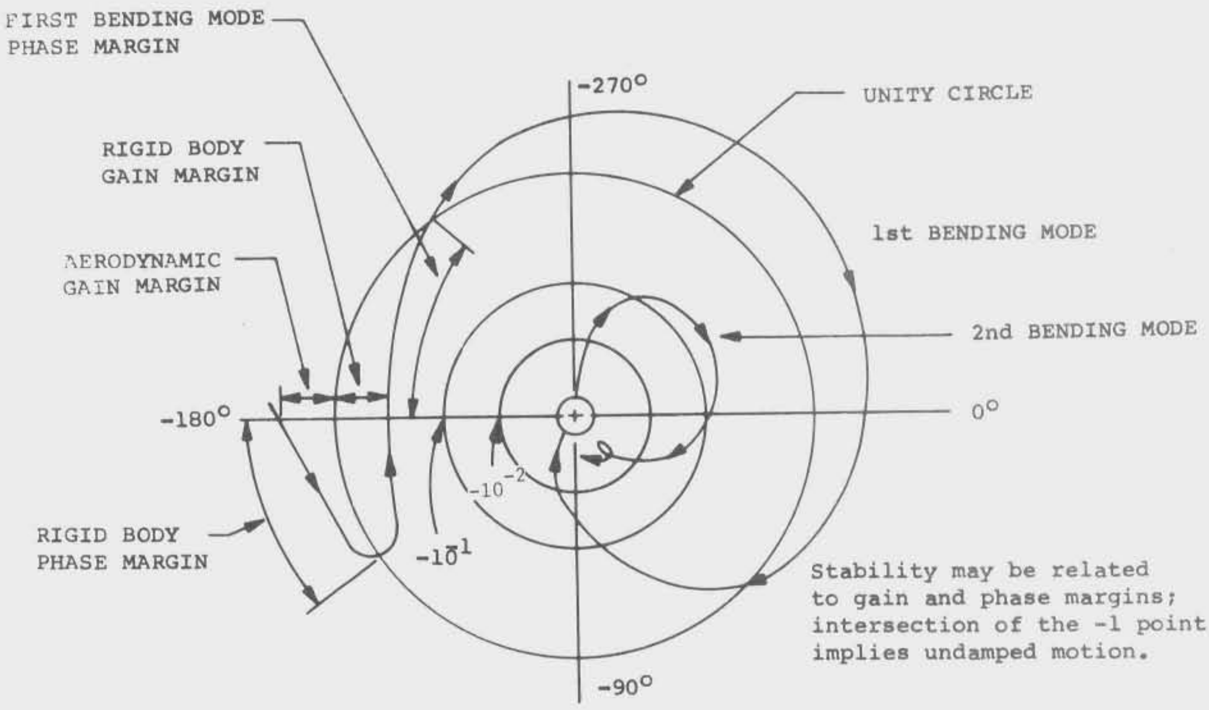


FIGURE 4.3.4.2-1. GAIN AND PHASE MARGIN DEFINITIONS WITH RESPECT TO W-PLANE NYQUIST PLOT OF ATTITUDE CONTROL SYSTEM MODEL OPEN LOOP FREQUENCY RESPONSE (OPEN AT THE ACTUATOR)

TABLE 4.3.4.2-III. S-IC STAGE, PITCH-YAW

Rigid Body Coefficients

Time (sec)	C_1 ($1/sec^2$)	C_2 ($1/sec^2$)	K_3 (m/sec^2 rad)	K_4 (m/sec^2 rad)	K_5 (sec-rad/m)	K_7 (m/sec^2-rad)
0.0	0.0	1.86	12.26	12.26		0.0
24.0	-.0042	1.94	13.90	14.03	.0124	.632
43.0	-.0251	2.05	15.72	16.12	.00562	2.95
59.0	+.041	2.15	17.31	18.44	.00342	7.39
70.1	-.161	2.23	18.30	20.35	.00254	8.83
79.0	-.351	2.30	20.15	22.02	.00202	9.17
99.0	-.427	2.45	25.6	26.41	.00124	7.58
110.0	-.288	2.55	29.08	30.10	.000945	5.43
130.0	-.114	2.77	36.53	37.44	.000625	2.64
146-0	-.033	3.02	45.74	45.79	.000465	1.18
146+0	-.033	1.51	22.86	22.89	.000465	1.18
179.0	-.0013	1.84	30.70	30.71	.000342	.08
211.0	-5.3×10^{-6}	2.54	45.86	45.86	.000245	.001

Actuator Transfer Function

$$W_{ss} = \frac{1.0}{.83745 \times 10^{-12} s^6 + .61362 \times 10^{-9} s^5 + .19388 \times 10^{-6} s^4 + .21398 \times 10^{-4} s^3 + .15119 \times 10^{-2} s^2 + .43546 \times 10^{-1} s + 1.0}$$

TABLE 4.3.4.2-IV. S-IC STAGE, PITCH-YAW

Bending Characteristics

Time (sec)	Mode	f_i(HZ)	M(Kg)	ζ_i	Y_{iQ}	$Y_{iQ'}(\frac{1}{M})$	$Y_{iP'}(\frac{1}{M})$	$Y_{iR'G}(\frac{1}{M})$
	1st	1.56	49596.	.005	.2	+.0102	-.0394	-.0394
	2nd	3.28	64623.	.005	-.14	-.0122	-.0638	-.0638
T=0	3rd	5.21	298997.	.005	.48	+.0453	-.0677	-.0677
	4th	7.26	257439.	.005	1.0	+.1142	+.0138	+.0138
	1st	1.70	52171.	.005	.23	+.0146	-.0427	-.0427
	2nd	3.31	58318.	.005	-.14	-.00689	-.0602	-.0602
T=70	3rd	7.85	517278.	.005	1.0	+.1378	-.04016	-.04016
	4th	9.21	173115.	.005	1.0	+.2087	+.0118	+.0118
	1st	2.31	31313.	.005	.16	+.1102	-.053	-.053
	2nd	8.19	48195.	.005	.34	+.0748	+.0401	+.0401
T=211	3rd	12.0	10840.	.005	.04	+.0118	-.01968	-.01968
	4th	18.9	38003.	.005	.2	+.0748	-.0512	-.0512

TABLE 4.3.4.2-V. S-IC STAGE, PITCH-YAW

Sloshing Characteristics

Time (sec)	Tank	f_{sj} (HZ)	ζ_{sj}	M_{sj} (Kg)	ℓ_{sj} (M)
	S-IC Lox	.336	0.025-0.057	203840.1	-5.46
	S-IC RP-a	.339	0.023-0.058	61467.7	8.94
T=0	S-IVB Lox	.6	.04	6345.9	-24.36
	S-IVB LH_2	.415	.001	3523.1	-28.16
	S-IC Lox	.411	0.025-0.057	206610.3	-1.2
	S-IC RP-a	.407	0.023-0.058	139690.7	15.2
T=70	S-IVB Lox	.732	.04	6345.9	-25.5
	S-IVB LH_2	.508	.001	3523.1	-29.3
	S-IC Lox	.431	0.025-0.057	206115.8	-.5
	S-IC RP-1	.425	0.023-0.058	138989.8	15.4
T=79	S-IVB Lox	.77	.04	6345.9	-25.5
	S-IVB LH_2	.533	.001	3523.1	-29.3
	S-IC Lox	.461	0.025-0.057	206115.	+.05
	S-IC RP-1	.45	0.025-0.057	135500.	15.7
T=91	S-IVB Lox	.829	.04	6345.9	-25.5
	S-IVB LH_2	.57	.001	3523.1	-29.3
	S-IC Lox	.515	0.025-0.057	196000.	1.25
	S-IC RP-1	.495	0.025-0.057	129000.	16.3
T=110	S-IVB Lox	.921	.04	6345.9	-25.4
	S-IVB LH_2	.64	.001	3523.1	-29.2

TABLE 4.3.4.2-VI. S-IC STAGE, PITCH-YAW

Sloshing Characteristics					
Time (sec)	Tank	f_{sj} (HZ)	ζ_{sj}	M_{sj} (Kg)	ℓ_{sj} (M)
	S-IC Lox	.56	0.025-0.057	182000.	2.95
	S-IC RP-1	.52	0.025-0.057	113000.	17.39
T=130	S-IVB Lox	1.04	.04	6345.9	-24.7
	S-IVB LH_2	.717	.001	3523.1	-28.46
	S-IC Lox	.582	0.025-0.057	158000.	4.55
	S-IC RP-1	.53	0.025-0.057	91000.	18.69
$T=146^-$	S-IVB Lox	1.16	.04	6345.9	-23.61
	S-IVB LH_2	.8	.001	3523.1	-27.41
	S-IC Lox	.406	0.025-0.057	158000.	4.55
	S-IC RP-1	.373	0.025-0.057	91000.	18.69
$T=146^+$	S-IVB Lox	.82	.04	6345.9	-23.61
	S-IVB LH_2	.56	.001	3523.1	-27.41
	S-IC Lox	-	-	0	-
	S-IC RP-1	-	-	0	-
T=210.9	S-IVB Lox	1.16	.04	6345.9	-16.6
	S-IVB LH_2	.805	.001	3523.1	-20.4

4.3.4.2 (Continued)

The first bending mode frequency at liftoff (which is normally the time of lowest first mode frequency) for the INT-20 is significantly larger than the corresponding mode of the Saturn V 1.56 cps to 0.9 cps). This difference increases as the flight progresses. Separation of the control frequency from the bending frequency is not as severe a problem as experienced on the Saturn V control system design. Therefore, an increase in rigid body phase and gain margins can be expected.

The ratio of the control moment coefficient (C_2) to the aerodynamic moment coefficient (C_1) drops to a slightly lower value for the INT-20 vehicle during the region of high q. Some of the increased stability margins implied in the previous statements must therefore be sacrificed in order to maintain comparable aerodynamic gain margins.

2. S-IC Stage, Roll

Reference 3.1.3.6-1 contains the mass distribution data required for the stability analysis of this control plane. However, there is presently no torsion data available for the INT-20 vehicle. To facilitate analysis, torsion data corresponding to the Saturn V, S-IC stage, roll plane (Reference 3.1.3.6-3) was used. It is felt that this substitution will not invalidate the study results, since the INT-20 will be shorter, stiffer causing higher frequency torsional modes. Therefore, this torsion should represent a worse case than anticipated in INT-20 data.

Table 4.3.4.2-VII depicts the roll data used. Comparison with Saturn V, S-IC stage roll plane data indicates that the INT-20 S-IC stage roll plane will have (due to a reduction in the roll plane moment of inertia) about 10% more control authority available.

3. S-IVB Stage, Pitch and Yaw

Limited data were available in all areas for this stage, however, a sufficient amount was obtained to allow the system to be studied at ignition and cutoff. Knowledge gained through experience on the Saturn program indicates that a design of the S-IVB control system based on ignition and cutoff alone would be adequate since no significant vehicle dynamics changes take place during the burn. Mass characteristics were obtained from Reference 3.1.3.6-1. As no bending data were available on the baseline INT-20, Reference 3.1.3.6-4 data presented the ignition and

TABLE 4.3.4.2-VII. S-IC STAGE, ROLL

Rigid Body Parameters and Coefficients

Time (sec)	I_{RR} ($Kg.m^2$)	$R' \times L_{BR}$* (Newt*m)	C_{2R} ($1/sec^2$)
0	3.44×10^6	1.25189×10^8	36.39
24	3.44×10^6	1.26938×10^8	36.90
43	3.44×10^6	1.310222×10^8	38.09
59	3.44×10^6	1.356204×10^8	39.42
70	3.44×10^6	1.38674×10^8	40.31
79	3.44×10^6	1.40663×10^8	40.89
91	3.44×10^6	1.42469×10^8	41.41
99	3.44×10^6	1.431148×10^8	41.60
123	3.44×10^6	1.43771×10^8	41.79
146-	3.44×10^6	1.43855×10^8	41.82
146+	3.44×10^6	0.719276×10^8	20.91
163	3.44×10^6	0.719325×10^8	20.91
179	3.44×10^6	0.719333×10^8	20.91
211	3.44×10^6	0.719334×10^8	20.91

*L_{BR} = 4.6228 m.

TABLE 4.3.4.2-VII. S-IC STAGE, ROLL (Continued)

Torsional Characteristics

Time	Mode	f_{ri} (HZ)	GJ_r ($Kg \cdot M^2$)	ζ_{ri}	Θ_{ib}	Θ_{iQ}
	1	6.03	4402.18	0.005	-0.012	.0543
0.0	2	7.99	2723.92	0.005	0.0171	-.0728
T.0	3	9.66	1754.10	0.005	-0.0115	.0191
211	4	10.92	22562.8	0.005	0.062	.1004
	5	13.56	70789.38	0.005	0.0089	-.1203
	6	13.87	197884.	0.005	-0.0582	-.1039

4.3.4.2 (Continued)

cutoff bending characteristics for the INT-20 vehicle that was studied in 1966 as part of contract NAS 8-20266. Slosh data at both flight times were improvised using Reference 3.1.3.6-2 and the best available information concerning propellant loading and longitudinal acceleration.

Table 4.3.4.2-VIII lists the design data for this stage. The important control system coefficients and parameters for the two flight times given are very similar to Saturn V S-IVB Stage values.

c. Results of the Stability Analysis

Using the analytical techniques and design data previously presented, digital stabilization filters and associated gain programs were developed for pitch, yaw, and roll control planes of the S-IC Stage and pitch-yaw for the S-IVB. A summary of the resulting stability margins are presented in Table 4.3.4.2-IX (w-plane Nyquist plots are included in Appendix C.1, Figure C.1-1 through C.1-13). The AS-504 (with present analog control system) stability margins are included for comparison. The results indicate that the stability margins for the INT-20 vehicle are comparable to, and in most cases better than, the AS-504 stability margins.

Table 4.3.4.2-X displays the form of the digital filter and the necessary coefficients for the S-IC stage pitch/yaw and roll filters and the S-IVB stage pitch/yaw filter.

The recommended gain profiles (K versus time) are shown in Figure 4.3.4.2-2 where the dependent variable, K, is defined in the preceding table. The S-IC Stage, pitch-yaw profile indicates a significant reduction in gain following the high dynamic pressure portion of the trajectory. Fuel slosh characteristics were the primary motivation for the particular form of reduction shown. Due to the extremely large vehicle longitudinal accelerations during this portion of flight, the slosh natural frequencies obtain a level that would be detrimental to stability if the control loop gain was not lowered. Following cutoff of two outboard engines at T = 146 seconds, there is a corresponding increase in gain. As the engine cutoffs reduce the longitudinal acceleration, the slosh frequencies drop to a level where sloshing dynamics no longer play a dominant role in the gain selection. It was determined that the S-IC Stage roll plane gain could be piecewise constant with the only gain change necessary occurring when the two outboard engines are cutoff. As the S-IVB design was based only on the boundary times of the trajectory lifetime, its gain program is shown as a simple ramp from ignition to cutoff. This off course is subject to further analysis as a more complete data package becomes available.

TABLE 4.3.4.2-VIII. S-IVB STAGE, PITCH-YAW

Rigid Body Parameters and Coefficients

	$I_{XX}(Kg \cdot M^2)$	$X_{cg}(M)$	R'(Newtons)	M(Kg)	$C_2(\frac{1}{sec^2})$	$K_4(\frac{M}{rad \cdot sec^2})$
T = 0	1.24×10^7	9.87	911885.	178585.1	.7258	5.11
T = 473	4.74×10^6	17.31	911885.	75473.7	3.33	12.1

Bending Characteristics

Time (sec)	Mode	$f_i(HZ)$	$\bar{M}_i(Kg)$	ζ_i	$\bar{Y}_{iQ}$	$Y'_{iQ}(\frac{1}{M})$	$Y'_{iP}(\frac{1}{M})$	$Y'_{iRG}(\frac{1}{M})$
	1st	2.88	3671.5	.005	.05	.0106	.0098	.0098
	2nd	6.48	4558.3	.005	-.045	-.0268	-.0039	-.0039
T = 0	3rd	11.52	18592.97	.005	.185	.1259	-.1575	-.1575
	4th	13.61	29707.5	.005	-.21	-.1772	.325	.325
	1st	3.49	2595.9	.005	.0355	.0268	.0275	.0275
	2nd	7.18	5631.5	.005	-.45	-.0472	-.0453	-.0453
T = 473	3rd	12.63	14048.5	.005	.34	.0051	.0138	.0138
	4th	45.24	141341301.6	.005	8.0	-20.08	+8.66	8.66

Sloshing Characteristics

Time	Tank	$f_{sj}(HZ)$	ζ_{sj}	$M_{sj}(Kg)$	$\ell_{sj}(M)$
T=0	Lox	.39	.04	6345.9	3.24
	LH_2	.27	.001	3523.1	-.56
T=473	Lox	.314	.00026	2989.	15.71
	LH_2	.154	.00067	391.2	13.15

TABLE 4.3.4.2-VIII. S-IVB STAGE, PITCH-YAW (Continued)

Actuator Transfer Function

$$W_{SS} = \frac{1.0}{.11671 \times 10^{-8} S^5 + .73902 \times 10^{-6} S^4 + .18779 \times 10^{-3} S^3 + .54782 \times 10^{-2} S^2 + .10815 S + 1.0}$$

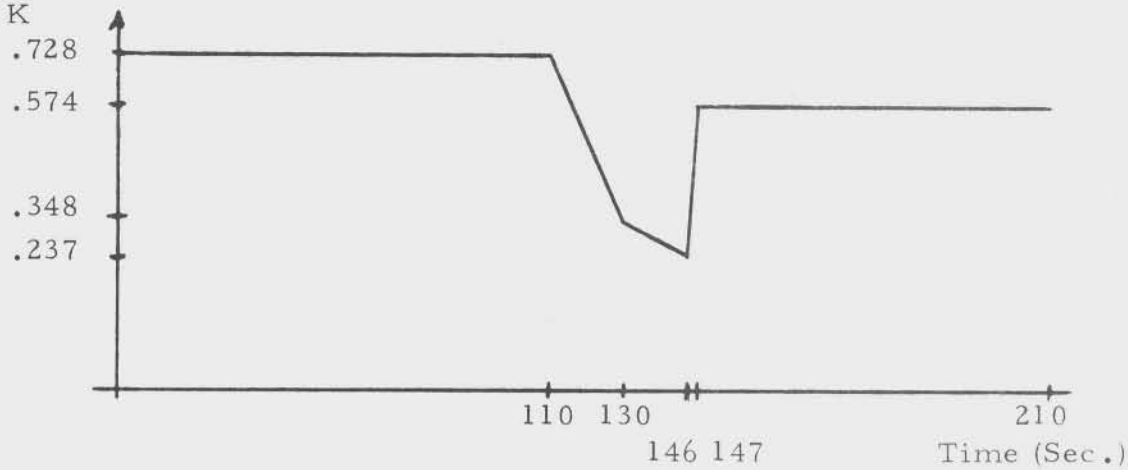
TABLE 4.3.4.2-IX. COMPARISON OF MINIMUM STABILITY MARGINS FOR THE INT-20 S-IC STAGE AND AS-504 S-IC STAGE

	S-IC STAGE				S-IVB STAGE	
	P - Y		ROLL		P - Y	
Stability Margin	AS-504	INT-20	AS-504	INT-20	AS-504	INT-20
Aerodynamic Gain Margin (db)	5.8	6.0	-	-	-	-
Rigid Body Gain Margin (db)	8.7	6.0	9.0	9.9	5.2	7.5
Rigid Body Phase Margin (deg.)	26.6	30.0	37.0	32.0	31.7	35.0
Slosh Gain Margin (db)	1.5	6.0	-	-	-	-
Lox Phase Margin (deg.)	-	-	-	-	25.5	86.0
First Bending Mode Phase Margin (deg.)	59.3	72.0	-	-	-	-
Second Bending Mode Gain Margin (db)	6.3	14.0	-	-	-	-

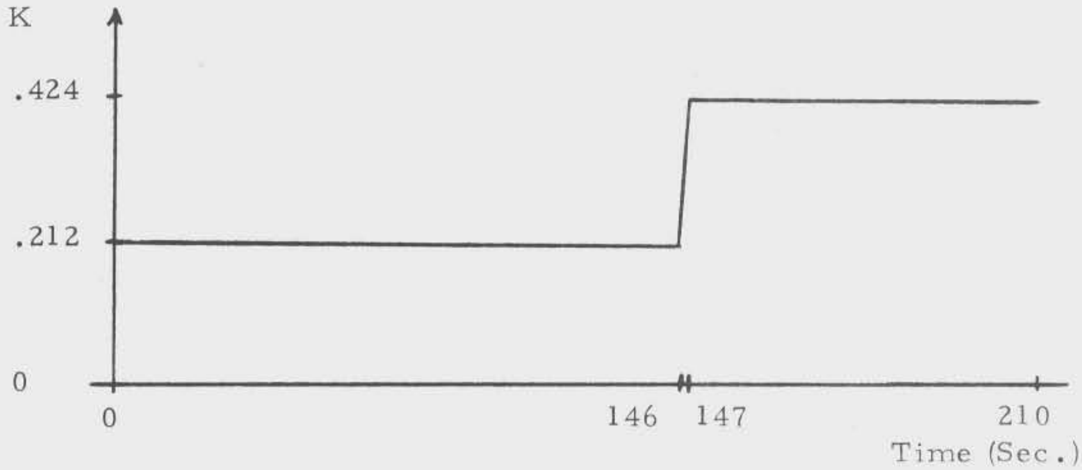
TABLE 4.3.4.2-X. DIGITAL FILTER COEFFICIENTS

Coefficient	S-IC STAGE P - Y	S-IC STAGE ROLL	S-IVB STAGE P - Y
P_0	1.00000	1.00000	1.00000
P_1	-0.79442	-0.52994	-1.84321
P_2	-0.99291	-0.95117	0.13282
P_3	0.80151	0.57877	1.84404
P_4	0.00000	0.00000	0.86634
Q_0	1.00000	1.00000	1.00000
Q_1	-2.18576	-1.30031	-3.03191
Q_2	1.76949	0.68452	3.50154
Q_3	-0.55537	-0.13512	-1.82236
Q_4	0.00000	0.00000	0.35944

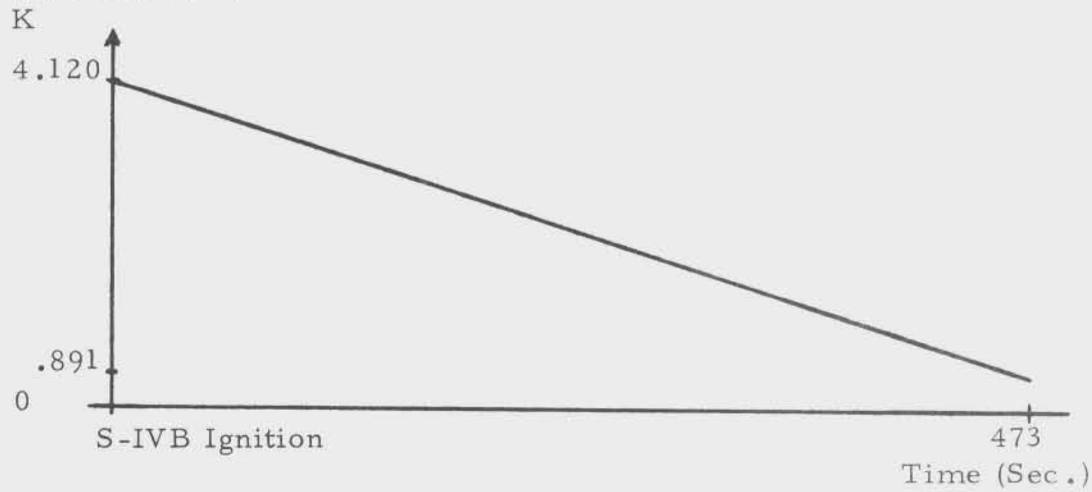
$$F(Z) = K \frac{P_0 + P_1 Z^{-1} + P_2 Z^{-2} + P_3 Z^{-3} + P_4 Z^{-4}}{Q_0 + Q_1 Z^{-1} + Q_2 Z^{-2} + Q_3 Z^{-3} + Q_4 Z^{-4}}$$



(a) S-IC, Pitch-Yaw



(b) S-IC, Roll



(c) S-IVB, Pitch-Yaw

NOTE: See Table 4.3.4.2-X for Definition of Parameter K

FIGURE 4.3.4.2-2. CONTROL GAIN PROFILES

4.3.4.3 Simulated Performance Analysis

a. Description of Simulator

The intent of the simulation studies was to demonstrate the operation of the S-IC stage digital control system in the presence of realistic disturbances and to evaluate the effects of certain system nonlinearities upon vehicle performance. Therefore, an existing all-digital simulator (called BOOSTR) previously used for Saturn response studies was modified to include the effects of a digitally-controlled INT-20 vehicle. The original BOOSTR program simulates a vehicle model described by a set of planar equations of motion for each axis, which includes rigid body, bending, sloshing, and lateral acceleration dynamics. An ideal platform is simulated along with an actuator model that includes position and rate limiting. All vehicle data are time varying. Wind velocity, speed of sound, and air density are interpolated as functions of altitude. Control law implementation includes the effects of having a digital computer in the control loop; i.e., quantization and computational delay. For the INT-20 study, the following disturbances were included:

Pitch plane guidance profile given in Reference 3.1.3.6-1.

AS-505 tower clearance maneuver.

AS-505 roll maneuver.

Various design wind profiles.

The major changes made to BOOSTR to adapt it for the INT-20 were:

Shutting down two outboard engines at T = 146 seconds of flight time.

Inclusion of digital filters in the minor loop.

Inclusion of time-varying rather than step-changing control gains.

Providing alternate quantum level in commanded engine deflection.

b. Nominal Response to 95% Design Winds

The vehicle wind response analysis was performed utilizing the digital simulation described in the preceding paragraphs. Two wind profiles (obtained from Reference 4.3.4.3-1) were used: 95% February winds and 95% May winds.

4.3.4.3 (Continued)

The 95% February winds were used to determine vehicle performance under worst case conditions (largest wind velocities). Responses to these winds are given in Appendix C.2, Figures C.2-1 through C.2-10. The maximum excursions of important system parameters are given below:

Maximum engine deflection (β_e): Pitch = -0.56°, Yaw = 0.58°.

Maximum attitude error (ψ): Pitch = -1.28°, Yaw = 0.74°, Roll = 0.46°.

Maximum attitude rate ($\dot{\phi}$): Pitch = -0.95°/s, Yaw = -1.1°/s, Roll = 1.42°/s.

Maximum angle of attach (α): Pitch = -9.8°, Yaw = 8.4°.

Responses to 95% May winds are also included in Appendix C.2, Figures C.2-11 through C.2-20. These responses were obtained to allow comparison of the INT-20 vehicle control system performance with that of the present Saturn V (February wind responses not being available with the Saturn V). Table 4.3.4.3-I summarizes the pitch plane responses of the two vehicles. Since the trajectory for each vehicle is different, exact control variable comparisons are meaningless. However, as the two trajectories are of the same form, the small deviations noted in the table verify the acceptability of the INT-20 control system performance.

c. Alternate Implementations

Inspection of the nominal responses discussed in the preceding section indicates three undesirable characteristics which are inherent in this form of digital control. These are:

Low level limit cycling in the roll plane attitude error (shown in Figure C.2-3).

Noisy engine deflection (Figures C.2-9 and C.2-10).

Engine deflection transient response to a guidance command discontinuity (Figures C.2-9 and C.2-10).

Before discussing these characteristics further and posing procedures for improving them, it should be noted that these phenomena do not rule out the use of digital control on the INT-20 vehicle. Rather, this section discusses the results of an attempt to minimize the effects of disturbances and nonlinearities on system response.

TABLE 4.3.4.3-I. COMPARISON OF INT-20 AND SATURN V CONTROL VARIABLES FOR 95% MAY WINDS

		INT-20	Saturn V (AS-505 Data)
Magnitude of Maximum Engine Deflection (β_e) (Pitch Plane)	Deg	0.46	0.52
Magnitude of Maximum Attitude Error (ψ) (Pitch Plane)	Deg	1.04	0.96
Magnitude of Maximum Attitude Rate ($\dot{\phi}$) (Pitch Plane)	Deg/Sec	0.90	0.84
Magnitude of Maximum Angle of Attach (α) (Pitch Plane)	Deg	4.40	5.10

4.3.4.3 (Continued)

Attitude error limit cycle and engine gimbal angle noise result from quantization inherent in A/D and D/A conversion of the control variables. The quantizing process is a nonlinear operation which yields a discrete set of output amplitude levels for a continuous range of input signals.

Therefore, while an input signal is within a quantum level, the system is insensitive to small variations in the signal and the control loop is essentially opened. The vehicle will diverge until a quantum level is broached, then a discrete jump equal to the quantum level will be seen at the output. In nominal operation, the A/D quantization (on vehicle attitude, θ) is 0.00279 degrees and the D/A quantum level (on beta command, β_c) is 0.06 degrees. Since the D/A quantization is much larger, it is the primary cause of the limit cycle and noise characteristics.

The engine displacement transients are produced because a lead-type digital filter (differentiation over a certain frequency range as required for rigid body stabilization) is employed in the feed forward path following the guidance command. Therefore, when there is a discontinuity in the command signal, an instantaneous spike will be seen at the output of filter.

In an attempt to improve these undesirable characteristics, the following two techniques were investigated:

Rescaling within the LVDC to obtain a finer quantum level on the commanded engine gimbal angle (β_c).

Altering the difference equation implementation to eliminate differentiation of the guidance command.

Reducing the quantum level on the commanded engine gimbal angle is possible with a minimum modification. The present LVDA output range is ± 12.24 volts or ± 15.3 degrees, the scale factor being 0.8 volt/degree. Using digital control, the output signal is the commanded engine deflection, and as such, the maximum required range would be ± 5.15 degrees for the S-IC stage. Changing the output scale factor to 2.4 volts/degree would reduce the D/A quantum level from 0.06 to 0.02 degree. However, this change introduces an effective limit on the rate of change of the engine deflection command (β_c).

The present software has an 0.48-degree limit on the amount of change in attitude error than can occur from one computation cycle to the next. With a 0.04-second minor loop cycle period, the rate of change of the attitude error is limited to 12 degrees/second. This 0.48 degree corresponds to the amount

4.3.4.3 (Continued)

of voltage set as the discompare level of the coarse comparator in the LVDA. A hardware modification is necessary to change this voltage level. With this voltage level and the indicated rescaling, this software limit becomes 0.16 degree. As a new engine command will be computed every 0.08 seconds, a 2 degree/second limit on the commanded engine signal would result. With appropriate software logic, this limit may be raised to 4 degrees/second. This is possible because the ladders will continue to be updated every 0.04 second with every other update being a past value of the previously computed engine command. A test will be added to the program to determine if the signal is going to be limited. If it is, the amount by which it will exceed the limit is saved, passed through the limiter again, and the resultant excess applied as an additional engine command 0.04 second later. In nominal flight, even the 2 degree/second limit would never be approached. However, it is possible that in certain failure modes the engine rate limit phenomena would be undesirable.

Digital simulation results for the control system employing the fine quantum level on β_c are displayed in Table 4.3.4.3-II and are compared to the simulation results obtained using the original quantum level on β_c (0.06°). Pertinent parameter responses to 95% February winds are included in Appendix C.2, Figures C.2-21 through C.2-30.

With the nominal vehicle attitude, θ , quantum level (0.00279) reducing the β_c quantum level to 0.0202° reduces the roll attitude error noise level from 0.28 to 0.09 degrees and reduces the steady-state engine gimbal angle noise level from 0.06 to 0.02 degrees. When the backup resolver quantum level on θ (0.089°) is used, no differences in attitude error or engine gimbal angle steady-state noise level are displayed. The improved nominal operation response emphasizes the necessity for investigating the incidental impacts of rescaling the LVDA output.

To eliminate the effect of differentiating a discontinuity in the command signal and still retain the necessary lead required for rigid body stabilization, explicit rate derivation was investigated. Figure 4.3.4.3-1 is a block diagram of the attitude control system employing rate derivation. The transfer function for the rate derivation was derived in the w-plane and is of the form of a band limited differentiator thereby providing the necessary lead in the low frequency rigid body region and gain reduction in the high frequency bending mode region. The transfer function for the rate derivation block is:

TABLE 4.3.4.3-II. SUMMARY OF PERFORMANCE ANALYSIS RESULTS FOR DIFFERENT CONTROL SYSTEM IMPLEMENTATIONS

		LEAD FILTER		LEAD FILTER REDUCED QUANTIZING LEVEL ON β_e		DERIVED RATE	
Signal Conversion Quantum Levels (Deg.)	β_c	.06000	.06000	.02020	.02020	.02020	.02020
	θ	.00279	.08900	.00279	.08900	.00279	.08900
Magnitude of Peak Transient on β_e Due to Initiation of Guidance (Deg.)	P	.18	.18	.18	.28	.06	.08
	Y	.60	.60	.58	.62	.22	.23
Magnitude of Peak Steady State Noise Level on β_e	P	.06	.06	.02	.06	.02	.18
	Y	.06	.18	.02	.18	.02	.18
Magnitude of Peak Steady State Noise Level on ψ, Roll (Deg.)		.280	.089	.090	.089	.100	.089
Magnitude of Peak Attitude Error ψ (Deg.)	P	1.38	1.43	1.40	1.40	1.80	1.85
	Y	0.74	0.78	0.75	0.78	1.30	1.15
	R	0.46	0.45	0.42	0.44	0.69	0.72
Magnitude of Peak Pitch Plane Engine Deflection (Deg.)		.56	.60	.54	.58	.62	.66

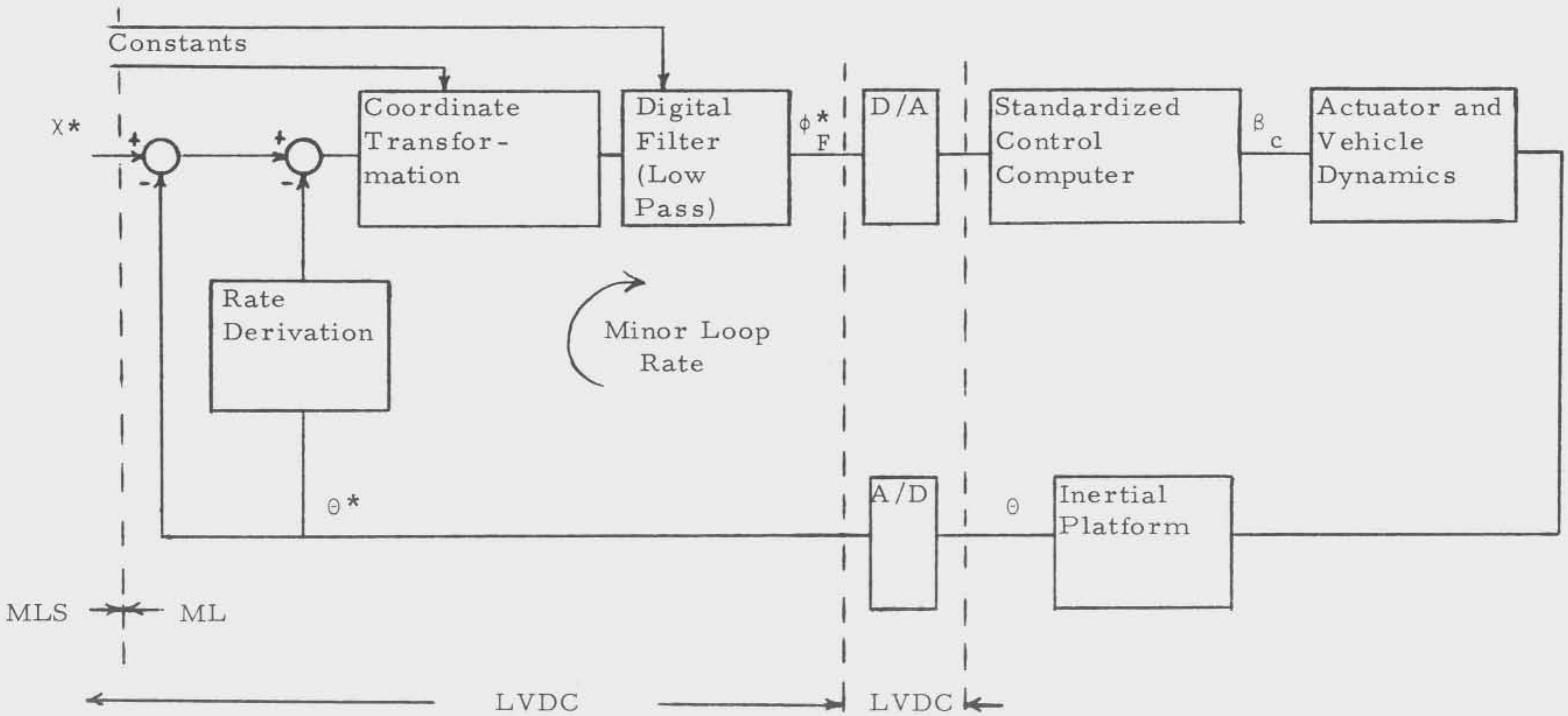


FIGURE 4.3.4.3-1. BLOCK DIAGRAM OF DIGITAL CONTROL SYSTEM EMPLOYING RATE DERIVATION

4.3.4.3 (Continued)

$$F(w) = \frac{2a_1}{T} \quad \frac{w}{w+1} \qquad (1)$$

or in terms of z

$$F(z) = \frac{a_1}{T} \quad \frac{z-1}{z}$$

where a_1 is the gain of the compensator chosen to meet stability requirements.

Digital simulation results for the control system employing explicit rate derivation are presented in Table 4.3.4.3-II and are compared to the previously discussed lead compensator implementation. Time responses to 95% February winds are included in Appendix C.2, Figure C.2-31 through Figure C.2-40. For quantum levels on θ and β_c of 0.00279^o and $.0202^o$ respectively, the rate derivation implementation reduces the peak transient effect in β_c from 0.18^o to 0.60^o in pitch and from 0.6^o to 0.22^o in yaw. However, the pitch plane attitude error response is noticeably altered in form. It has a larger peak magnitude during max q and has a significantly larger value at S-IC stage cutoff. The peak pitch-plane engine deflection is also slightly increased. The response using the back-up quantum level on θ produces similar comparisons with the lead compensator system.

4.3.4.4 Flight Program Requirements

The primary impact of digital control for the INT-20 would be making the necessary flight program modifications to implement the additional tasks assigned to the LVDC. An assessment of that impact can be made by determining the additional computer operations and memory storage locations required. If these requirements can be accommodated by the present flight program philosophy using the aforementioned split minor loop, minimum impact would accrue.

For the systems defined in Section 4.3.4.2 and considering a forward loop lead compensation implementation, the following requirements were determined:

4.3.4.4 (Continued)

S-IC Stage	Inst./Minor Loop	Total Additional Memory Locations*
Pitch	78	96
Yaw	78	96
Roll	78	96
S-IVB Stage		
Pitch	101	120
Yaw	101	120

The above numbers were determined by making a detailed count of the instructions and data words required to implement the various digital filters and then including a 30% factor to account for scaling the fixed point arithmetic and sequencing logic. A typical LVDC instruction requires 82 μ sec; therefore, the total additional computation time for the S-IC stage would be about 19 ms and for the S-IVB stage, 16.5 ms. With the effective minor loop time being 80 ms, these computation times would create no problems. The memory location requirement totaled 528 words. Flight programming estimates indicate that 19% of the GFP (generalized flight program to be used as AS-507 and subsequent 500 vehicles) 32,000 word capacity is at present unused. Therefore, an abundance of storage space is available.

These computer requirement estimates were based on a specific digital control system design for a specific set of design data. As previously discussed, this data is preliminary and a data change that would necessitate a more complex set of stabilization equations would not be an unlikely occurrence. Therefore, Figure 4.3.4.4-1 was prepared which relates how the computer load would be affected by having to consider different order of digital filters. Based on Saturn digital control system design work, the maximum expected order was set at eight. Assuming, as a worse case condition, that all of the pitch-yaw control planes would require eighth filters, and the S-IC roll plane a fourth order, the additional computation time for the S-IC stage would be 38 ms and for the S-IVB stage, 31 ms. The required memory would be 1200 locations. Therefore, even under these extreme conditions, it would still be theoretically possible, using a split minor loop, to assimilate the digital control functions in the flight program.

4.3.4.5 Implementation Considerations

a. Introduction

Implementation planning must give serious attention to the impact on the facility hardware and software capability and the scheduling compatibility due to emphasis on the flight program verification and the verification of Flight Control

*Simplex Memory Locations - There are two instructions per location. These numbers are actually double the basic count as in the LVDC all memory requirements are duplex.

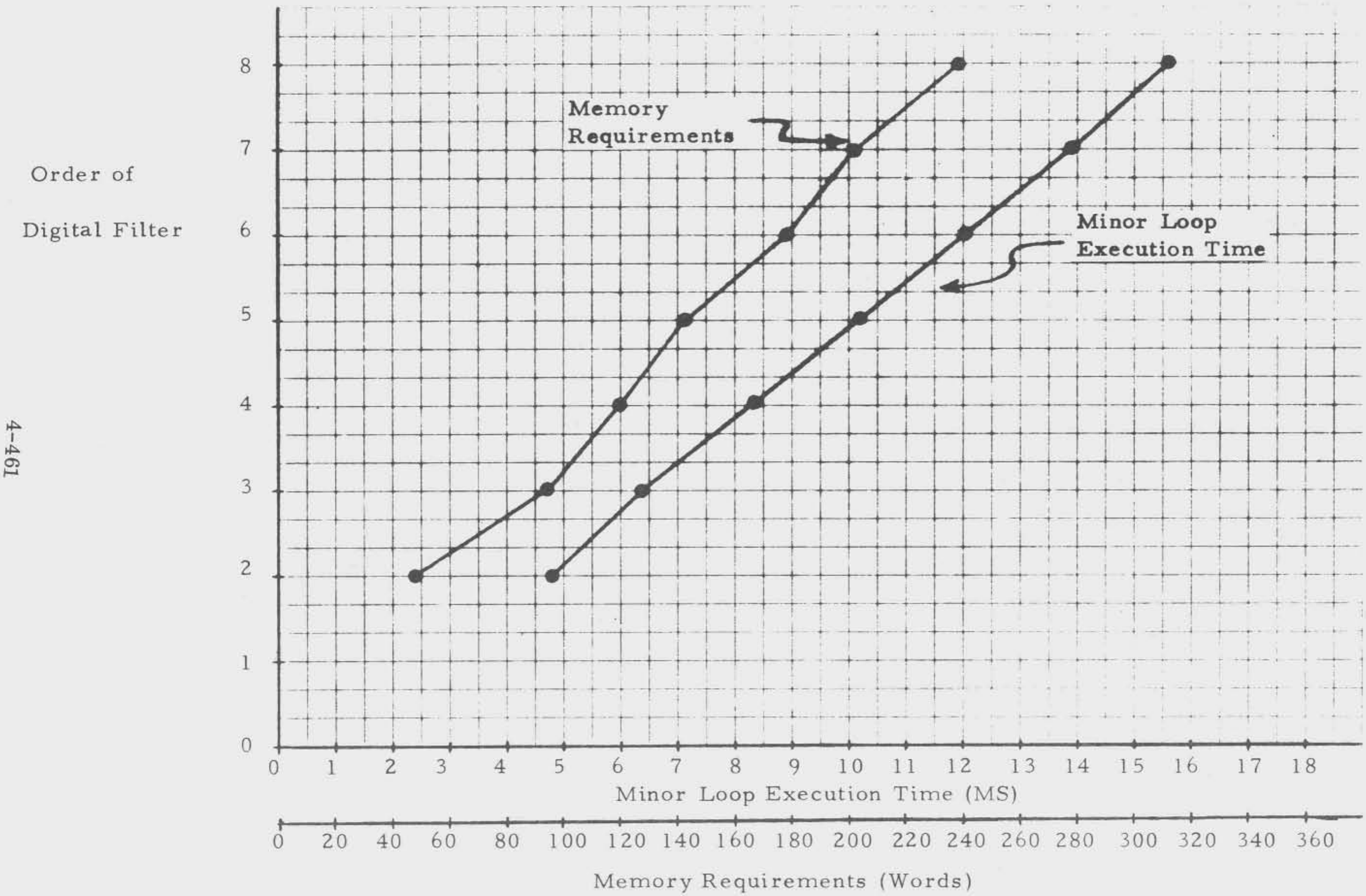


FIGURE 4.3.4.4-1. ADDITIONAL LVDC REQUIREMENTS FOR IMPLEMENTATION OF A DIGITAL FILTER

4.3.4.5 (Continued)

Dynamics. The following discussion treats the schedule and facility impact. The cost impact is beyond the scope of this investigation and is to a degree dependent on the availability of simulation laboratory equipment and rates of Uprated Saturn I, Saturn V, and INT-20 vehicles in the time period of consideration.

b. Summary of Changes to the Flight Control System

A preliminary study has been completed to determine the redesign necessary to reconfigure the present Flight Control Computer (FCC) to the Standard Control Computer (SCC). The two basic requirements are:

A first order RC filter to be included for noise suppression. This filter is standard; i.e., it will not change at staging, is independent of mission/payload, and has no gain change requirement.

The control system containing the SCC will not use the CSP/Rate Gyro outputs for thrust vector control; however, the min-mod APS implementation will use these rate signals.

Therefore, the SCC will receive only three inputs (the filtered $\psi_{Y,R,P}$) as opposed to the six inputs for the FCC during first stage burn. The rate signals would be needed during S-IVB for APS control.

Due to the above requirements, twenty-nine modules will be removed from the FCC for SCC application. These modules, as well as the fifty-eight modules required for the SCC, are tabulated in Table 4.3.4.5-I.

In addition to the modular changes shown in the table, the following will require a complete redesign:

Wiring harness.

Six (out of seven) motherboards.

Three (out of three) switching circuits.

Since twenty-nine modules are to be removed from the FCC to make an SCC, weighted modules will have to be installed to maintain the mechanical integrity of the SCC. Weighted modules are used to prevent a redesign of the packaging of the SCC and/or requalification effort.

All the redundancy and design techniques of the FCC will be retained in the SCC. The input DC Amplifiers of the FCC are retained in the SCC to provide a constant load on the LVDA ladders and to provide the proper DC gain from the LVDA to the 50 ma Servo Amplifiers. The spatial system of the SCC remains the same as the FCC.

TABLE 4.3.4.5-I. MODIFIED STANDARD CONTROL COMPUTER MODULE REQUIREMENTS

Module Name	Present No.	Rqd SCC	Δ
50 ma Servo	8	8	0
DC Amps	20	11	-9
DC Amp Scal bds	3	3	0
Buf DC Amp Sc Bds	2	0	-2
Filters	24	0	-24
Spatial Amps	9	9	0
Spatial Comp	3	3	0
T/M Amps	3	3	0
Spatial Sync	1	1	0
Rate Gyro Filter	1	0	-1
Limiters	3	3	0
Ramp Gen	1	1	0
Servo Comp	2	2	0
Servo Sc Bds	4	2	-2
Swit Cont Bd	1	1	0
Matrix Sw Mod	1	1	0
Telemetry Filter	1	1	0
Noise Filter	0	9	+9
	87	58	- 29

4.3.4.5 (Continued)

The software changes consist of implementing the difference equations within the present minor loop. The split minor loop concept which was discussed previously is recommended for this implementation. The split minor loop will allow the present minor loop interrupt timing to be used, thus avoiding a major reprogramming effort. In addition to the difference equation implementation, program logic to alter the filtered attitude output scale factor and the rate limit on this output would be required.

c. Flight Program Generation

The flight program equations are documented for each mission in the Equation Defining Document. Appropriate equations for the following vehicle functions are included:

Guidance.
Navigation.
Control.
Sequencing.
Tests.
Telemetry Functions.

The impact on the EDD effort would be primarily limited to two of the above functions: (1) Control and (2) Sequencing. The associated software changes noted in the previous section would be specifically defined for the digital control implementation. In addition, the switch selector functions associated with the hardware switching within the FCC would be eliminated from the flight sequence. Also, this effort would include a definition of the required logic to correctly change the filter coefficients and gains throughout boost. The impact in these two areas would be most significant for the first vehicle launched with a digital control system.

d. Flight Program Checkout

The flight program checkout is presently accomplished on a simulator using a System/360, Model 44 with an LVDC/LVDA and the required interface equipment. The main purpose of this facility is to debug the flight program using a six-degree of freedom vehicle simulator. The impact on the present operation would be very minimal. It would be primarily due to a more complex minor loop which could possibly increase program checkout time. In addition, the six-degree of freedom vehicle simulator would require a modification of its present control law.

4.3.4.5 (Continued)

e. Flight Program Verification

The flight program verification uses two different six degree of freedom simulators: (1) a System/360 digital simulator and (2) an all digital simulator which includes an LVDC/LVDA operating in real time. Perturbation cases are selected such that the flight program is fully exercised and all logic and program constants verified. Digital control implementation would require minor modifications to both of the simulator's control law. In addition, cases to verify the digital control logic, gains and filter coefficients would need to be included as part of the present flight program verification plan.

f. FCC Verification Facility

There exist two FCC dynamic analysis checkout facilities. The primary equipments in each area are a Control Computer Console (CCC) and a Milgo 4100 Analog Computer with associated peripheral equipment. The CCC normally operates in conjunction with the following Saturn flight-type equipment:

Control/EDS Rate Gyros.
Control Signal Processor.
Control Relay Package.
Control Accelerometers (Uprated Saturn I Booster)

Each FCC undergoes an extensive series of test to verify its flight worthiness. These tests include:

Linearity and Mixing.
Static Gain Test.
Servo Amplifier Null and Noise Test.
28 Volt Power Supply Variation.
Switch Point Utilization Test.
Engine Cant Test.
Open Loop Channel Frequency Responses for Filters.
Nyquist Frequency Response (including vehicle dynamics simulated on Milgo 4100).
Closed Loop Transient Responses.
Wind Responses.
Cross Coupling and Common Mode Test.
Spacecraft Control Test.

4.3.4.5 (Continued)

From the above sequence of tests, it is seen that each FCC is treated as a development piece of hardware. After the development of the first SCC, it is foreseen that this phase of testing could be eliminated. The rationale is that the mission variant portion of the control computer has been removed and the unit is dynamically invariant because of the removal of the stability filters. Thus for the SCC, an appropriate acceptance test would replace the present FCC checkout procedure significantly reducing the cost and freeing the two Milgo 4100 Analog Computers for other applications.

g. Guidance and Control Evaluation Facility

The present test setup of the G&C evaluation facility is shown in Figure 4.3.4.5-1. The equipment and objectives of this simulation would not change with digital control implementation. A modification to this simulation however would be necessary. The analog filters on the Milgo 4100 would be deleted (a very minor modification in terms of total impact). Since the minor loop timing will remain the same, no further modifications would be necessary other than scaling changes for the filtered attitude error.

h. Digital Filter Verification Facility

The flight program verification as discussed in section e will determine if the digital control system logic and constants are correctly implemented. Verification to this degree would be synonymous to checking static gains and component values for the filters in the FCC. In line with the present philosophy of dynamic testing, it has been determined that the w-plane frequency response (open loop Bode and Nyquist) can be verified on a hybrid simulation including the flight program with the digital filter implementation and vehicle dynamics on the Milgo 4100. Shown in Figure 4.3.4.5-2 is the test setup illustrating the necessary equipment to accomplish this dynamic verification. This verification would be performed on presently existing equipment of the all digital simulation laboratory flight program verification (section e).

i. Flight Verification Experiment

Digital control offers the flexibility needed by the Saturn Derivative programs. It will allow the standardization of the control computer, thus reducing the cost and the required lead time to build. It is an accepted means of controlling aerospace vehicles, e.g., Minuteman and both the CSM and LM constitute examples of operational digital control systems.

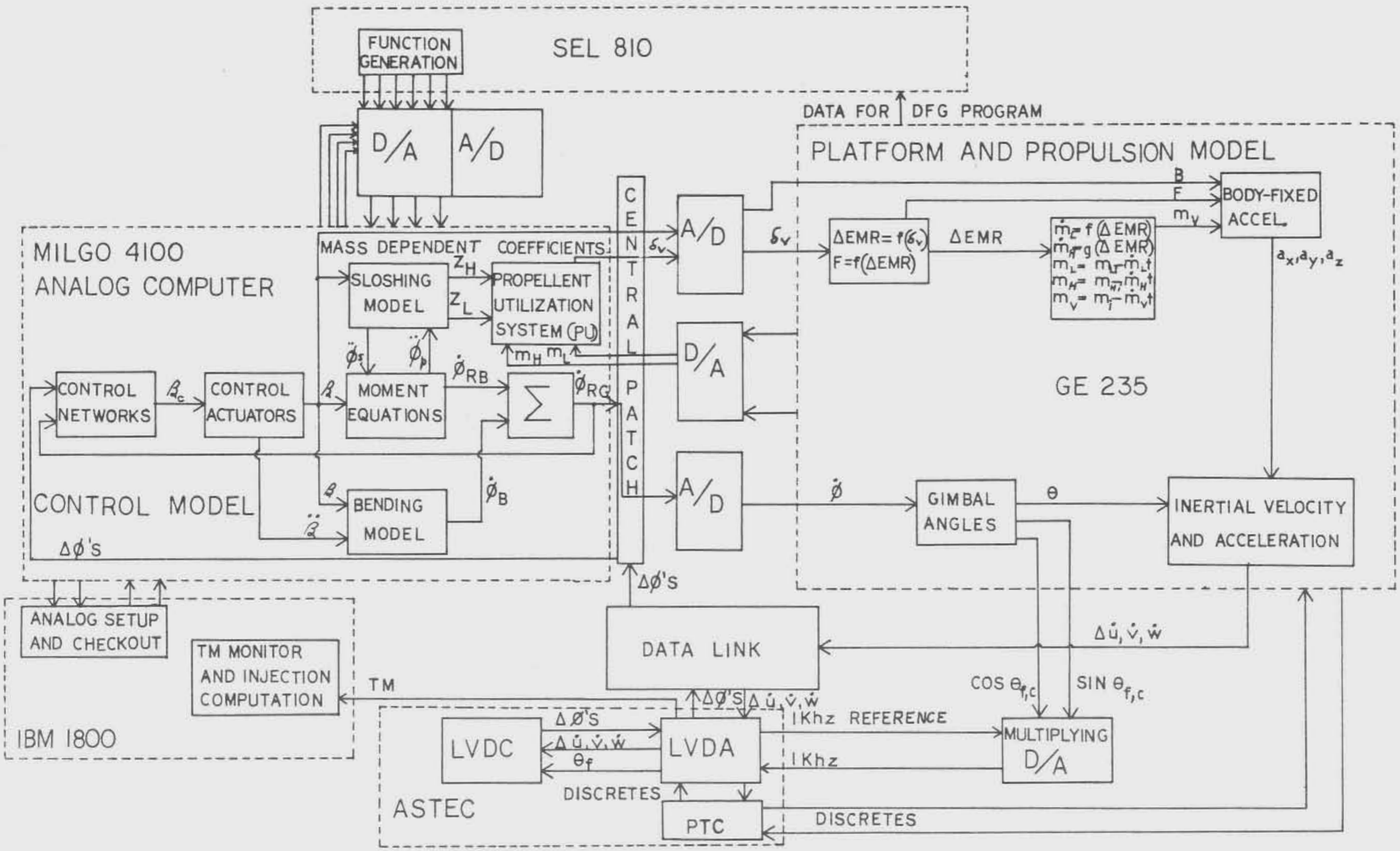


FIGURE 4.3.4.5-1. GUIDANCE AND CONTROL EVALUATION FACILITY

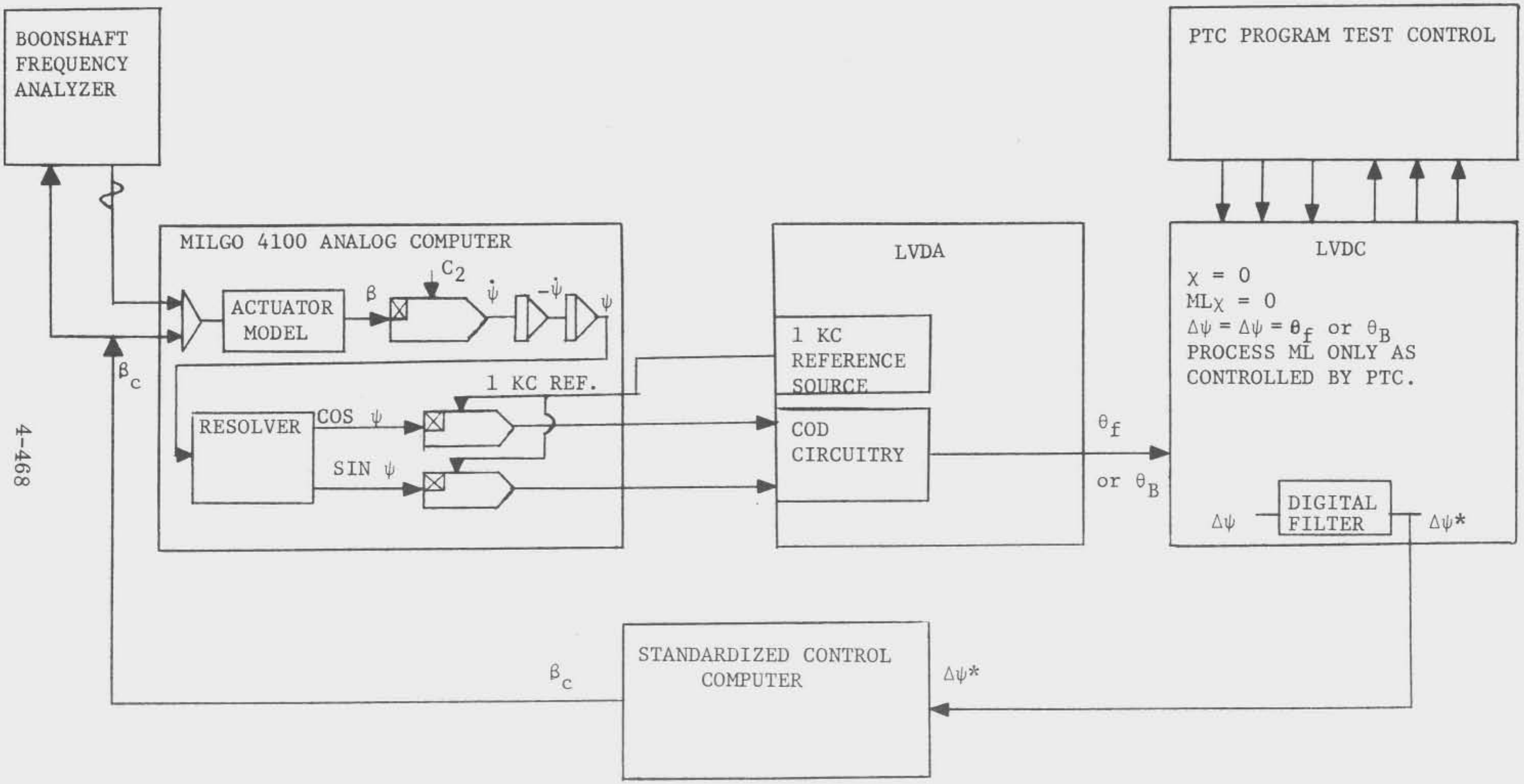


FIGURE 4.3.4.5-2. DIGITAL CONTROL VERIFICATION IN FREQUENCY DOMAIN

4.3.4.5 (Continued)

It is felt that the most economical approach to actual implementation of a digital control system is utilization of a fully operational system on the initial or break-in vehicle. If a subjective decision were made to verify analytical results with flight data, a "piggy-back" experiment could be flown on the first INT-20 vehicle.

The feasibility of this experiment has been examined. The required hardware and software changes were kept to a minimum while attempting to obtain a meaningful experiment. In brief, the experiment consists of implementing the digital control system in parallel with the present FCC. In this manner, the response of the digital control system can be compared to the response of the FCC in a flight environment with actual flight inputs.

The software changes necessary for this experiment would include those normally required to implement digital control, while maintaining the present FCC flight program functions. The hardware changes would be more numerous than for operational implementation. Two additional ladders would be made operational with minimal cost. One ladder could be used for conversion of the filtered ψ_p or ψ_y on a time shared basis, with ψ_r on a continuous basis on the second ladder. With these two ladder outputs, comparisons could be made between the four pitch mag-amps of the FCC to the four pitch mag-amps of the SCC. The SCC would be working into dummy loads (open-loop from the standpoint of affecting engine deflection). Using PIO codes to switch from pitch to yaw mag-amps, comparisons could be made for the four pitch mag-amps during one time segment and the four yaw mag-amps during another time segment. A special wiring harness would be required for this "piggy-back" operation. In addition, a telemetry Measuring Rack (50Z66650-1) with Channel Selector (50Z12361 or 2) must be added to properly monitor the additional signals.

j. Schedule Impact

It was determined that there would be no schedule impact on the normal IU development cycle if the INT-20 utilizes digital control. However, as shown in Figure 4.3.4.5-3, the filter release and the start of SCC production dates can be delayed. The filter release date is moved to the right since this data is no longer required in order to produce the SCC. The new filter release date would precede initiation of flight program verification by two weeks. SCC production is delayed four weeks due to the checkout requirements being decreased.

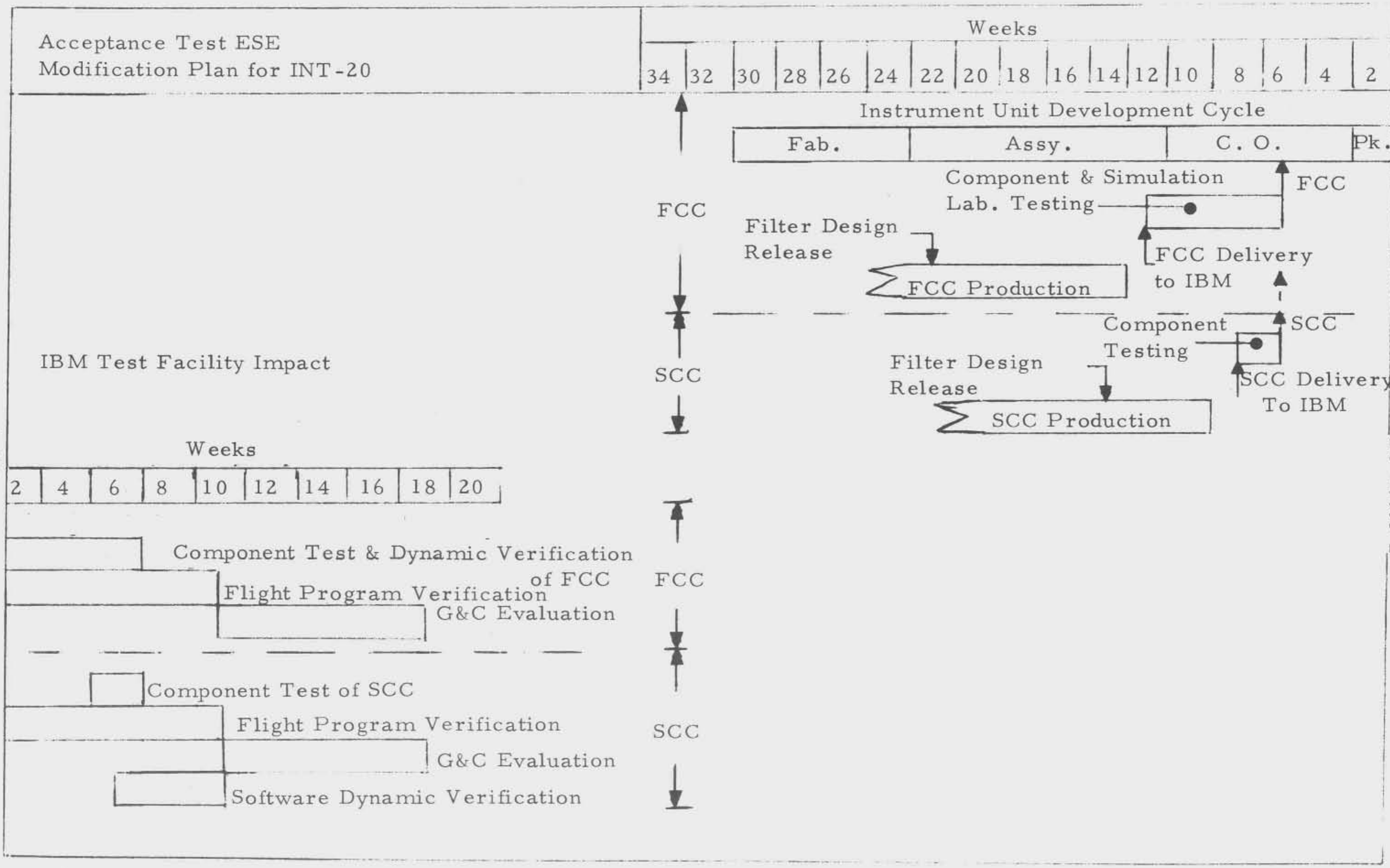


FIGURE 4.3.4.5-3. COMPARISON OF FCC/SCC TIME SCHEDULES FOR INT-20 VEHICLE

4.3.4.5 (Continued)

Figure 4.3.4.5-3 also indicates that the total time that the IBM Huntsville test facility would be involved per vehicle would not change. Testing emphasis however would shift from FCC hardware testing to a more involved Flight Program checkout procedure.

4.3.4.6 Summary

Frozen point w-plane Nyquist studies have demonstrated the feasibility of obtaining stability margins with the digital INT-20 control system which are comparable to those of the Saturn V. The order of the required digital filters designed for both stages is relatively low and incorporation of these equations into the flight program is easily accomplished using a split minor loop.

The results obtained in the simulation analysis lead to the recommendation that the lead type compensator difference equations be implemented in the feed-forward path. This recommendation is based on the following facts:

The engine transients resulting from guidance non-linearities are acceptable (post flight data on prior Saturn V's have shown engine transients of the same order of magnitude).

While explicit rate deviation eliminates the aforementioned engine transients, the vehicle attitude response has deteriorated. Larger attitude errors exist both during Max Q and at S-IC Stage cutoff.

Additionally, it is recommended that a more detailed investigation be made of rescaling the LVDA output so as to effect a finer quantization level on the engine command without severely limiting engine deflection rate. The response studies indicate a definite performance advantage using the finer level.

The impact of implementing digital control on the INT-20 may be considered in two categories - break-in and recurring. The important items to be considered in the break-in impact are:

Evolution of FCC to SCC. As this conversion involves modification of current flight operational hardware, this item must be considered as the most significant impact.

Altering the flight program. This would include digital filter implementation, gain program and the required timing logic.

Modifying checkout procedures and vehicle simulations. The primary effort in this area would be the establishment of a flight program dynamic verification capability using existing simulation laboratory equipment.

4.3.4.6 (Continued)

The recurring impact may be summarized as follows:

There would be no impact on the IU development cycle.

SSC production and control system design could be initiated at a later date.

Design changes would require software constant changes only. There would be no SSC refabrication and retest cycles required.

Considering all the aforementioned items, there is nothing that would prohibit digital control system implementation.

4.3.5 INT-20/J-2S

The payload performance characteristics of the INT-20 vehicle were generated using a J-2S engine in the S-IVB stage. Three missions were investigated for the J-2S application and are:

a. Direct, coplanar ascent to low Earth circular orbits (the range of orbit altitude studied being from 100 to 300 NM)

b. Direct, coplanar launch through a 100 NM circular orbit (No coast time in the orbit is assumed) to various energy levels.

c. Direct, coplanar launch to a 100 NM parking orbit followed by an S-IVB burn-coast-burn maneuver into a synchronous orbit.

The results of this study are presented in Figures 4.3.5-1 through 4.3.5-3.

With the exception of the assumed vehicle weights (as presented in Table 4.3.5-I), the assumptions used in generating the data presented in this coordination sheet were basically the same as those employed in generating the INT-20 study baseline.

a. Launch from the AMR with a launch azimuth of 90° and a liftoff thrust to weight ratio of 1.25.

b. No mixture ratio shifts were employed in either the S-IC or S-IVB (a mixture ratio of 5:1 was assumed for the S-IVB).

c. A 3.8 second coast was flown between S-IC final engine cutoff and S-IVB ignition.

d. Maximum longitudinal acceleration was limited to 4.68 g's by shutting down 2 F-1 engines at t = 146 seconds and then staging ballast at final F-1 engine cutoff with the S-IC.

e. Aerodynamic characteristics assume an MSFC double angle nose cone.

f. For all missions, flight performance reserves of 3/4 percent are accounted for in the S-IVB. For the synchronous orbit and high energy missions, a launch window reserve of 60 m/sec is also accounted for in the S-IVB.

4.3.5 (Continued)

The inert weights employed in this study (as shown in Table 4.3.5-I) are based on the stage weights definition presented in Reference 4.3.5-1. These weights differ from those used to generate the baseline INT-20; for this reason, the J-2S performance data presented in this coord sheet is comparable only to the J-2 performance data which is also presented in this section.

The results of this study show that application of the J-2S engine to the INT-20 configuration results in a payload gain of from 2,000 to 10,000 lbs depending upon the mission.

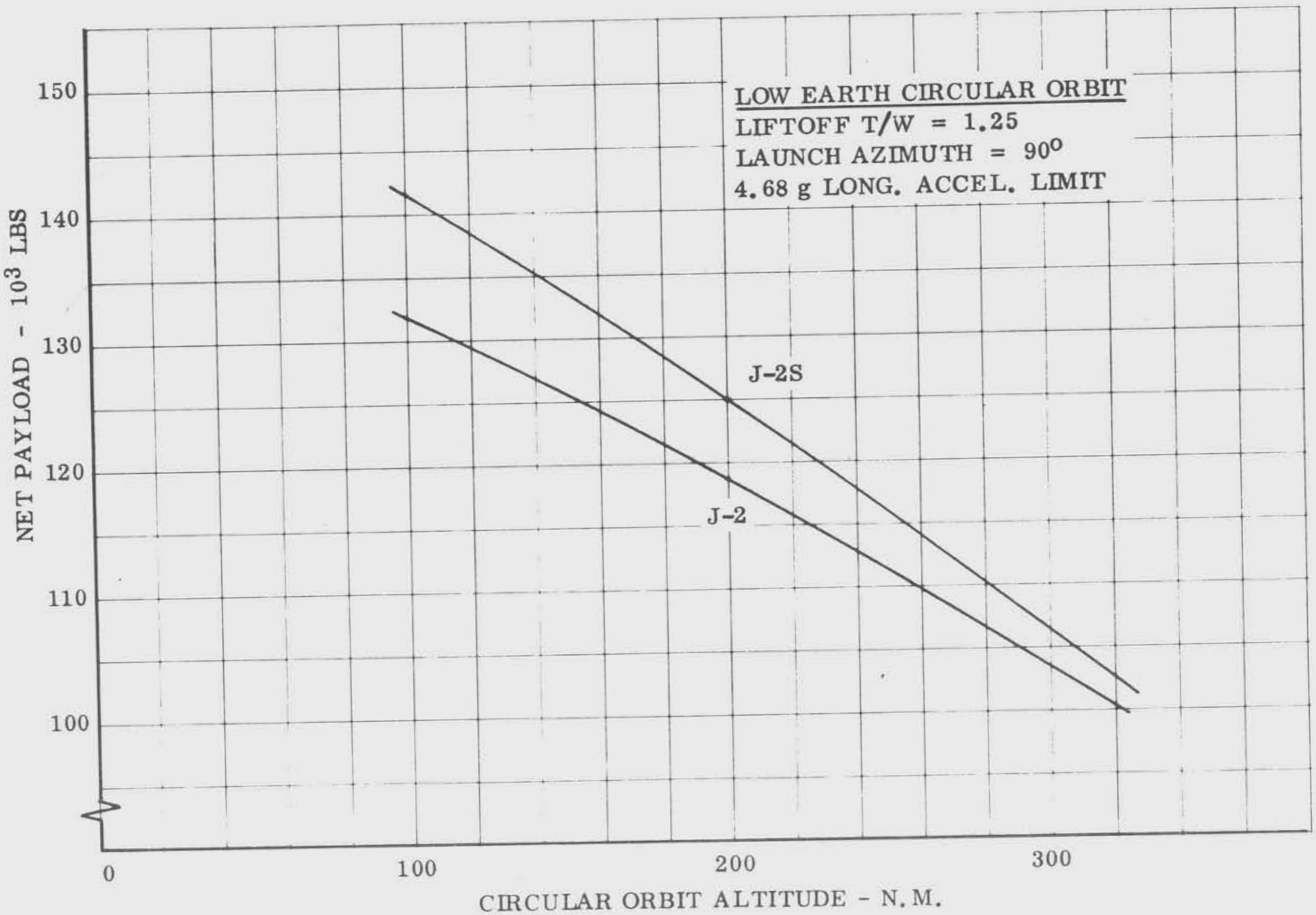


FIGURE 4.3.5-1 INT-20/J-2S LOW EARTH CIRCULAR ORBITS

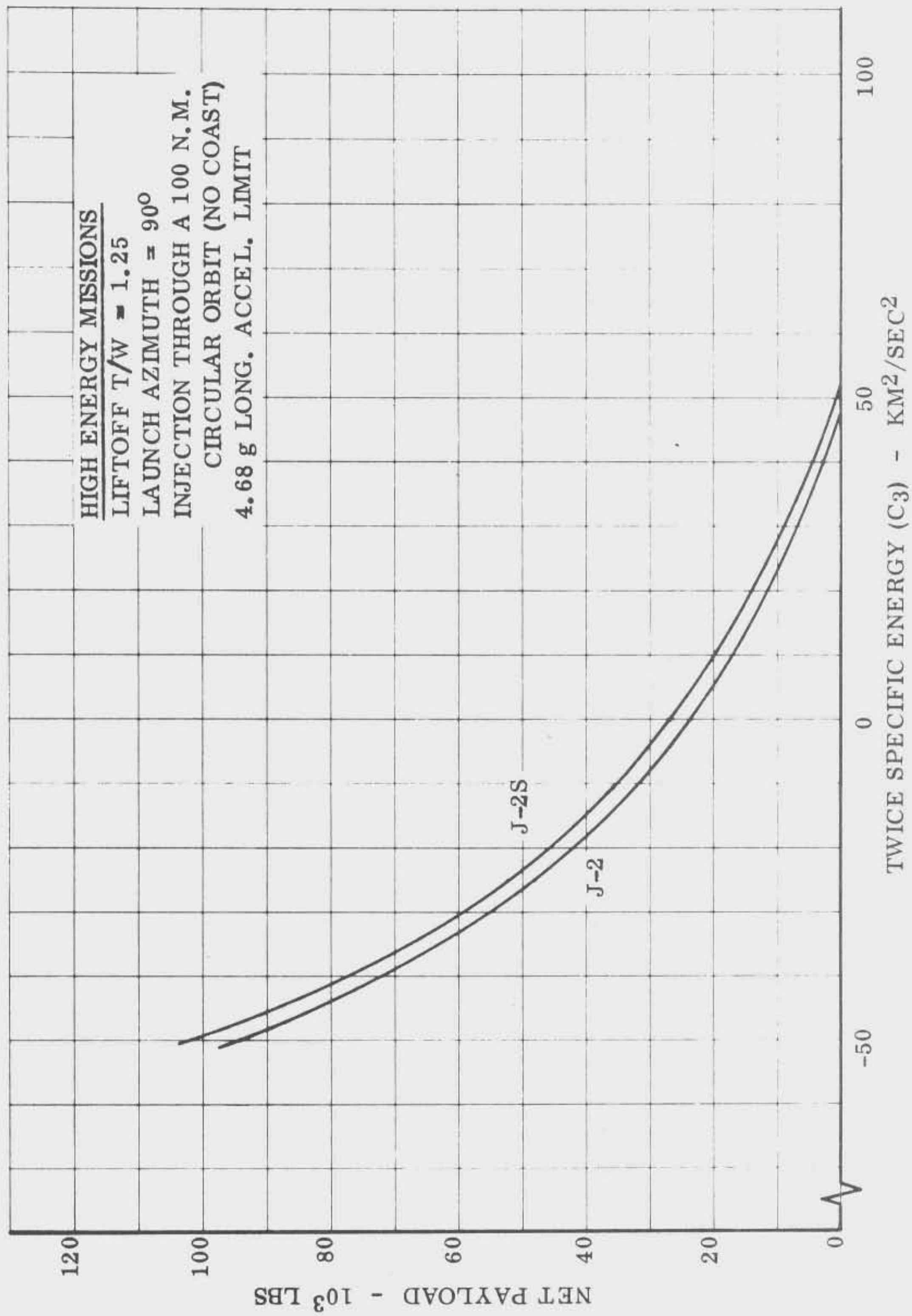


FIGURE 4.3.5-2 INT-20/BIG G HIGH ENERGY MISSIONS

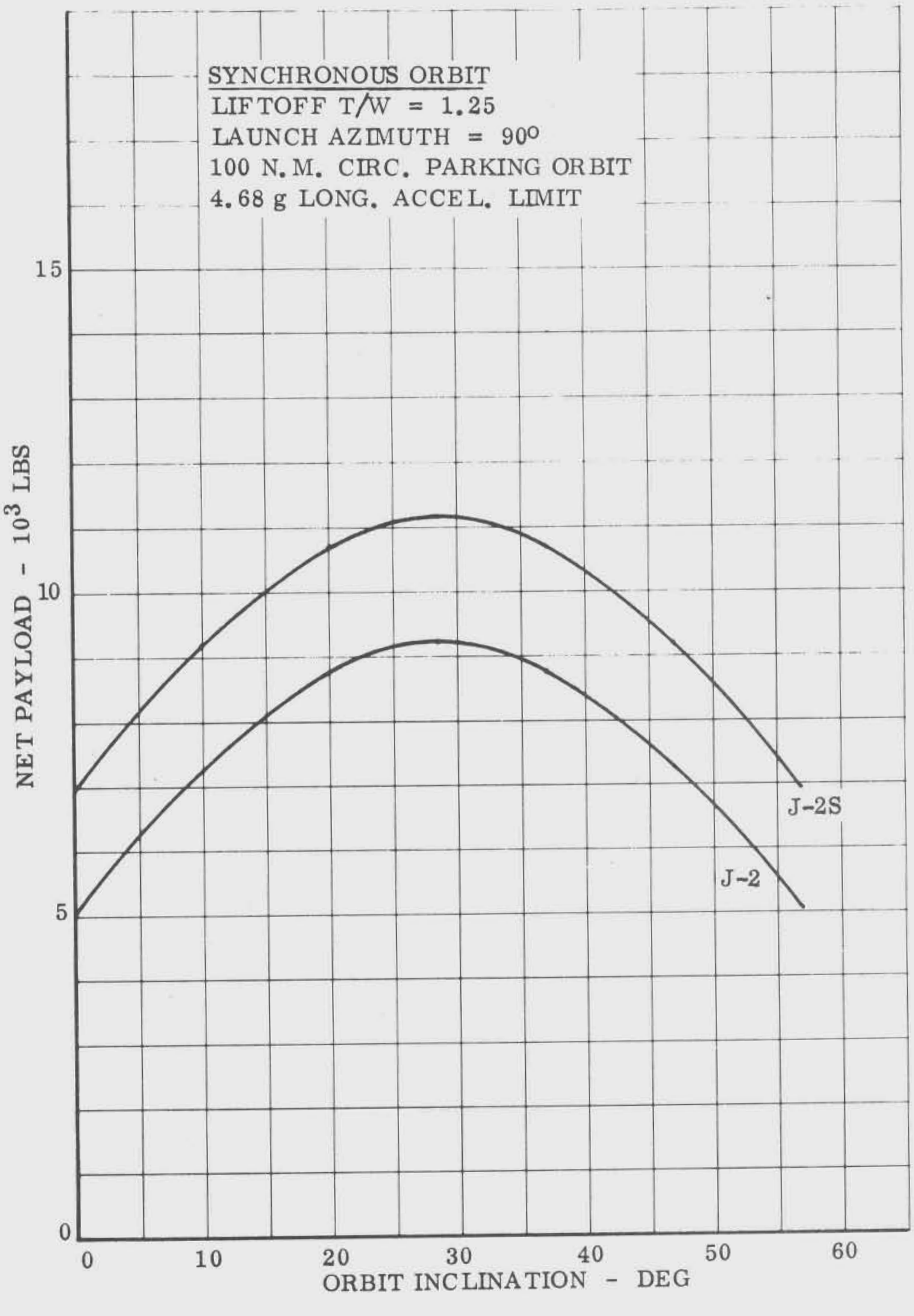


FIGURE 4.3.5-3 INT-20/BIG SYNCHRONOUS ORBITS

TABLE 4.3.5-I

SATURN V DERIVATIVES (NAS8-30506)

J-2S ENGINE APPLICATION - VEHICLE DEFINITION

		LEO & HIGH ENERGY MISSIONS		SYNCHRONOUS ORBIT MISSIONS	
		J-2	J-2S	J-2	J-2S
Liftoff Weight	lbs	4,870,400	4,870,400	4,870,400	4,870,400
Sea Level Thrust	lbs	6,088,000	6,088,000	6,088,000	6,088,000
Sea Level Specific Impulse	sec	263.58	263.58	263.58	263.58
Propellant Consumed	lbs	4,122,325	4,122,325	4,122,325	4,122,325
Stage Weight at Separation	lbs	339,364	339,199	339,364	339,199
Vacuum Thrust	lbs	205,000	237,500	205,000	237,500
Vacuum Specific Impulse	sec	426	434.6	426	434.6
Propellant Capacity	lbs	230,000	230,000	230,000	230,000
Weight Loss in Parking Orbit	lbs			2,517	3,804
Weight Loss in Transfer Ellipse	lbs			1,692	2,946
Stage Weight at Separation	lbs	27,546	27,294	29,292	28,197
Astrionics Equipment	lbs	4,303	4,303	4,675	4,675

4.4 MINIMUM CHANGE S-IC

The baseline S-IC stage configuration for INT-20 as defined in Section 4.2.2.1 is based on the INT-20 baseline trajectory which limits the S-IC acceleration to 4.68 g at both two and four engine cutoff. It is necessary to revise the lower fuel bulkhead base gores, as defined in Section 4.2.2.1.a.4(c), to provide the structural capability required to maintain a 1.4 factor of safety for this baseline trajectory. The existing Sat V lower fuel bulkhead design could be used, while maintaining a 1.4 factor of safety, for INT-20 by revising the trajectory such that the acceleration during the critical period is reduced to an accepted level. This is accomplished for the second iteration trajectory (Figure A-23 of Appendix A) by cutting off the first two engines at 126 seconds, thus reducing the critical acceleration from 4.68 g to 3.68 g. This revised trajectory is the same as recommended for the retrofit S-IC (Section 6.1.1.2).

The impact of the S-IC configuration without the revised gores for INT-20 would be to reduce the delta INT-20 baseline weight by 300 pounds.

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SECTION 5
PHASE III DEVELOPMENT PROGRAM PLAN

5.0 GENERAL

The Development Program Plan (Resources Plan) presents the essential elements and milestones to implement an INT-20 (S-IC/S-IVB) vehicle program.

Programs studies are:

2 INT-20s with 2 Saturn Vs per year
4 INT-20s with 2 Saturn Vs per year
3 INT-20s with 3 Saturn Vs per year
2 INT-20s per year with no Saturn Vs
4 INT-20s per year with no Saturn Vs

A "Resource Summary" sheet for each program is shown on Figures 5.0-1 through 5.0-8. Each summary sheet shows cost and schedule information as follows:

a. Funding distribution curves.

b. Total cost for INT-20s and Saturn Vs (includes hardware, support and launch) for a five year program.

c. INT-20 average unit cost for "incremental cost method" and "distributed cost Method" for programs with both Saturn Vs and INT-20s.

   The INT-20 "incremental cost method" assumes an existing Saturn V program and determines the increment of cost to add INT-20s to the Saturn V program.

   The INT-20 "distributed cost method" assumes a Saturn V/INT-20 mixed program and distributes each element of cost proportionately between the INT-20 and the Saturn V.

   Note that the average unit cost of an INT-20 is different by the two methods but for each method the total program cost is the same.

d. INT-20 Development Cost.

e. Authority to proceed date.

f. First INT-20 launch date.

g. The Saturn V delivery schedule.

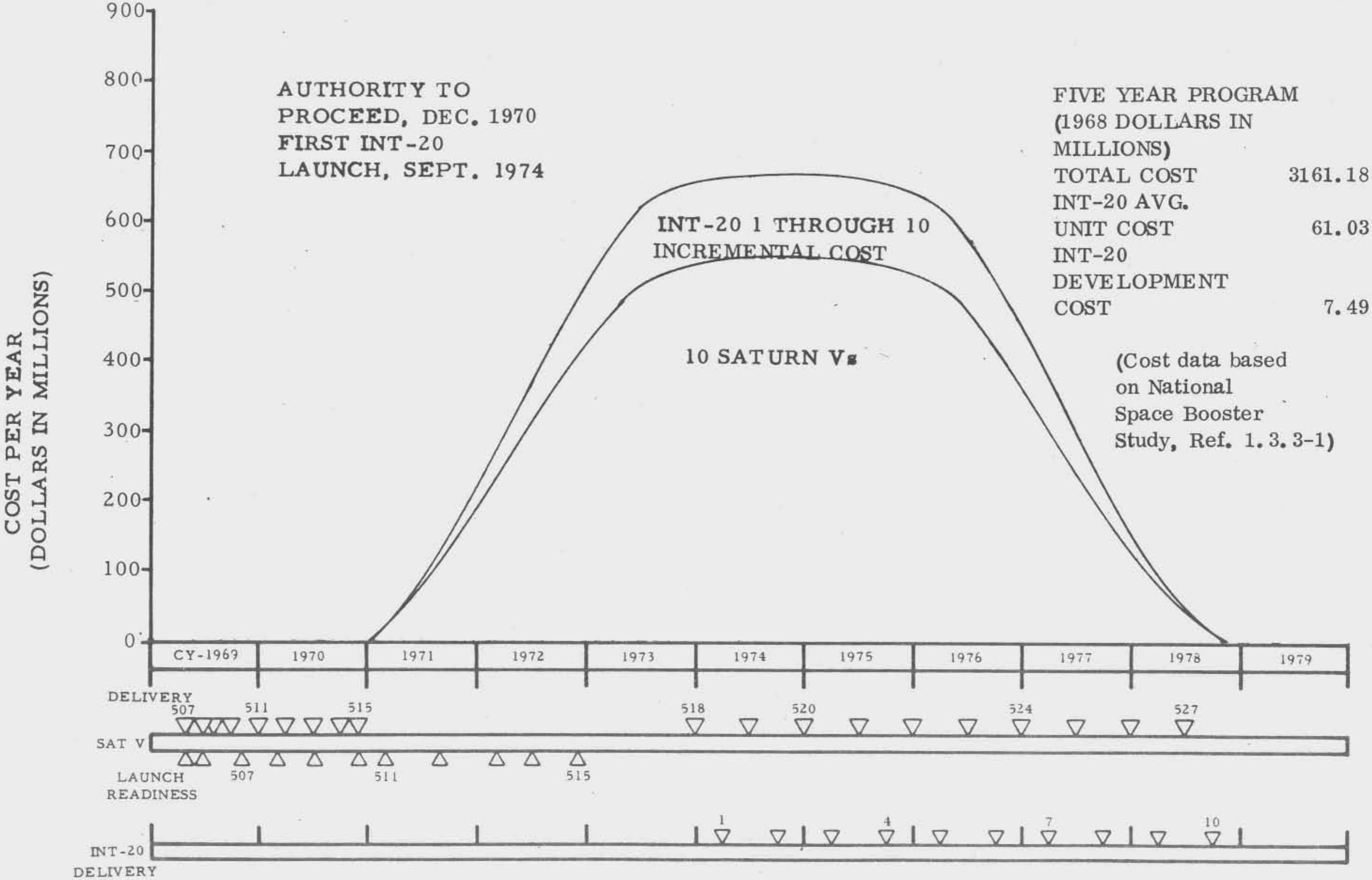

FIGURE 5.0-1 RESOURCE SUMMARY, 2 INT-20s WITH 2 SATURN Vs PER YEAR, NO STATIC FIRING (INCREMENTAL COST METHOD)

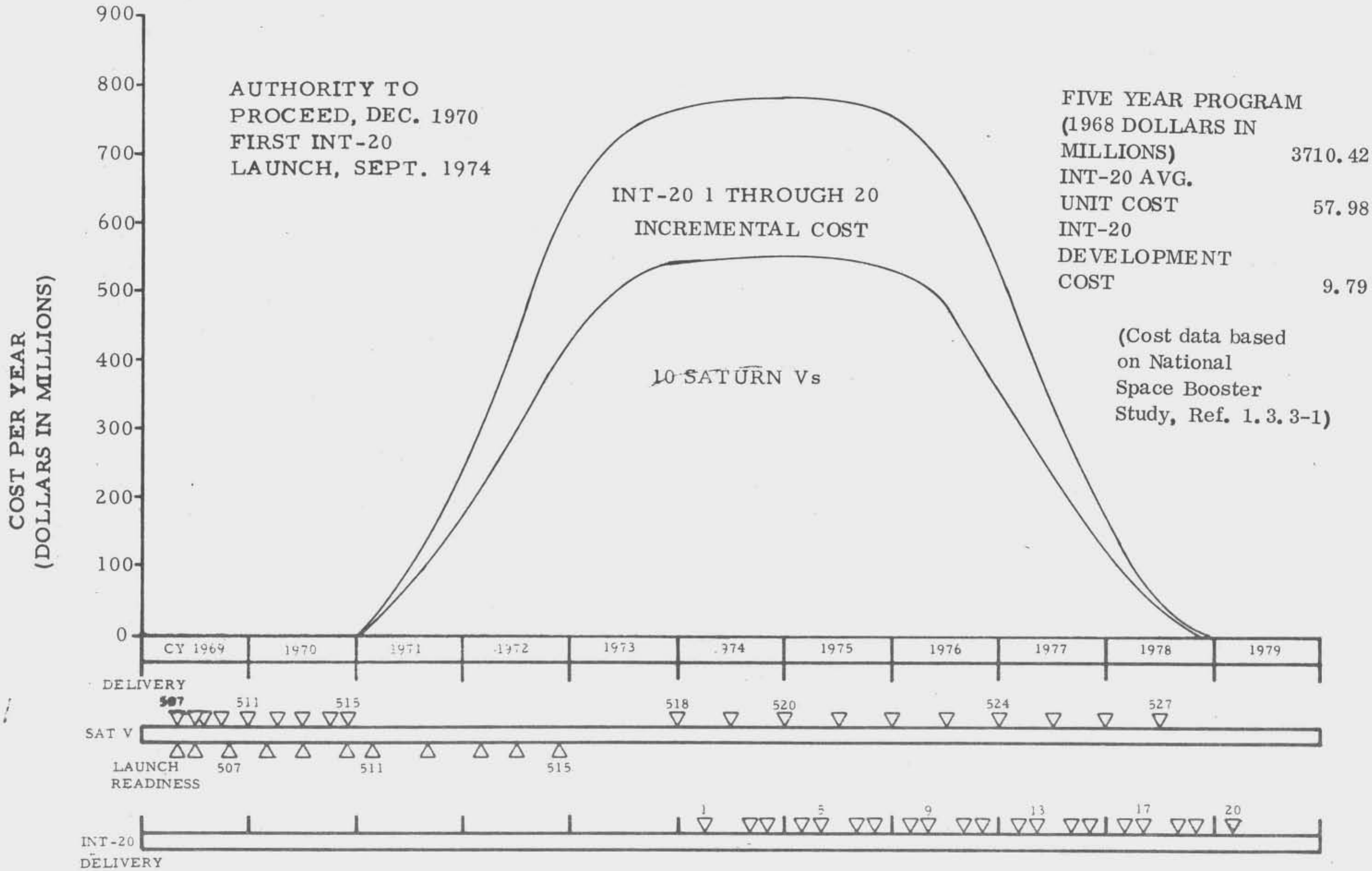

FIGURE 5.0-2 RESOURCE SUMMARY, 4 INT-20s WITH 2 SATURN Vs PER YEAR, NO STATIC FIRING (INCREMENTAL COST METHOD)

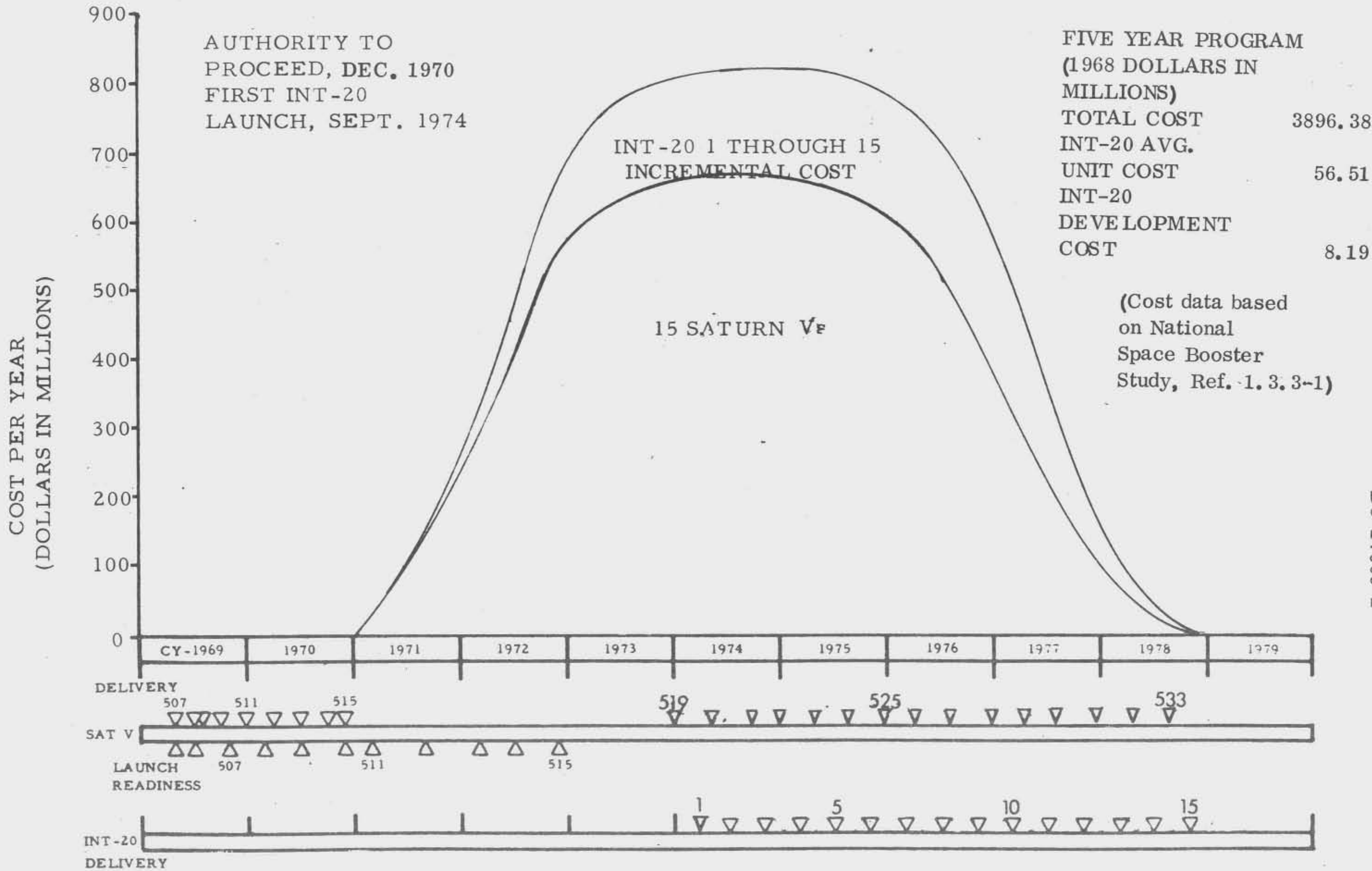

FIGURE 5.0-3 RESOURCE SUMMARY, 3 INT-20s WITH 3 SATURN Vs PER YEAR, NO STATIC FIRING (INCREMENTAL COST METHOD)

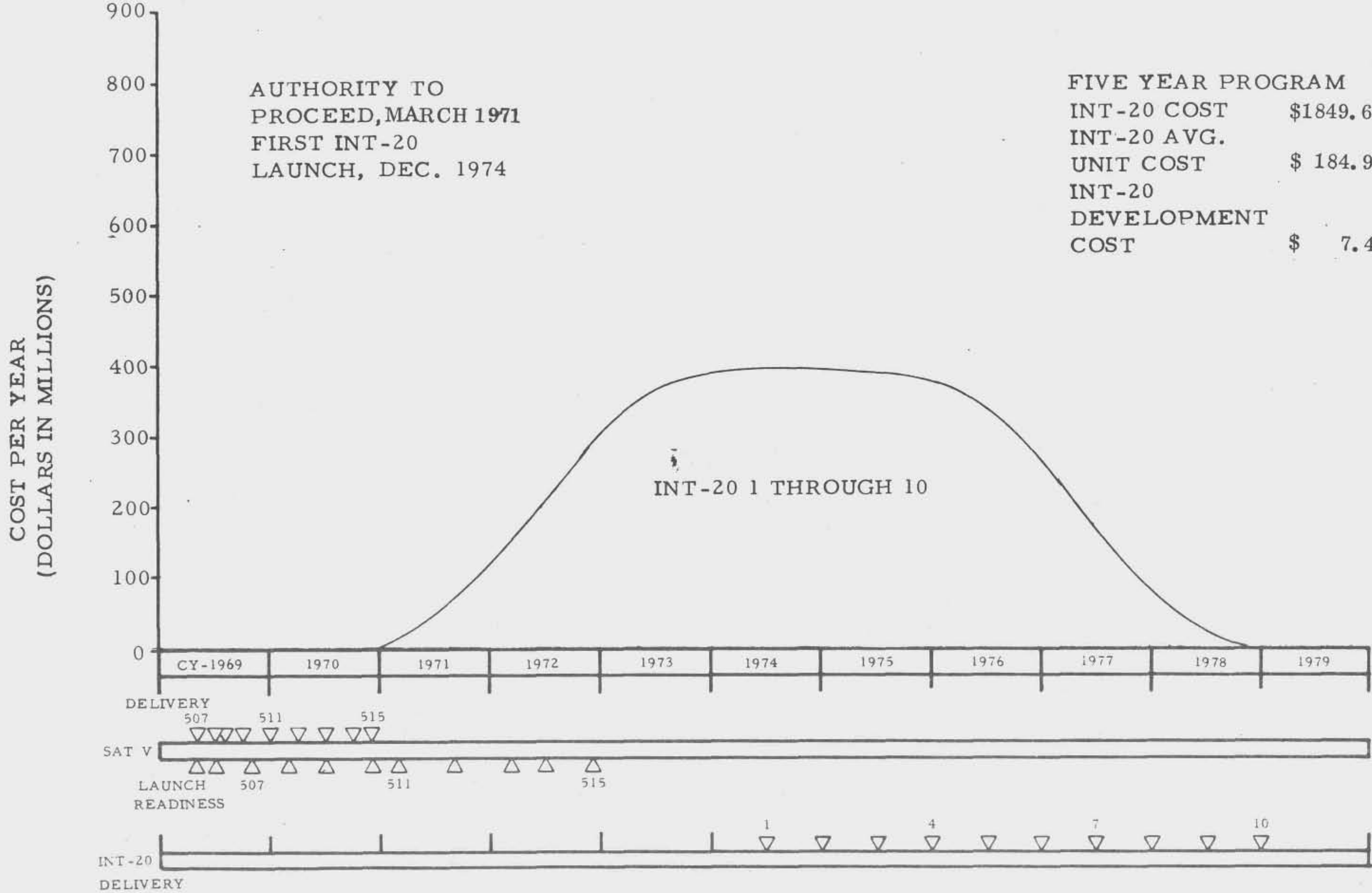

FIGURE 5.0-4 RESOURCE SUMMARY, 2 INT-20s PER YEAR WITH NO SATURN Vs NO STATIC FIRING

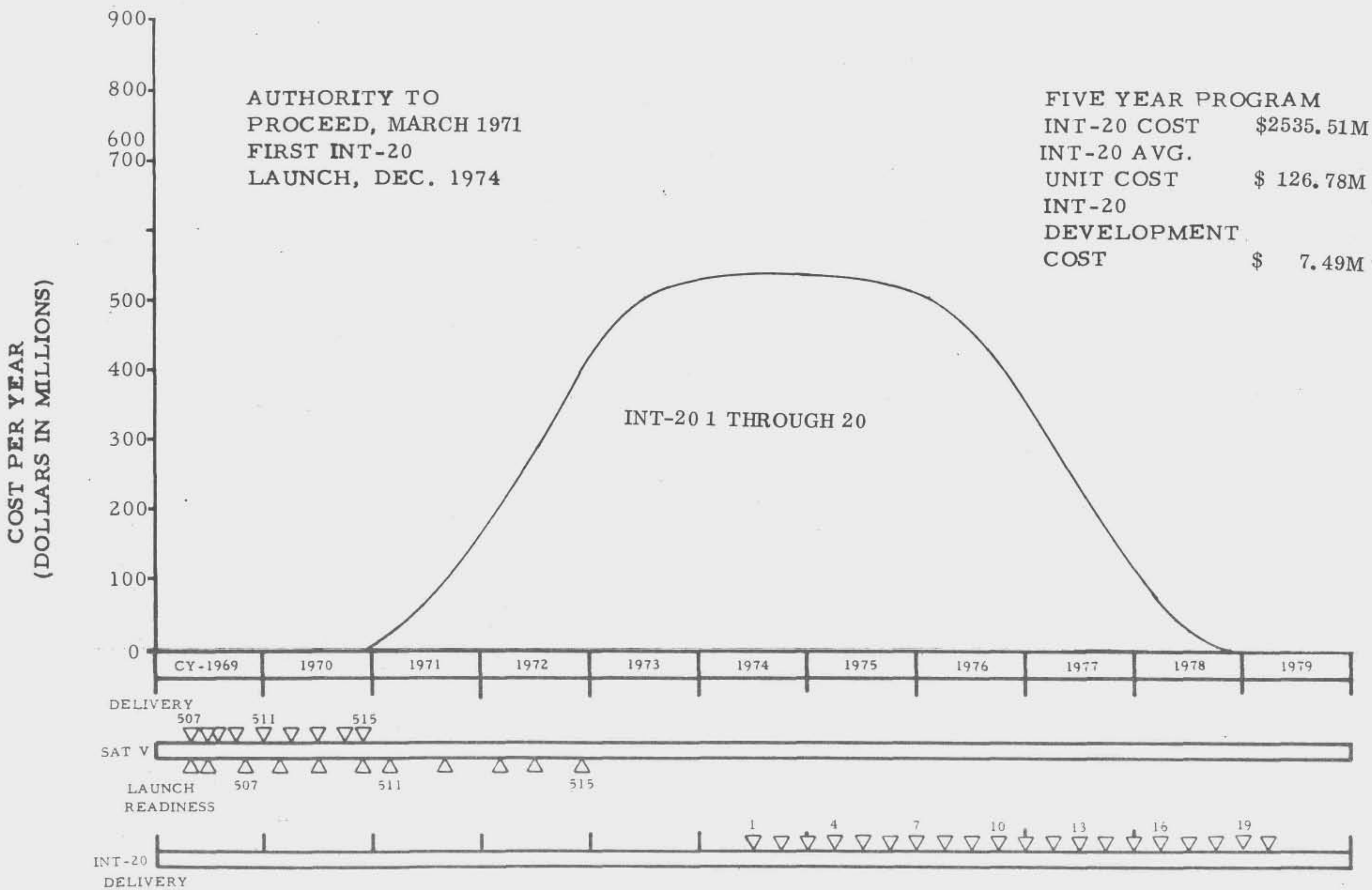

FIGURE 5.0-5 RESOURCE SUMMARY, 4 INT-20s PER YEAR WITH NO SATURN Vs, NO STATIC FIRING

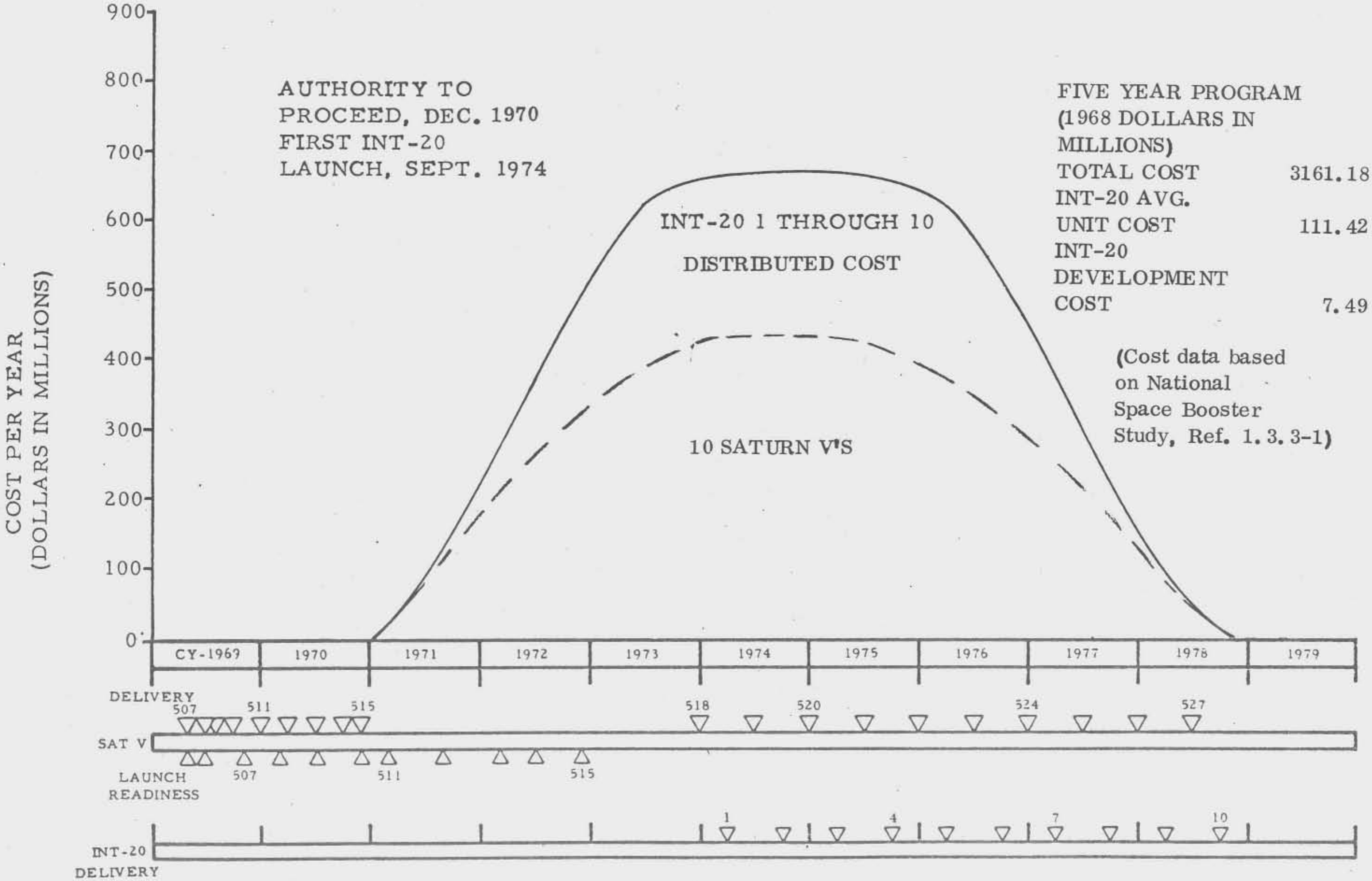

FIGURE 5.0-6 RESOURCE SUMMARY, 2 INT-20s WITH 2 SATURN Vs PER YEAR, NO STATIC FIRING (DISTRIBUTED COST METHOD)

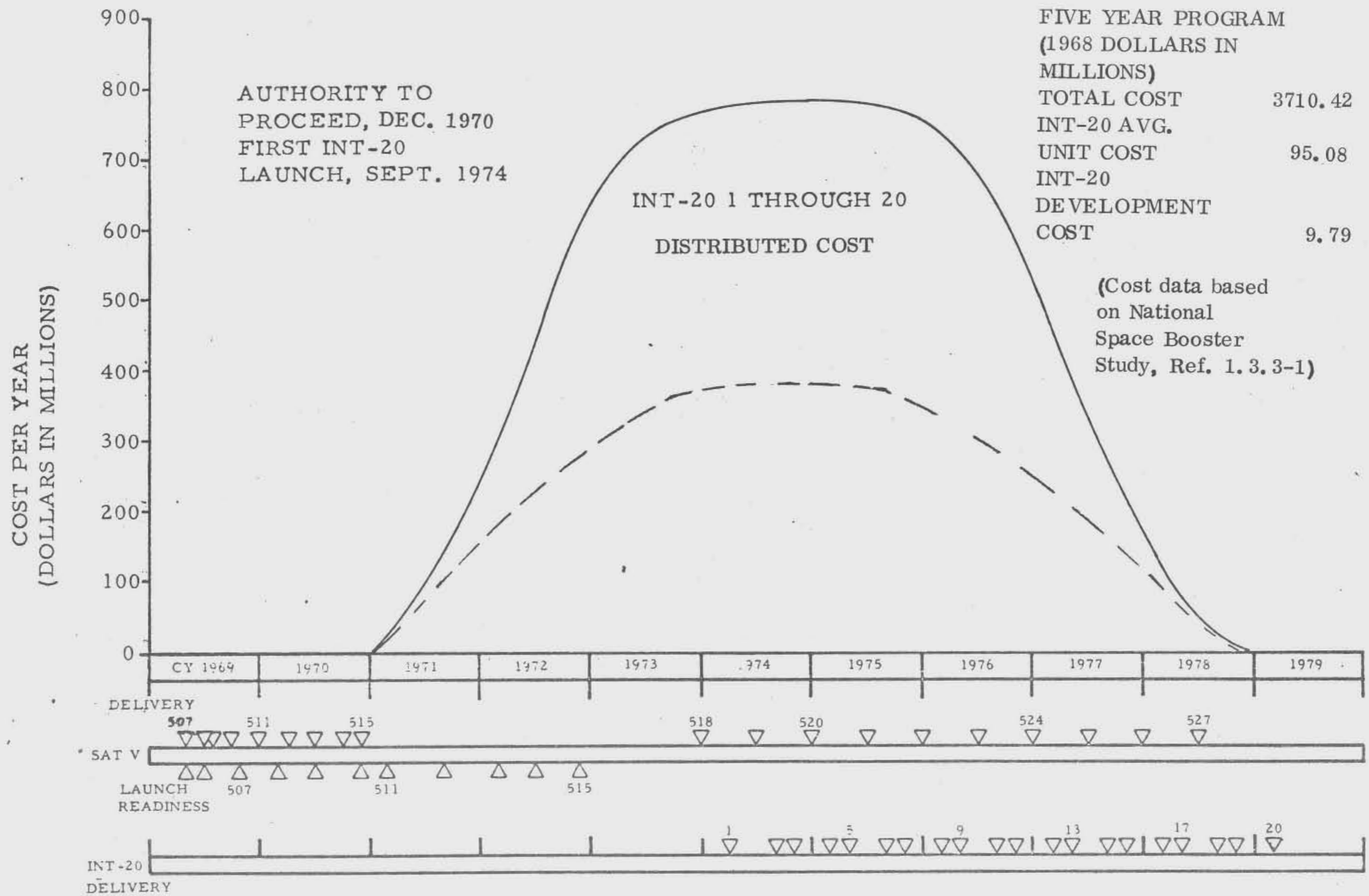

FIGURE 5.0-7 RESOURCE SUMMARY, 4 INT-20s WITH 2 SATURN Vs PER YEAR, NO STATIC FIRING (DISTRIBUTED COST METHOD)

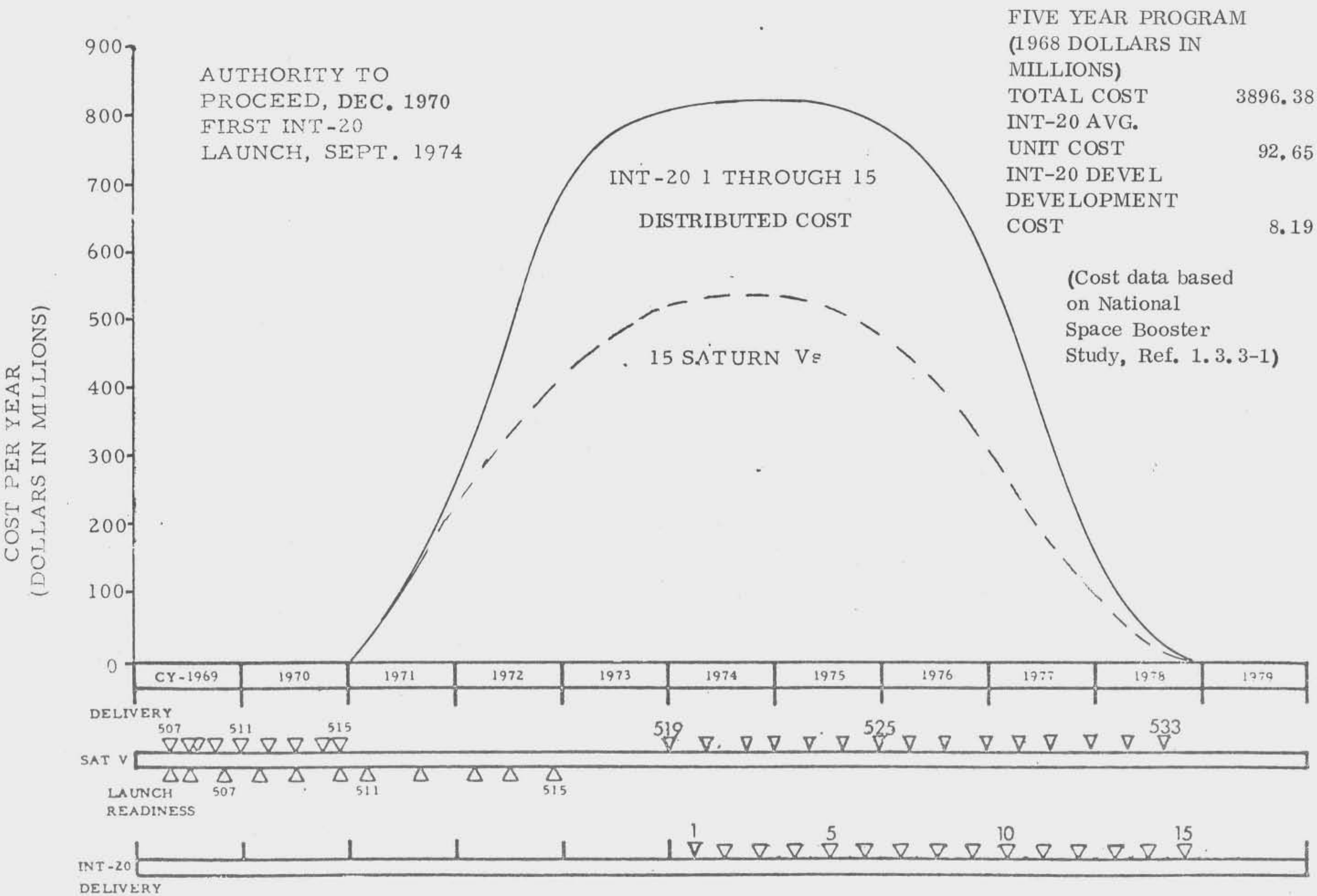

FIGURE 5.0-8 RESOURCE SUMMARY, 3 INT-20s WITH 3 SATURN Vs PER YEAR, NO STATIC FIRING (DISTRIBUTED COST METHOD)

5.0 (Continued)

h. The INT-20 delivery schedule

Cost data in this study are based on the "National Space Booster Study," Part One, Cost Analysis of Current Launch Systems, Saturn Systems Presentation, Contract NASW-1740, October 3, 1968, by Chrysler Corporation Space Division" (Reference 1.3.3-1).

The costs in this study do not reflect cost reduction programs now underway at NASA. The INT-20 launch and launch support costs were estimated by The Boeing Company.

The Development Program Plan includes:

A Design Plan which describes and schedules engineering effort necessary to prepare documentation.

A Test Plan which identifies the test articles necessary for development and schedules the test program.

A Manufacturing Plan which includes a statement of revisions and additions to tooling and manufacturing process and schedules for the production of test articles and stages.

A Facility Plan which identifies any new or modified brick-and-mortar construction needed for manufacture, test or launch. A schedule for facility modification is also provided.

A Schedule Plan which integrates the flow time requirements of design, test, manufacturing and facility implementation to provide the minimum practical time for delivery of the first flight article.

A Cost Plan which provides budgetary estimates for design, development, test and evaluation; and for production of the stages and launch of the vehicle.

The Development Program Plan for the INT-20 is based on supplemental effort to the present Saturn V program and uses the same facilities, equipment, procedures, and organization, supplemented or modified as necessary.

Ground rules and guidelines for preparing the INT-20 development program plan follow:

a. The program outlined to qualify the vehicle for operational flights shall include all facility modifications hardware, and test operations for all necessary ground testing (all-systems tests, dynamic test vehicle, injection stage test, etc.)

5.0 (Continued)

b. Man rating is required.

c. Funds will be assumed available as required.

d. The Saturn V INT-20 Program will not interfere with the existing Apollo delivery schedule.

e. A program definition phase (PDP) of at least six months will be required prior to stage development.

f. Stage development time will be consistent with completion of a test program.

g. Scheduling will not be calendar-oriented but will be based upon an assumed first Flight (Mid 1974) and appropriate time phasing to launch. (Amended by enclosed schedules).

h. Current stage acceptance test firing cost will separately be identified.

i. Maximum use will be made of existing facilities and tooling.

j. Cost analyses will be separated into two parts, (1) Non-Recurring or Development Costs including design, development, test and evaluation activities plus any man-rating flights and (2) Recurring or production costs. Costs for man rating flights will be stated separately. Recurring costs (and schedules) will be prepared assuming a rate of INT-20 production of two and four per year without the Saturn V, two and four INT-20 with two Saturn Vs per year, and three INT-20s with three Saturn Vs per year production.

k. Costs and schedules will be based on a one-shift, five-day week for engineering and a two shift, five day week for manufacturing.

l. The operational program will be at the rate of two, four and six deliveries per year and costs will be calculated for the first five years of operation (total of ten, twenty and thirty operational vehicles).

m. All stage, Instrument Unit and engine costs will be based on learning curve percentages, which will be coordinated with NASA.

n. Cost estimates will be in 1968 dollars without inflationary factors applied.

o. S-IC stage manufacturing facility costs, even though government owned, will be estimated. Costs at other government owned facilities (MTF, MSFC, KSC, Transportation, etc.) will be supplied by NASA/MSFC, if needed.

5.0 (Continued)

p. Costs for new and additional GSE/ESE needed at KSC will be included.

q. Spare parts costs will not be used.

r. Logistics planning is included in stage costs.

s. Costs will be shown in government fiscal year increments.

t. Costs will be total costs to the government, including all overhead and fee. All government manpower and transportation costs will be excluded.

u. Requirements and costs for dynamic test will be determined as a separate identity. The cost of removing the dynamic test stand from moth-ball condition will be determined by MSFC.

v. Cost of stage static test will be identified as a separate entity.

w. The study cost numbers will be based on those presented by the "National Space Booster Study," (Reference 1.3.3-1).

5.1 DESIGN PLAN

5.1.1 Vehicle

The engineering design will be performed by the respective stage and IU contractors. The F-1 and J-2 engines of the Saturn V will be used in the INT-20 and there is no indicated need for an engine design plan. The vehicle design plan describes documentation necessary to manufacture the INT-20 stages and the Instrument Unit. The S-IC design consists primarily of changing drawings and documentation to delete engine related hardware for the center F-1 engine. The S-IVB design consists primarily of analyzing environmental and mission differences experienced by the S-IVB on the two-stage INT-20 vehicle. The IU design consists of updating engineering drawings and documentation for the INT-20 mission, and modification of the Saturn IB IU checkout equipment to check out INT-20 IUs if the combined Saturn V/ INT-20 rate exceeds five per year.

System Engineering and Integration design effort is required to prepare analyses and documentation to support the INT-20 Earth orbital mission. Current SE&I functions are limited to those needed for the Saturn V Lunar Missions.

Design is preceded by a six-month's Program Definition Phase to prepare CEI Part I Specifications.

The INT-20 vehicle SE&I schedule is shown on Figure 5.1-1.

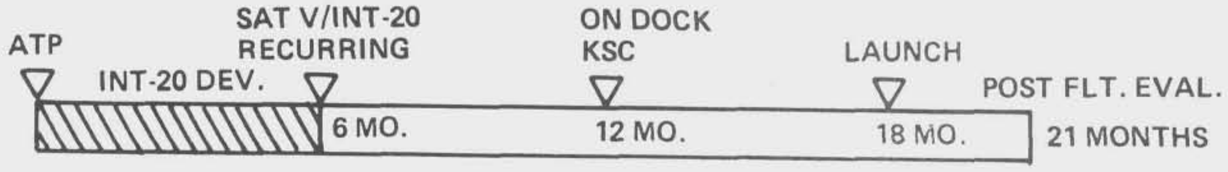

FIGURE 5.1-1 VEHICLE SE&I SCHEDULE

5.1.2 S-IC CONFIGURATION MANAGEMENT PLAN

The configuration management plan for INT-20 is based on a two phase evolution of the Contract End Item (CEI). The first phase is identified as the "Definition Phase." It encompasses the design and development of the End Item and is governed by performance and design requirements defined by a Part I CEI specification. The second phase, identified as the "Acquisition Phase," encompasses the production, testing, and delivery of the End Item. It is governed by requirements defined by a Part II CEI specification.

5.1.2.1 Definition Phase

A Program Definition Phase (PDP) of six months will be required prior to INT-20 contract go-ahead (Authority to Proceed, ATP). The following tasks will be included in the PDP:

1. Establish basic S-IC delta design requirements.
2. Determine the impact of Follow-ON S-IC design changes.
3. Prepare the Part I Contract End Item Specification.
4. Prepare and release engineering documentation for long lead items.
5. Initiate procurement source review and release Requests for Quotes (RFQ's) to vendors.
6. Start engineering design activity.
7. Start manufacturing planning.
8. Establish INT-20 documentation release system.

Completion of the above tasks during the Program Definition Phase will serve as a basis for accomplishment of the following Definition Phase tasks which will be initiated subsequent to INT-20 Authority to Proceed.

a. Authorize Long Lead Procurement

Engineering and procurement source review activities necessary to prepare for authorization of long lead procurement at the time of INT-20 contract go-ahead will be accomplished during the Program Definition Phase. These activities include: (1) identify long lead items, (2) prepare preliminary long lead item documentation, (3) release Engineering Advanced Material Releases (EAMR's), (4) prepare and release Requests for Quotes (RFQ's) for long lead items to vendors, and (5) negotiate with vendors and prepare purchase orders for long lead items. Hence, at the time of INT-20 contract go-ahead the long lead item purchase orders can be released and INT-20

a. Continued

procurement can start.

b. Finalize CEI Requirements

The results of this study and the Program Definition Phase will be evaluated and the INT-20 CEI requirements will be finalized. The part I CEI prepared during the Program Definition Phase will be up-dated if required. The resulting specification will be an addendum, which identifies delta requirements, to the Part I CEI for S-IC-3 through S-IC-10 (a part I CEI specification does not exist for S-IC-11 through S-IC-15).

c. Prepare Preliminary Drawings

Prepare design layouts, schematics, circuitry diagrams and preliminary design drawings as required to define and analyze component and system changes to the selected baseline stage. This design and analysis will use the design data of this report as a guideline for the design configuration and will be responsive to the requirements established during the Program Definition Phase. This task will be supported by stage and GSE/ESE systems and component analysis to establish criteria for the detail design and to corrobrate design decisions.

d. Implement a Development Test Program

The preliminary design study has identified no specific new requirements which would necessitate development testing to support INT-20 stage component or system design.

e. Perform a Preliminary Design Review (PDR)

A Preliminary Design Review will be conducted to:

(1) Verify that the selected design meets the design requirements.

(2) Verify compatibility with other systems equipment and facilities.

(3) Verify the producibility of the selected design.

f. Implement a Reliability Test Program

Based on the preliminary evaluation of the critical S-IC flight components and the INT-20 requirements defined in this study, no reliability re-testing will be required. A re-evaluation of the reliability test requirements will be necessary during the Definition Phase.

g. Implement a Qualification Test Program

Only two components have been identified for qualification testing. A re-evaluation of requirements and their impact on components of the INT-20 design will be required during the Definition Phase.

h. Establish Firm Design Requirements

Firm design requirements will be predicated on final stage performance requirements and systems analysis data.

i. Prepare Final Engineering Design Documentation

(1) The configuration definition under paragraph 4.2.2.1 of D5-17009-2 and the "add" and "delete" listing in Appendix A, Section 3 of D5-17009-3 are a measure of the design documentation task. This work will consist of the following:

(a) Prepare drawings of INT-20 peculiar new hardware designs.

(b) Prepare revisions of affected baseline configuration drawings.

(c) Prepare duplicate or revised Engineering Orders (EO's) to facilitate release of approximately 3000 existing baseline configuration EO's against INT-20 effectivities.

(d) Generate (by computer) a new set of all applicable baseline configuration Engineering Assembly Parts Lists (EAPL's) for release against INT-20 effectivities.

i. Continued

(2) The above tasks are based on the use of a new configuration data base for the INT-20 stages as shown in Figure 5.1.2-1. Several other documentation plans were investigated for configuration control of a mixed SATURN V S-IC/INT-20 production. These included:

(a) The use of a single data base for both SAT V and INT-20 Stages with sequential effectivity numbers.

(b) The use of a single data base for both SAT V and INT-20 Stages with block effectivity designations.

(c) The retrofit kit method, based on a standard 5 engine stage production for all effectivities plus retrofit kits for INT-20 effectivities.

The dual data base method is proposed as the system most manageable and adaptable to the existing S-IC automatic (computerized) documentation release system. The kit system is definitely not recommended for production implementation because of its high cost impact.

j. Conduct a Critical Design Review

The critical design review will:

(1) Assure compatibility of the CEI, as designed with the Part I CEI Specification (addendum).

(2) Assure compatibility of the completed design, as reflected on the Engineering Drawings, with the Interface Control Drawings.

(3) Verify compatibility of the completed design by review of analytical and test data.

k. Release Final Engineering Drawings

l. Begin manufacture of the first article CEI and basic S-IC design components for subsequent CEI's.

m. Establish Contract Acceptance Test Requirements.

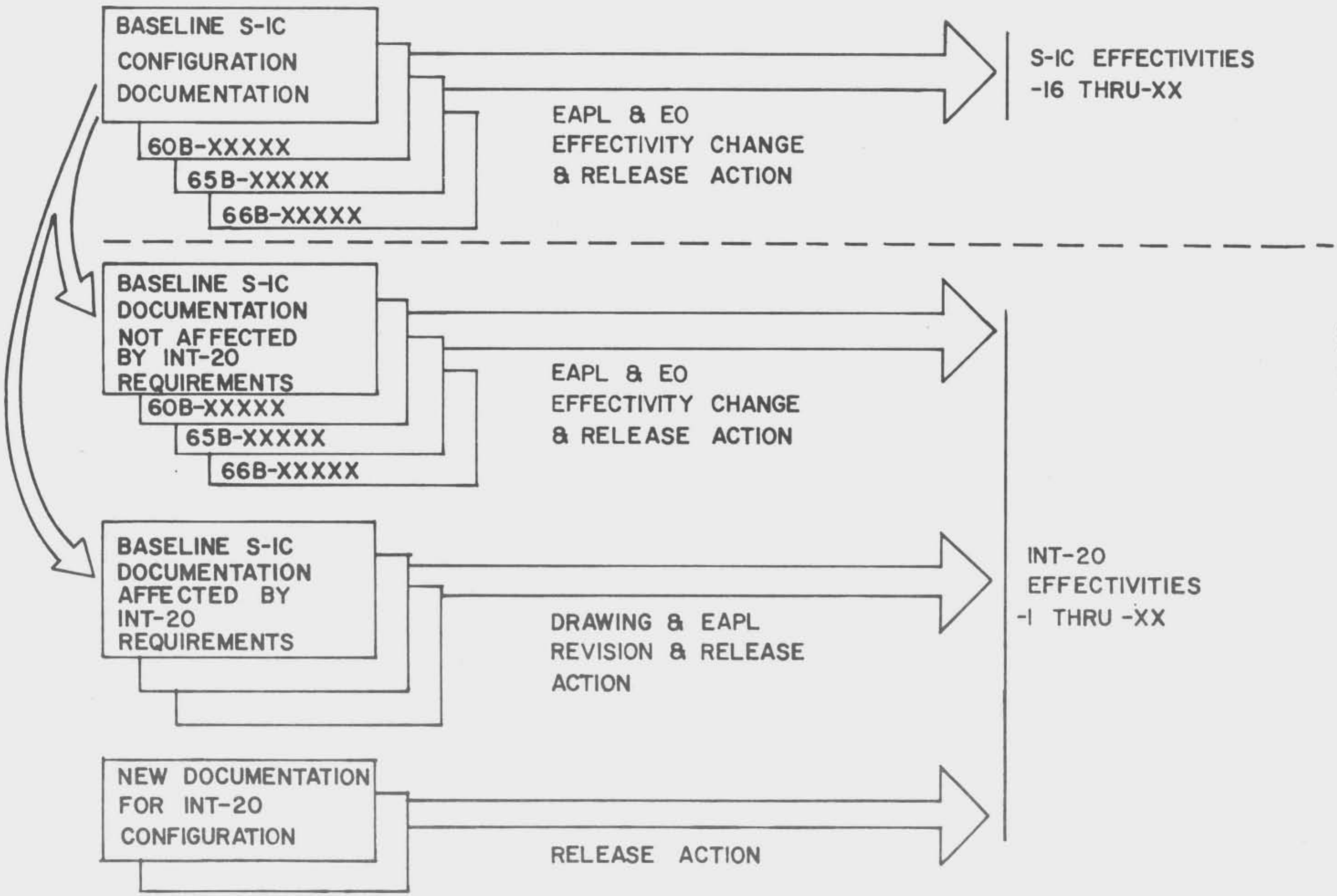

FIGURE 5.1.2-1 S-IC/INT-20 DOCUMENTATION PLAN (FOR PRODUCTION)

n. Begin Preparation of a Part II CEI Specification.

o. Complete Manufacturing of the First Article CEI.

5.1.2.2 Acquisition Phase

Requirements and tasks applicable to the Acquisition Phase of the configuration management program for INT-20 are as follows:

a. Perform First Article Configuration Inspection (FACI). FACI shall be performed to the delta (INT-20 peculiar) requirements to the basic S-IC-10 stage design.

   1. Verify that the "as-built" configuration is identical to the configuration documented on the Class I engineering drawings.

   2. Verify compatibility between the "as-qualified" configuration and the "as-manufactured configuration." Differences between configuration of qualification tested units and FACI'd units shall be recorded.

   3. Validate the acceptance test requirements specified in the Part II CEI Specification by direct comparison of the test methods and test data with the performance/design requirements.

b. Implement FACI approved Part II CEI Specification.

c. Proceed with manufacturing of CEI's subsequent to FACI'd article.

5.1.2.3 Configuration Management Schedules

The time phasing of critical Definition Phase events is shown on Figures 5.1.2-2 and 5.1.2-3. As noted in Section 5.2.2, these schedules are based on in-sequence production of INT-20 configuration stages without reallocation of production hardware.

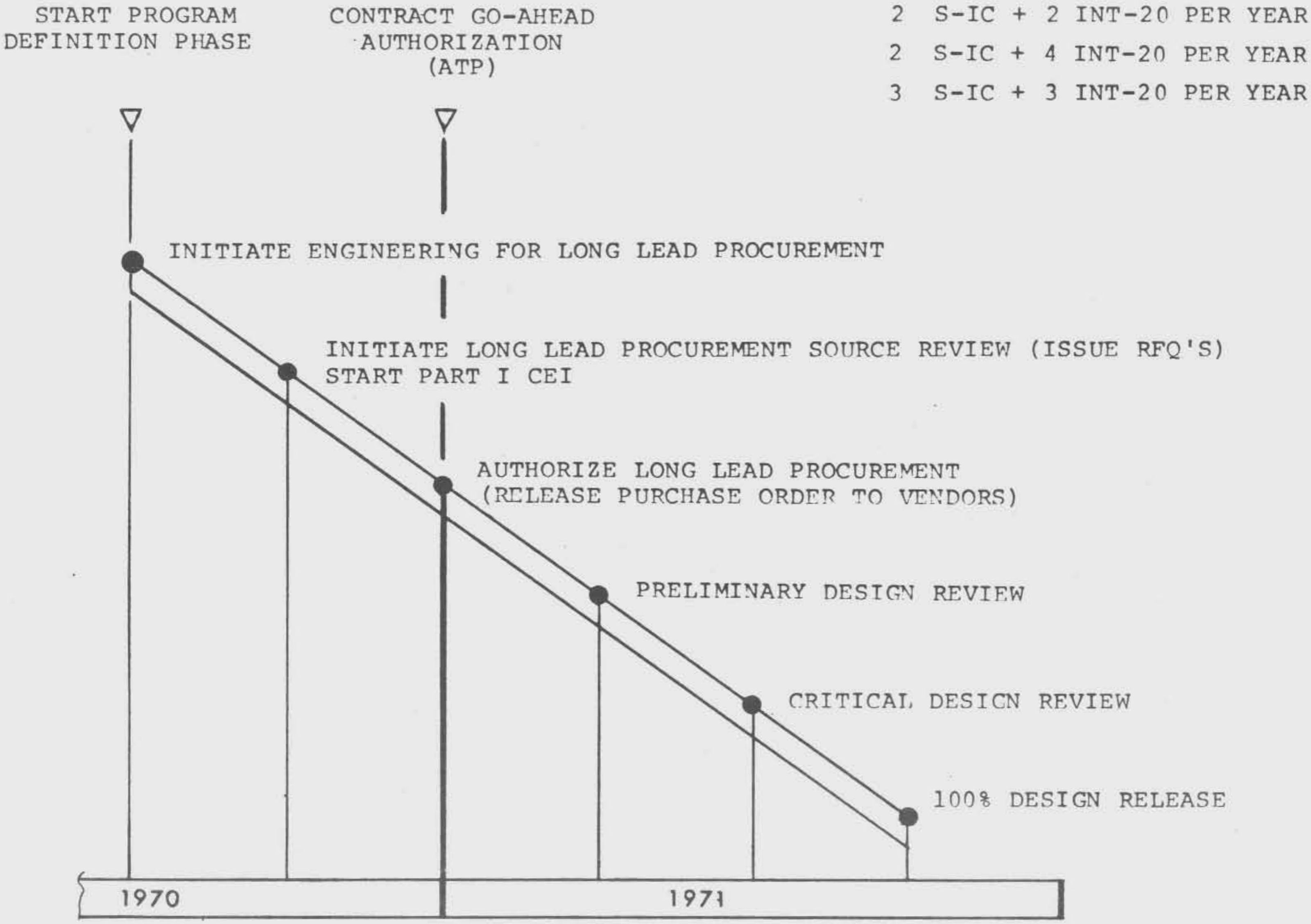

FIGURE 5.1.2-2 S-IC DEFINITION PHASE MILESTONES

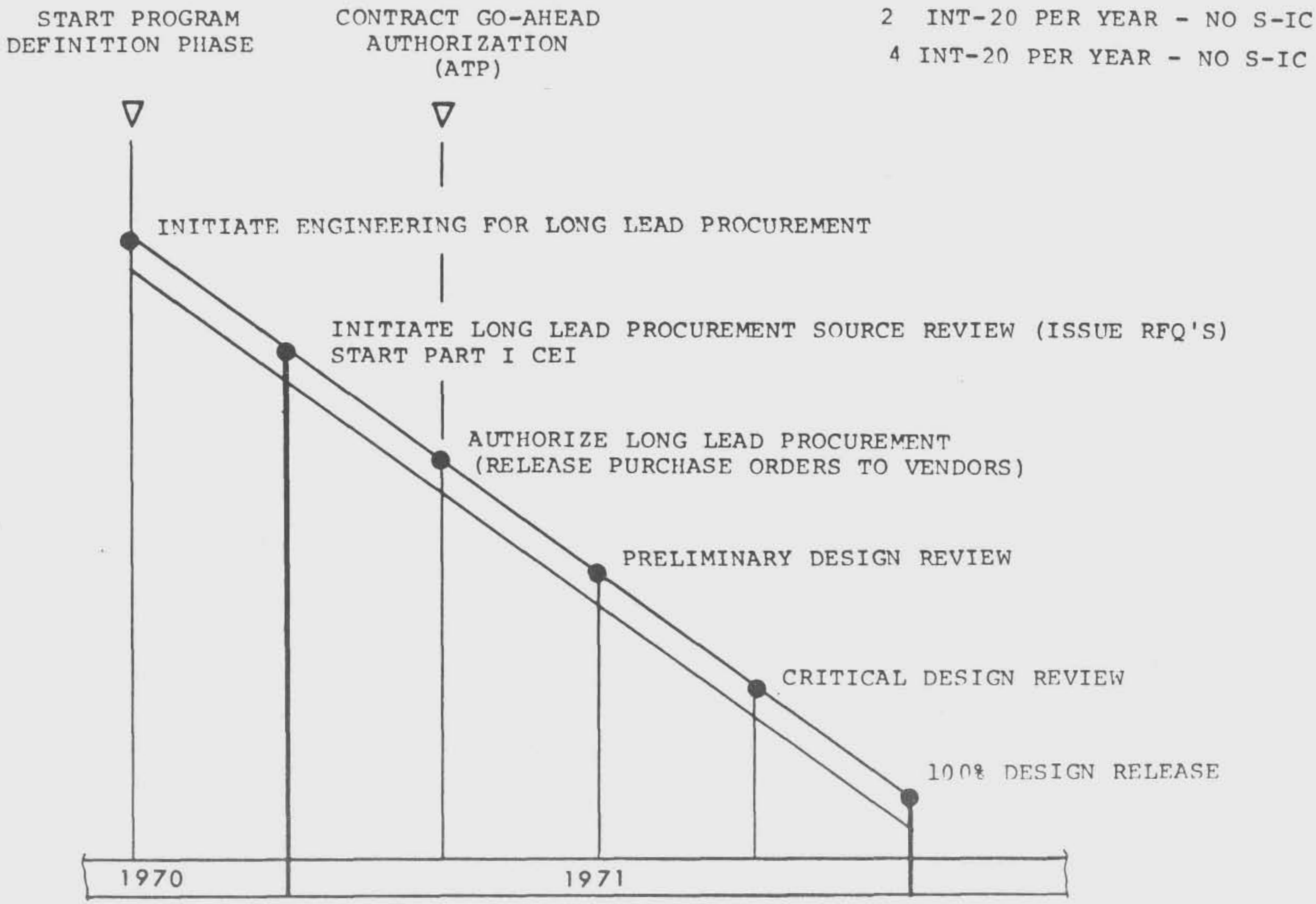

FIGURE 5.1.2-3 S-IC DEFINITION PHASE MILESTONES

5.1.3 S-IVB Stage Design Plan

The development program plan for the INT-20/S-IVB stage is based on supplemental effort to a concurrent Saturn V/S-IVB program and utilizes the same resource base, including personnel, facilities, equipment, procedures, and organization, supplemented or modified as necessary.

5.1.3.1 Program Definition Phase

Program plans and design requirements will be identified. A specific area of investigation will be the expected higher acoustic levels for the S-IVB stage due to its being closer to the S-IC stage engines. The need for qualification of critical components to the expected higher levels must be analyzed to determine test plan requirements. If required, these would be in the nature of extensions to the previous qualification testing envelopes to minimize retesting. Concurrently, the need for redesign of any components due to the higher levels will be assessed. The six month period assumed for PDP should be more than sufficient for the effort required.

5.1.3.2 Analysis and Design

Design requirements will be reviewed and analyzed to determine those specific departures from the existing Saturn V/S-IVB specifications due to environment differences, mission differences, and identified deletions and modifications to existing components. These departures were identified in Sections 4.2.3, 4.2.4, and 4.3.3 of this report. The impact of these departures on component design and interfaces must be documented and the least cost methods of implementing them determined. Revisions and modifications to existing S-IVB production and interface control drawings will be accomplished. In many cases design memorandum effectivity deletions and revisions will suffice. In a few instances, such as some instrumentation wire harnesses and some interstage options, new top drawings may be required. In general, the design effort for the baseline INT-20/S-IVB (minimum modifications) is approximately half that required for the alternate INT-20/S-IVB (maximum deletions). The design effort schedule is presented in Figure 5.1.3-1 and is essentially the same for either the baseline or alternate. After two months of environment and mission impact analyses, those production and ICD drawings unaffected by the INT-20 requirements will be identified, and initial release should occur by the third month. Analysis is complete by the fourth month with final drawing release and design complete by the fifth month. Repressurization system deletions and instrumentation/wire harness modifications are the pacing items.

This schedule assumes no significant redesign is required due to expected higher acoustic levels of the INT-20 environment.

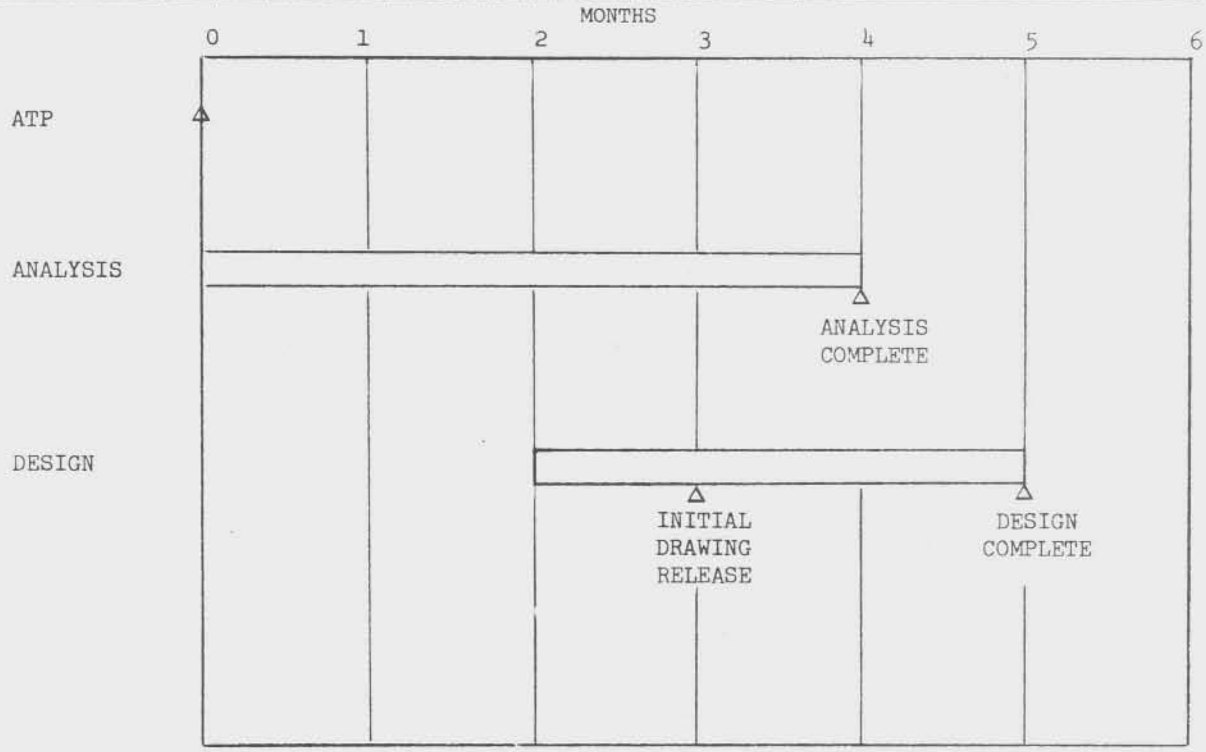

Figure 5.1.3-1. INT-20/S-IVB Design Plan Schedule

5.1.4 IU Design Plan

The IU design plan encompasses only the IU assembly hardware and the ground support equipment (GSE) for systems test which are affected by the INT-20 vehicle. The design plan involves the following:

IU stage description.

IU Ground Support Equipment Modifications.

Schedule of the engineering effort to prepare drawings and documentation for the INT-20 fabrication.

A minimum-modification approach will be taken to design the required changes to the INT-20-IU hardware, and provide one basic configuration level of the components. With appropriate substitution of flight programs, the software will enable the control and sequencing of hardware and events so that the IU has common usage for the Saturn V or the INT-20. Specific signal channelization, analytical studies, configuration of simulation software, and specific mission imposed changes are released on a mission-to-mission, vehicle-by-vehicle basis with a specific engineering release by numbered IU.

The design plan, in general, therefore follows the normal cycle of prerelease engineering rework with updating on the basis of vehicle effectivity. In the following paragraphs, specific, one time nonrecurring design effort is described.

5.1.4.1 IU Stage Description

The Instrument Unit is a cylindrical structure 6.6 meters (260 in) in diameter and 0.9 meters (36 in) in height, mounted on top of the S-IVB stage.

The structure of the IU consists of three 120-degree segments of aluminum honeycomb sandwich-joined to form a cylindrical ring. After assembly of the IU, a door provides access to the electronic equipment inside the structure. This access door has been designed to act as a load-carrying part of the structure in flight. In addition, the structure contains an umbilical door which is spring-loaded to close after retraction of the umbilical arm at liftoff. The IU structure provides a path for static and dynamic loads resulting from the payload above the IU.

The electronic equipment boxes of the IU are mounted on coldplates which are attached to the inner side of the cylindrical structure. The electronic equipment in the S-IVB stage is mounted in a similar way. This arrangement provides clearance for the landing gear of the Lunar Module sitting on top of the IU and for the bulkhead of the S-IVB tank extending into the IU.

5.1.4.1 (Continued)

The IU contains the equipment necessary to:

Perform guidance and control of the vehicle from liftoff to separation of the payload.

Aid in radar tracking of the vehicle.

Perform a command link for control of the vehicle from the ground.

Provide temperature control for the electronic equipment in the IU and the S-IVB stage forward skirt.

Telemeter data to ground receivers.

5.1.4.2 Schedule

a. IU Ground Support Equipment Modifications

The facility for assembly of IU's has a separate checkout complex for the Uprated Saturn I and for the Saturn V. Each of the sections is rate limited to approximately five IU's of each type per year. The study has established that INT-20 and Saturn V IU's are sufficiently alike that when either IU arrives at the final Saturn V checkout station, minor electrical modifications will have been made in the facility within the scope of vehicle-to-vehicle change activity. On the basis that a Saturn V and INT-20 IU are alike, a combined Saturn V/INT-20 delivery rate of six per year would saturate the Saturn V checkout complex which is rate limited to approximately five per year. Prior ECP activity has established the feasibility of converting the Uprated Saturn I line to either US-1 or Saturn V which would then permit the facility to accommodate INT-20, Saturn V, Uprated Saturn I in an in-line or a retrofitted basis. The modification can be delayed until firm mission planning or contractural arrangements dictate six per year rates. The lead time required for conversion is shown in Figure 5.1.4.2-1. The one year period need only preceed the final checkout of the first INT-20 IU. The worst case would be a conversion design cycle beginning July 1, 1972, three months after ATP.

b. Flight Control Computer/Filter Design Schedule

The flight hardware development activity for the INT-20 IU involves only minor changes to the FCC to provide addition gain switching in the S-IC burn. To preserve the interchangeability of Saturn V and INT-20 FCC's, the S-II stage circuitry will be retained. Shown in Figure 5.1.4.2-2 is the normal FCC routing

5.1.4.2 (Continued)

into the manufacturing schedule and the FCC design modification time period. Note that there is no impact. The required nine weeks from final filter release to delivery to IBM is retained. Between delivery and entry into the final IU checkout, the FCC is given acceptance testing and simulation laboratory dynamic testing.

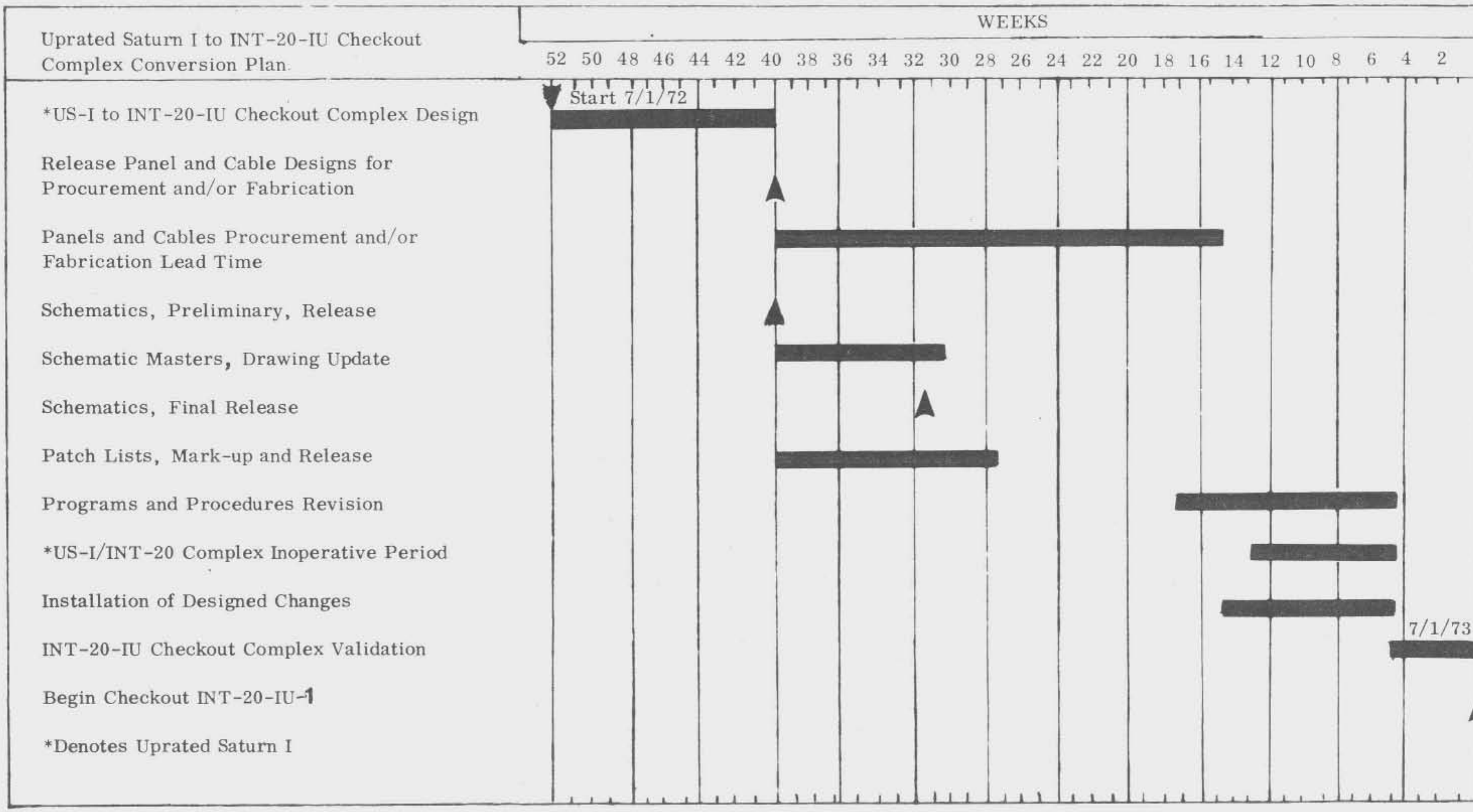

FIGURE 5.1.4.2-1. UPRATED SATURN I TO INT-20-IU CHECKOUT COMPLEX CONVERSION PLAN

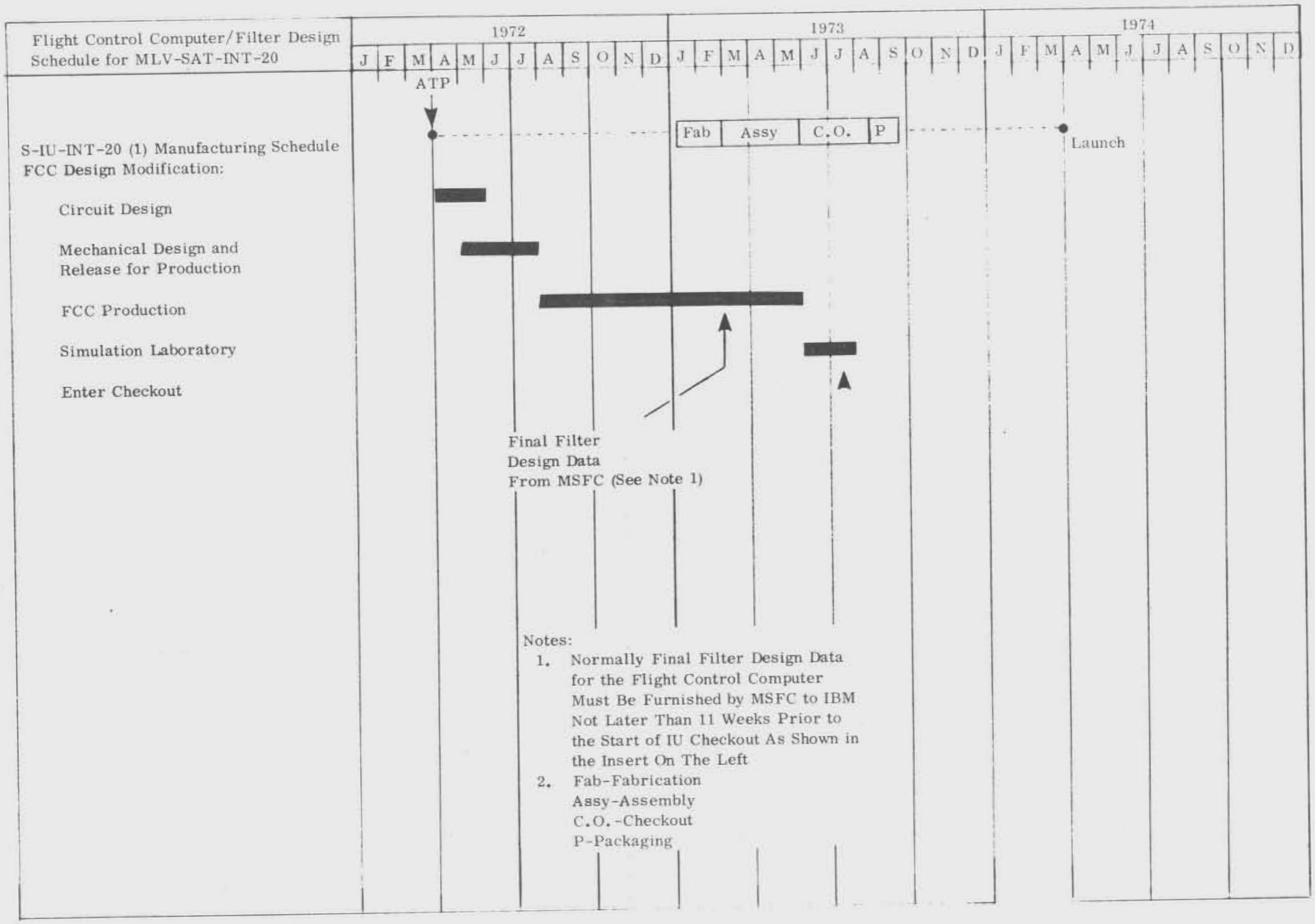

FIGURE 5.1.4.2-2. INT-20 FCC FIRST DELIVERY

5.2 TEST PLAN

The test plan for the vehicle and each stage of the INT-20 outlines the testing program required to qualify new or revised parts, components and systems; verification of design; manufacturing and test changes; and end item tests including test firing of stages and a vehicle test flight if deemed to be necessary. F-1 engine testing is included with the Vehicle Test Plan.

5.2.1 Vehicle Test Plan

5.2.1.1 Dynamic Test

A dynamic test is not necessary for the INT-20 vehicle. Dynamic characteristics of components and inputs to the vehicle guidance and control system can be obtained from analysis and correlation of Saturn V flight data, Saturn V dynamic test data and flexure model tests. When the MLV payload configuration is established, dynamic testing may be needed for a "short stack" dynamic vehicle which consists of an S-IVB, IU and payload.

5.2.1.2 First Operational Flight

The first flight of the INT-20 is operational and should perform a useful unmanned mission. The unmanned flight is considered necessary because the structural and functional configuration differences between the INT-20 and the Saturn V are significant and because the new design separation interface can only be qualified by flight. The unmanned flight is considered necessary whether the first INT-20 vehicle is made from retrofitted Saturn V stages or initially fabricated in the final INT-20 configuration. The requirement for the first flight to be unmanned is derived from The Boeing Company only. McDonnell Douglas and IBM feel that the first flight could be manned with respect to the INT-20 S-IVB stage and Instrument Unit, respectively.

5.2.1.3 Wind Tunnel Test

Wind tunnel force and pressure model tests will be needed to determine aerodynamic characteristics of the INT-20 configuration and payload. The wind tunnel tests would be performed in government wind tunnel facilities and could be performed during or before Phase C. Duration of wind tunnel testing is estimated to be 10 to 12 months. Materials and manpower would be government furnished and no cost is indicated in this study.

5.2.1.4 F-1 Engine Test

The S-IC stage with four F-1 engines, when part of the S-IC/S-IVB vehicle, has a longer duration engine firing than the S-IC stage of the Saturn V. The F-1 engines of the Saturn V have a firing duration of about 160 seconds. F-1 engines have been

5.2.1.4 (Continued)

test fired up to about 194 seconds. The F-1 engines of the S-IC/S-IVB vehicle must fire about 230 seconds for a 100 NM Earth orbital mission and up to about 240 seconds for synchronous orbit and space probe missions. Therefore, a test program must be established to qualify F-1 engines for a firing duration of at least 230 seconds for manned flights and 240 seconds for unmanned flights.

The F-1 engine test program requirements were determined by NASA-MSFC. The test program would be performed on the NASA-MSFC engine test stand by NASA personnel. Two auxiliary propellant tanks will be added to the permanent tankage of the test stand to provide the additional propellant needed for the longer duration firing. The additional tankage will provide propellant for at least 240 seconds of F-1 engine firing. Static firing test will be made on one F-1 engine with five hot firings up to a duration of 240 seconds. MSFC estimates a cost of $225,000 for hardware, material, propellants and data tape procurement. Actual tests will require about three months with a three months preparation period and about two months to write the test reports. The schedule is shown on Figure 5.2.1.4-1.

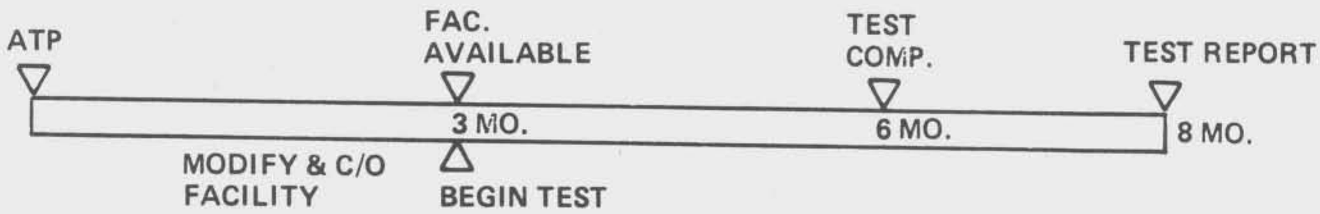

FIGURE 5.2.1.4-1 F-1 ENGINE TEST PROGRAM SCHEDULE

5.2.2 S-IC Test Plan

Baseline S-IC requirement and configuration changes identified by this preliminary study have been evaluated to determine the need for added or revised testing. Consideration was given to new component test requirements, to system and component changes which could require development or validation testing, and to environment and flight profile changes which could invalidate the baseline S-IC-10 components and systems qualification and reliability status.

5.2.2.1 Development Tests

No component or system development test requirements have been identified by this study.

5.2.2.2 Qualification Tests

a. Qualification testing for new or revised components

   Only two hardware items defined for the INT-20/S-IC configuration require qualification testing. These items are:

   1. The lengthened fuel loading probe defined in FIGURE 4.2.2.1-23.

      This item will require vibration testing in accordance with requirements established during the Design Definition Phase.

   2. The LOX interconnect spool support defined in FIGURE 4.2.2.1-15.

      This item will require static load testing in accordance with requirements, for the design of the part, established during the Design Definition Phase.

      All other new or revised components can be qualified by analysis or by similarity.

b. Requalification of existing qualified hardware to revised requirements

   Preliminary requirements identified by this study indicate that the functional and environmental conditions for the INT-20 are either the same as or less severe than for the baseline S-IC stage except for flight duration time, local area aerodynamic heating, tank pressure, and base region heating. The impact of these conditions on the qualification status of S-IC parts was evaluated during this study. No requalification tests are considered necessary based on this preliminary assessment; however, qualification status of existing hardware should be a subject for additional study during the Design Definition Phase.

5.2.2.2 (Continued)

c. Reliability Testing

There is no need for reliability testing the INT-20 first stage reliability critical components. This preliminary assessment was based on the following:

1. The INT-20 first stage has no additional reliability critical components than the baseline S-IC stage.

2. The INT-20 first stage reliability critical components are exposed to the same or less severe external and operating environments.

3. The flight time for the INT-20 first stage is a maximum of 216 seconds.

4. The S-IC stage CEI Specification reliability design objective is 0.95. For the first flight of INT-20, the projected first stage reliability exceeds the S-IC CEI Specification reliability design objective (See Section 5.0 of Appendix A, INT-20 Reliability Assessment).

A reassessment of the reliability critical components against INT-20 requirements, however, would be required during the Design Definition Phase.

d. End Item Test Plan

A draft of End Item test requirements, based on the changes defined in 4.2.2.1 of this study, was prepared to impact the INT-20 configuration and requirement changes on test and checkout functions and equipment. These preliminary requirements are documented in 66B10920.

e. Other Tests

Acceptance tests and checkout requirements and procedures for the INT-20 first stage will be generated during the Design Definition Phase and will be responsive to the final design requirements for the stage. These requirements and procedures include the following:

1. Stage Test and Checkout requirements and procedures.

2. Specifications and criteria for prelaunch checkout and launch operations.

3. Acceptance test procedures for MAF and MTF.

The impact on KSC acceptance test procedures, prelaunch checkout procedures and launch operations is not included in this report.

5.2.3 S-IVB Stage Test Plan

5.2.3.1 Development Test

No development tests will be required for the INT-20/S-IVB stage (baseline or alternate).

5.2.3.2 Qualification

No qualification tests will be required due to the modifications or deletions of S-IVB subsystems for the INT-20; however, due to expected higher acoustic levels during boost flight it is probable that some critical components must be requalified. Data obtained on the S-IVB during the Saturn V flights indicate that dynamic levels during liftoff on some S-IVB critical components are higher than previously predicted (these components have been subsequently requalified to the higher levels). Since the acoustic levels on an S-IVB flown as second stage on the S-IC booster are estimated to be about 25% higher than the levels on the Saturn V/S-IVB, it is anticipated that some components would need requalification. Specific acoustic and vibration data from static firing of the S-IC stage with 4F-1 engines, and/or specifications imposed by NASA would be required to perform a detailed evaluation of each critical component to determine requalification test requirements. Such an investigation would be required in the Program Defininition Phase. Since this information and scope are not available at this time, a brief evaluation has been performed based on projections of the Saturn V acoustic and vibration data. This evaluation indicates that approximately 10 percent of the S-IVB critical components may require requalification for INT-20 use. This number may increase or decrease depending upon the results of the PDP investigation. Based on the brief evaluation the following ten components and/or subassemblies were selected as probable requalification items and represent the scope of the effort anticipated.

1. Chilldown pump - LOX
2. Chilldown pump - LH_2
3. PU probe - LOX
4. PU probe - LH_2
5. PU electronics
6. LOX NPV duct
7. LOX NPV valve
8. LOX shut-off valve
9. LOX internal vent line
10. LH_2 diffuser

It is estimated that the requalification test effort would not exceed six months duration. Test fixtures and equipment are available, but test articles would have to be procurred and fabricated. Assuming zero or minimal redesign of these articles, procurement and fabrication lead time of four months prior to start of testing is adequate. Figure 5.2.3-1 indicates the test plan schedule.

5.2.3.3 Structural Testing

Predicted loads (ultimate factor of safety = 1.4) are within the S-IVB stage structural capability with the possible exception of aft interstage peak acceleration loads with structural heating. The suitable application of insulation material to the aft interstage is expected to provide adequate structural margins. Therefore, no structural testing would be required.

5.2.3.4 Dynamic Testing

Sufficient correlation is now available among Saturn V dynamic test, analysis, and flight data to accurately predict INT-20 bending mode shapes and frequencies. No vehicle dynamic testing is anticipated.

5.2.3.5 Acceptance Testing

Due to deletions and modifications to S-IVB subsystems some acceptance checkout procedures will be revised. These revisions are of a minor

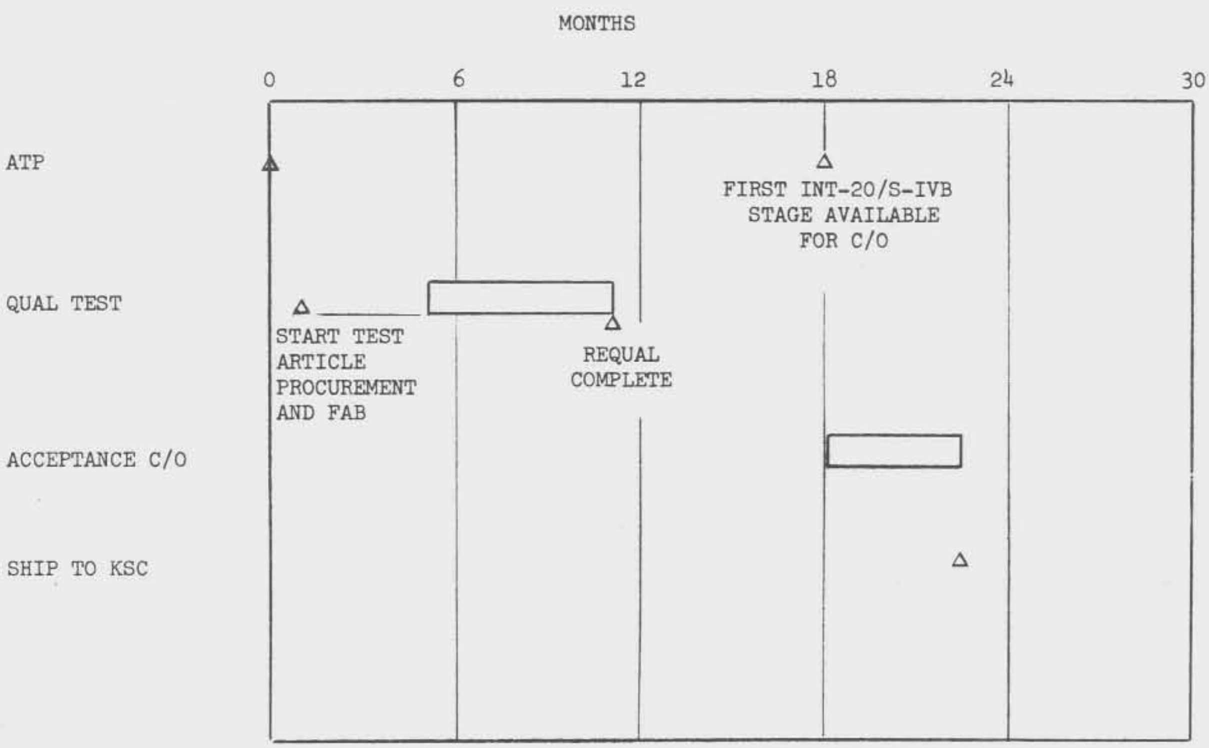

Figure 5.2.3-1. INT-20/S-IVB Test Plan Schedule

nature and will have no appreciable effect on the acceptance checkout. Following acceptance checkout at Huntington Beach, INT-20/S-IVB stages will be shipped directly to KSC for launch. It is presumed that static firings of S-IVB stages will have been terminated, and since the subsystems deletions and/or modifications of INT-20/S-IVB stages will be of a minor nature, no resumption of static firing will be required.

5.2.3.6 Flight Test Program

The nearly identical configuration of the INT-20/S-IVB with the Saturn V/ S-IVB except for deletion of unused systems and the requalification of critical components to the INT-20 acoustic and vibration levels should establish sufficient confidence to preclude the requirement for a flight test program. One possible problem area is the confidence which may be placed in the vehicle dynamic analyses. If this can be adequately established, the first flight of the INT-20/S-IVB may be considered primarily operational rather than test.

5.2.4 IU Test Plan

This section contains a brief discussion of the assembly and checkout operations that would be required to accommodate an IU configured for an INT-20 vehicle.

Reversibility from present Saturn V IU's to INT-20 configuration and vice versa can be achieved with minimum modification constraint to any hardware and consequently to any procedures. Minor modification to the FCC, Control Distributor and IU/S-IVB/S-II interface do not warrant requalification testing.

Therefore, there are no requirements peculiar to the INT-20 vehicle configuration.

a. Assembly

The assembly techniques would be the same as those required for present IU's.

b. Component Checkout Test

Component checkout would be the same as that used in the present test system components would receive through acceptance testing in the Huntsville facility or at the vendor.

c. IU System Test - Huntsville

Manufacturing checkout is a series of functional tests, which will demonstrate that all IU flight hardware will satisfy design and mission objectives and requirements when operating independently or compositely. Checkout proceeds from each individual subsystem and continues until "overall tests" are satisfactorily completed.

The checkout programs used during Systems Checkout consists of automated, semiautomated, and manual test procedures. The automated and semiautomated test procedures are used primarily in the Networks and Guidance and Control (G&C) tests. Manual test procedures are used primarily in the Radio Frequency (RF), Measurements, Telemetry (TM), and Electro-Magnetic Interference (EMI) tests.

5.2.4 (Continued)

The electrical and mechanical systems are tested in a sequence progressing from subsystem tests to overall systems tests. The checkout sequence is designed so that a complete IU checkout may be performed in a minimum amount of time to satisfy all test objectives of the test program while maintaining minimum running time on time-critical components.

An overall chart of the system checkout activities is shown in Figure 5.2.4-1. The major events and the test time for each event are shown beginning from the time the IU enters checkout until it is prepared for shipping.

Modification and/or deletions required for INT-20 configuration are considered on a minimum modification basis to retain reversibility. Selection of the S-V IU as a basic INT-20 configuration also reduces Huntsville Manufacturing impact to a minimum. The Flight Control Computer will require modification in order to provide four S-IC switch points instead of two presently used. Two presently reused switch points will be utilized to produce minimum impact on system test.

Interface wiring and switch selector wiring presently used for S-II stage function will not be necessary for INT-20 and will be carried as spare.

These modifications can be handled by normal fabrication, assembly and checkout.

5.2.4.1 Automated System Checkout Programs

The following subsystem checkout programs will require test parameter changes and/or minor program rewrite for INT-20 configuration.

a. Control Subsystem

1. Control Subsystem

2. A_1 gain

3. A_0 gain

4. Control Computer Relay Redundancy

5. Control System Nulls

6. Engine Deflection

5.2.4.1 (Continued)

b. Electrical Subsystem

1. PD&C
2. General Networks
3. Sinc Plug Drop

Documentation affecting system test (HSV and CKF) as a result of additional INT-20 requirements are:

a. Electrical Schematics

b. Component Schematics

1. FCC
2. Control Distributor

c. ICD's

d. IP&C List

e. System Test Specifications

f. Assembled Stage Test Procedures

5.2.4.2 Sequence of Testing

GUIDANCE AND CONTROL GROUP

Stabilizer Section:

	Test Title	Milestone (See Figure 5.2.4-1)
1.	St-124M Stabilizer Functional Test	7-8 11-12
2.	Stabilizer Power Up/Power Down Program Test	11-12
3.	ST-124M System Alignment Test	14-15
4.	ST-124M System and LVDC/LVDA Integrated Test	14-15

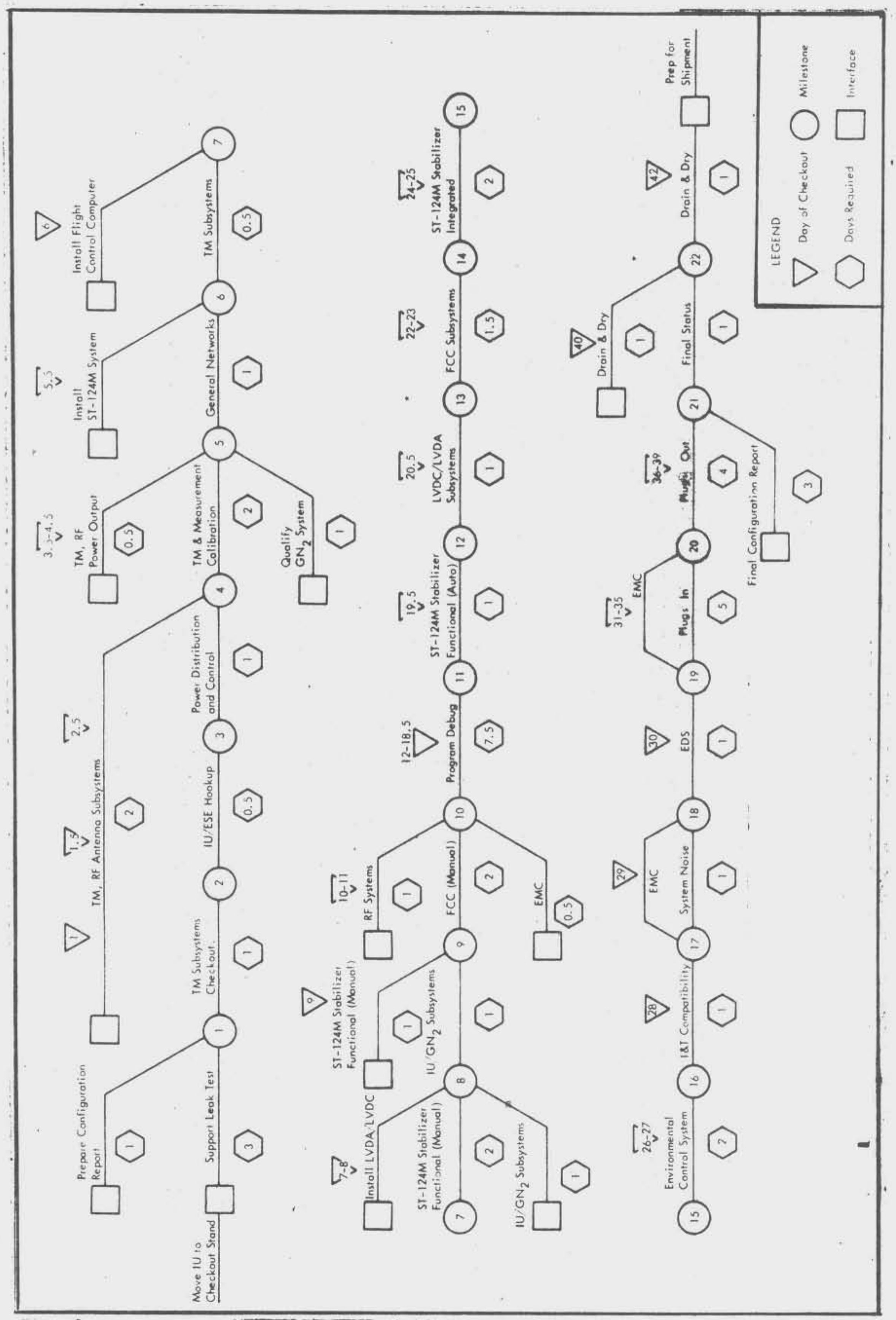

FIGURE 5.2.4-1. CHECKOUT ACTIVITIES FLOW CHART

5.2.4.2 (Continued)

Guidance Section

	Test Title	Milestone
1.	Measurement Identification Test	12-13
2.	LVDC/LVDA Switch Selector Test	12-13
3.	LVDC/LVDA Power and/or Redundancy Test	12-13
4.	LVDC/LVDA Accelerometer Processor Test	12-13
5.	LVDC/LVDA Ladder Output Test	12-13
6.	LVDC/LVDA Self-test	12-13
7.	LVDC/LVDA Discrete Input Test	12-13
8.	LVDC/LVDA Discrete Output Test	12-13
9.	LVDC/LVDA DDAS Test	12-13
10.	Computer Interface Unit Test	12-13
11.	Command System Test	12-13
12.	LVDA Pin Function Test	7-8

5.2.4.3 Facilities Test Plan

The Instrument Unit Systems Checkout Facility used to perform IU systems checkout and all related supporting tasks that are required to assure the achievement of the following objectives.

a. Operation compatibility of the IU with the ESE through the umbilical interface.

b. Operation compatibility of the IU with the special test equipment required in each of the satellite test stations and control room.

c. Compatibility of the IU with the spacecraft and lower stage simulators.

d. Operational integrity of the IU design and specific mission objectives.

e. Acceptability of the IU for flight performance as required by the IU Test Specifications.

Figure 5.2.4-2 is the functional room layout of the IU System Checkout Facility. It is essentially divided into two areas. The area in which the IU's will be located is the Hi-Bay Area. The remaining area in which the testing stations are located is the Checkout Complex Area. The Hi-Bay Area will accommodate two IU's.

Under the 30 ft high ceiling will be a traveling bridge crane, with a 46 ft center-to-center span, complete with a motorized trolley and electric hoist of five ton capacity. The crane will be utilized to transport the assembled IU within the Hi-Bay Area and to place it on the test stand. The test stands are equipped with leveling and alignment adjustments for the yaw, pitch and roll axis.

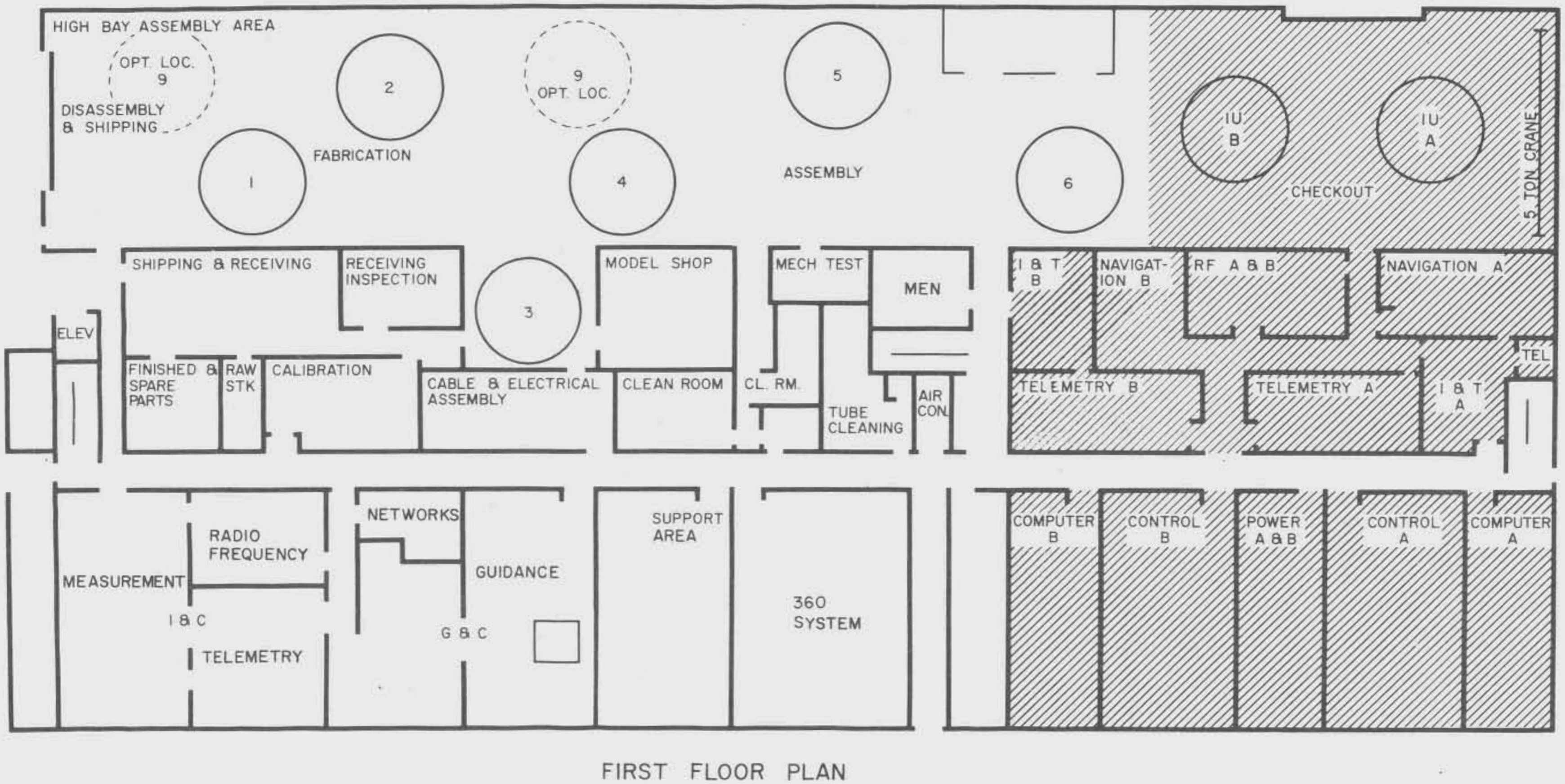

FIGURE 5.2.4-2 MANUFACTURING CHECKOUT AREA

5.2.4.3 (Continued)

The air conditioning system for the Hi-Bay Area is designed to maintain a relative humidity varying with the occupancy load from 57 percent maximum to 53 percent minimum while maintaining a temperature of 72°F.

Covered trenches in the floor of the Hi-Bay Area are used for the cabling and piping for the IU's to the various checkout rooms. Two umbilical supports are furnished to handle the umbilical cabling between the IU and its associated checkout station area.

Two Ground Support Cooling Units, one for each IU under test, are located in a room within the Hi-Bay Area. The unit is required during vehicle test to control the temperature of the equipment mounted in the IU. The servicing functions include filling the vehicle cooling system with methanol/water and purging the cooling system with gaseous nitrogen.

The S-IVB Heat Simulator, used during Environmental Control Systems Test and Thermal Conditioning subsystem test, shall provide a heat input to the thermal conditioning system variable between 0 and 9 kw. The unit is portable and is stored in an area providing maximum protection from damage.

Other equipment in the Hi-Bay Area includes the Electromagnetic Compatibility test consoles. These provide a source of discrete or broadband frequencies as required to verify compliance of the IU systems to the susceptibility requirements. They introduce simulated interference into the most critical points of the subsystem as it is being monitored for any malfunction caused by the interference.

5.2.4.4 Power Room

The following list indicates major equipment housed in the power room:

a. Monitoring Panels.

b. Nickel-Cadmium Batteries.

c. MG sets of Ground Control Computer System Power.

d. DC Power Supplies.

e. AC Power Supplies (400 cycle).

f. Interface Patch Racks.

g. Distribution Patch Racks.

h. Magnetic Amplifier Signal Conditioners.

5.2.4.5 Control Room

The Control Room is used to perform the overall test on the IU networks, Gas Bearing/Thermal Conditioning Subsystem Supply, Environmental Control System, and integrated testing under control of the Ground Control Computer System.

The equipment located in the Control Room consist of:

a. Environmental Control System Control Panels.

b. Environmental Control System Display Panels.

c. Control System Display Panels.

d. Guidance System Display Panels.

e. Power Equipment Status Panels.

f. Power Recorders.

g. Guse Racks.

h. Computer Analog Signal Conditioners.

i. Stage Interface Test Set.

j. Discrete Sequencing Displays.

k. Distributors and Patch Racks.

l. Digital Events Evaluator.

m. Ground Control Computer System Remote Display Console.

n. Overall Test Panels.

o. Gas Bearing System Control Panel.

p. Count Down Clock.

Control Room Networks functions are required for all subsystem testing. All power and most inputs and outputs from the IU pass through the Control Room.

5.2.4.5 (Continued)

The IU networks evaluation is accomplished by providing inputs and monitoring the responses of the electrical systems. The Control Room provides the following control and monitoring functions during IU checkout:

a. Ground and IU power sources.

b. Necessary Switching for Launch preparations.

c. Checkout of Guidance and Control functions in association with the Navigation Test Station.

d. Gas bearing and cooling system.

e. Necessary Vehicle Stage Interface simulation and substitution.

5.2.4.6 Telemetry Ground Station

The Telemetry Ground Station receives all airborne telemetry signals from the IU Telemetry Stations via an RF link. During systems testing these signals are recorded on magnetic tape for later demodulation and oscillograph recording for test evaluation on the onboard telemetry systems. The equipment associated with the telemetry ground station is as follows:

a. Antennas (mounted on the roof of the building).

b. PCM Ground Station.

c. SS/FM Ground Station.

d. FM/FM, FM, and PAM/FM Ground Stations.

e. Tape Recorders.

f. Oscillographs.

g. Calibration Equipment.

h. Telemetry Digitizing Equipment.

The Telemetry Digitizing Equipment will digitize the analog signals from the telemetry systems for input into the Ground Control Computer. The computer interfacing equipment provides for direct interrogation of the PCM or Digitizing equipment. The TM Ground Station is capable of receiving PCM and FM signals via an RF link from the IU. MSFC will decide if a separate TM Ground Station is required for the Saturn V complex.

5.2.4.7 Ground Control Computer Room

The Computer Room houses the Ground Control Computer system and its peripheral equipment. The computer provides a capability for program control, data processing and IU systems monitoring. The peripheral equipment provides the gating and conversion of data transmission between the computer and the IU. The Input/Output portion of the ground computer consists of paper tape readers, magnetic tape units, paper tape punch, line printer, CRT display unit, card reader, and card punch.

The computer will program all integrated testing and most subsystem testing. It interfaces with most of the test stations to control and monitor subsystem and system tests. Test results may be a hard copy output from the line printer, paper tape, or magnetic tape.

5.2.4.8 Instrumentation Checkout Room

The checkout equipment located in the Instrumentation Room consists of:

a. (Digital Data Acquisition System) DDAS Ground Station.

b. RACS Control Unit and Associated Display.

c. Special Instrumentation System Test Equipment.

d. Instrumentation Simulator.

e. Digital Automatic Checkout Equipment.

f. Remote Selector Indicator Unit.

The Instrumentation Checkout room contains the Digital Data Acquisition System ground station. This provides a coax link through the prelaunch phase to check out all instrumentation (except SS/FM and continuous channels of FM/FM and PAM/FM/FM). The DDAS Station is interfaced with the Ground Control Computer System for direct interrogation during systems and subsystems testing. The DDAS is also interfaced with the Control Room where there will be a meter display of critical measurements.

5.2.4.9 Navigation Systems Test Station

The Navigation Systems Test Station, under control of the Ground Control Computer System, processes the Guidance and Control Signals passing between the IU and the Ground Control Computer. The station consists of the following items:

5.2.4.9 (Continued)

a. Programmable Sequencer and Control Equipment.

b. Measurement Equipment.

c. Signal Output Equipment.

d. Flexible Interface Equipment.

During IU checkout, the Navigation Systems Test Station supplements the Ground Control Computer. All commands are executed under Ground Control Computer control to perform one of the following functions in the IU:

a. Select the signal to be measured.

b. Measure the signal or simulation of this signal by the stimuli generator.

c. Select a load condition.

5.2.4.10 RF Ground Station

The Common Radio Frequency Ground Station is used to perform the subsystems and systems testing of the IU Radio Frequency Systems. The RF room houses the test equipment necessary for individual RF systems control and monitoring. The RF Ground Station has no Ground Control Computer System interface. All tests are performed manually. Equipment located in the RF ground station consists of the following items.

a. Azusa Test Equipment.

b. Mistram Test Equipment.

c. C-Band Radar Test Equipment.

d. Radar Altimeter Test Equipment.

e. Command Receiver Test Equipment.

f. Antenna Checkout Equipment.

5.2.4.11 Component Acceptance Test Facilities

The Components acceptance test area, located in the IBM Operations building, is allocated a floor space of 6,657 sq ft. This area is divided into three sections which will be used to perform acceptance tests on components as follows:

Section One - Instrumentation Systems.

Section Two - Guidance and Control and Electrical Systems.

Section Three - Mechanical Systems.

5.3 MANUFACTURING PLAN

5.3.1 INT-20 Vehicle

Manufacturing plans for the INT-20 vehicle remain essentially the same as for the corresponding Saturn V stages.

5.3.2 S-1C Manufacturing Plan

The Manufacturing Plan for the First Stage of INT-20 remains the same as that of the S-1C outlined in Boeing Document D5-12561, "Boeing Manufacturing Plan for the S-1C Stage", with the exception of the changes identified below and in the INT-20 Stage design description is shown in Section 4.2.2.1.

5.3.2.1 Forward Skirt (60B14009)

No change is required for the adapter ring interface configuration. For the direct mating alternate configuration the forward skirt assembly sequence would not be changed. The only revision would be a modification of the bolt hole pattern at the interface of the S-1C and S-IV B stages. This revised pattern would contain only 130 of the present 216 S-1C holes plus 28 new 3/8 diameter holes.

A transfer template with the new hole locations and remaining S-1C hole locations would be supplied to Boeing Michoud by McDonnel-Douglas. Tooling revisions would consist of the addition of 28 bushings to the S-1C Forward skirt assembly fixture and the color coding of 130 of the present bolt locations on the Forward handling ring to indicate use with INT-20 stages. Proof loading of the ring would be performed with revised loads.

5.3.2.2 Oxidizer Tank (60B03101)

The center engine LOX standpipe assembly 60B41271-5 will be deleted. A ring will be added in place of the lower standpipe flange to support the cruciform in the same manner as the standpipe flange. In addition, the center LOX suction fitting will be capped off as shown in Method 2 of FIGURE 4.2.2.1-3 of the Stage Design description which is identical to the way it is currently capped off for hydrostatic test. The 23- inch diameter cover and floating flange as well as the cruciform support ring constitute the hardware additions to the oxidizer tank. They will also be produced using standard lathe turning and drilling methods. No tooling changes associated with the oxidizer tank are anticipated.

5.3.2.3 Intertank (60B29800)

No fabrication or tooling changes are anticipated in Intertank production.

5.3.2.4 Fuel Tank (60B25001)

a. Two flat 14-inch diameter cover plates will be fabricated and used with present seals and fasteners to cap off the inboard fuel suction elbows as shown in Figure 4.2.2.1-4 of the Stage Design Description.

5.3.2.4 Fuel Tank (Continued)

b. The thickness of the forward area of the aft fuel base gores will be increased by modification of the Numerical Control tape which produces the scallop pattern in the gore prior to bulge forming. A slight amount of development in the bulge forming of the new thickness may be required but is not expected to constitute a significant problem since the new configuration is within the thickness range currently being formed.

c. The 28-inch non-structural, non-sealing tunnel cover shown in FIGURE 4.2.2.1-4 of the Stage Design Description will be machined and fastened to the upper end of the inboard LOX Tunnel.

d. The new upper fuel instrumentation cover shown in FIGURE 4.2.2.1-6 of the Stage Design Description will be fabricated and installed in place of the existing similar 60B24510-3 cover.

e. Tooling Modifications

   The thickness of about half the buttons in three vacuum chuck base gore support blankets will be machined down to locate against the new membrane thickness during gore and bulkhead assembly.

5.3.2.5 Thrust Structure (60B18054)

The fabrication and installation of the center engine support struts, strut insulation, fittings, attach hardware and center engine adapter fitting will be deleted. These components are shown in FIGURE 4.2.2.1-7 of the Stage Design Description. To facilitate reversibility from an INT-20 to an S-1C it is anticipated that holes for the precise location of the center engine adapter fitting will be drilled during INT-20 thrust structure buildup. Fabrication and installation of the eight Inboard Fuel Suction Duct Support Links 60B19769-1 will be deleted.

The manufacturing sequence is not affected by the number of slow release devices to be installed on the vehicle at KSC.

5.3.2.6 Heat Shield Installation (60B20800)

Changes to the heat shield installation consist of deleting the center engine heat shield penetrations. This is accomplished by deletion of the relatively small panels with special cut-outs for the center engine and filling the area with the large standard panels of the same design as adjacent areas. The installation is simplified since fewer panels are required. Six (6) additional 60B20210 honeycomb flight panels and six (6) additional steel static firing panels will replace sixteen (16) existing honeycomb flight and sixteen (16) existing steel static firing panels. The basic tooling will not require change other than the color coding of drill bushings so that the grid structure tooling can be used for the four engine vehicle.

5.3.2.6 Heat Shield Installation (60B20800) - Continued

The center area heat shield support structure will be omitted and replaced with the simple square grid using beams and bracketry of existing design as shown in FIGURE 4.2.2.1-11 of the Stage Design Description.

The new inconel bracket shown in FIGURE 4.2.2.1-12 of the Stage Design Description will be fabricated and installed using existing hole locations. One roll-away access panel is required for static firing and its configuration is unchanged.

The exact amount of refurbishment after static firing must be determined by actual inspection.

5.3.2.7 Propulsion and Mechanical Subsystems

a. Oxidizer system

The components listed herein should be considered to require irridite of all aluminum surfaces and LOX cleanliness of all LOX or GOX wetted surfaces.

1. Oxidizer fill and drain (60B41012)

No changes are required to this system.

2. Oxidizer feed system (60B41014)

Installation of the Inboard LOX suction duct, LOX prevalve and PVC duct will be omitted as indicated in the shaded portion of FIGURE 4.2.2.1-14 and itemized in Section 3.2.3 of Appendix A of the Stage Design Description.

In order to support the LOX interconnect spool after deletion of the inboard LOX lines, it will be necessary to fabricate the 26.8-inch long 19.5-inch diameter spool assembly and flanges shown in FIGURE 4.2.2.1-15. Fabrication of this spool assembly will consist of machining the ends and outside only of a 20-inch O.D. 16.75-inch I.D. 27.5-inch long purchased 2219 Aluminum rolled ring forging. Four LOX cutoff sensors will be deleted and the bosses plugged in lines to engines 2, 4 and 5. Two LOX cutoff sensors will be added in existing bosses in lines to engines 1 and 3. No new tooling is required.

3. LOX interconnect system (60B41014)

Engine position 2 Interconnect Valve 60B41136-3 will not be installed and the new Interconnect spool shown in FIGURE 4.2.2.1-18 of the Stage Design Description will be fabricated and installed in its place. A temperature transducer will be installed in an existing boss in the center LOX interconnect spool. No new tooling is required.

5.3.2.7 a. (Continued)

4. Lox bubbling system (60B41221)

   The two small tube assemblies, union, adapter, check valve and union orifice listed in Section 3.2.6 of the Appendix A will be deleted as shown in FIGURE 4.2.2.1-19. The Tee will be capped and the spool boss plugged with the listed standards.

5. Oxidizer pressurization (60B51400)

   The GOX return duct, tube assembly 60B51404-1 and -5 support bolt on bracketry running from the center engine to the GOX manifold will not be installed. This item is shown in FIGURE 4.2.2.1-20 of the Stage Design Definition. A cover plate, Item 1 of FIGURE 4.2.2.1-21 will be machined and installed to cap off the center engine port of the GOX manifold. Pressure switches will be replaced.

b. Fuel System

The components listed herein should be considered to have all aluminum surfaces irridited and all Fuel or Fuel pressurization wetted surfaces cleaned for fuel service.

1. Fuel fill and drain (60B43014)

   The fuel loading probe 60B43006-25 will be lengthened by 14 inches as shown in FIGURE 4.2.2.1-22 and 4.2.2.1-23 of the Stage Design Definition. The longer probe can still be installed within the clearance of the Intertank if it becomes necessary to remove the probe for any reason.

2. Fuel feed system (60B43014)

   The inboard fuel feed system hardware aft of the inboard fuel suction fittings will not be installed. The 2 prevalves, 2 suction ducts and 2 fuel PVC ducts listed in Section 3.2.10 of Appendix A will not be installed.

   FIGURE 4.2.2.1-24 illustrates the items deleted. The capping off of the suction fittings is discussed in the fuel tank portion of this plan. Deletion of support bracketry for this system is shown in the thrust structure modification. No additional tooling is required.

5.3.2.7 b. (Continued)

3. Fuel pressurization system (60B49600)

The helium supply and return ducts 60B49022-1 and -3 will not be installed as indicated in FIGURE 4.2.2.1-25 of the S-IC Design Description. The inboard engine branch of the helium supply and return manifolds will be capped using the new cover plates identified as items 2 and 3 of FIGURE 4.2.2.1-21. The duct bolt on bracketry will be deleted. The orifice plates and pressure switches will be revised per Section 4.2.2.1.b.2.6 of the Engineering Design Description.

c. Auxiliary systems

1. Control pressure system (60B52500)

The lines, fittings and solenoid valves which are used to supply control pressure to the inboard fuel prevalves will not be installed. These 18 small items with a total weight of 5.5 pounds are listed in Section 3.2.13 of Appendix A with the two standards used to plug the system. FIGURES 4.2.2.1-26 and 4.2.2.1-27 illustrate the deleted lines.

2. Environmental Control System

No changes are required to this system.

3. Turbopump oxidizer seal (60B37601)

The turbopump oxidizer seal line to the center engine will be deleted as shown in FIGURE 4.2.2.1-28 of the Stage Design Description. The nine deleted components with a total weight of 7.2 pounds and the standard used to plug the system are listed in Section 3.2.16, Appendix A.

4. Radiation calorimeter purge

This system is expected to be used on the first two flight INT-20's only and will be located in the base heat shield. The installation is illustrated in FIGURE 4.2.2.1-29 of the S-IC Design Description and the items added are listed in Section 3.2.17 of Appendix A. The line itself is expected to be a bent tube weighing approximately 0.9 pounds. It should be noted that some parts of the turbo pump oxidizer seal system which were deleted in the paragraph above are listed as additions for the first two flight stages only in Section 3.2.17 of Appendix A.

5.3.2.7 c. 4. (Continued)

The net effect is that the items listed in Section 3.2.16 as being deleted and then listed again in 3.2.17 as additions are retained per S-1C configuration on the two first flight INT-20's and all items in both lists with the exception of the MC 238C8W plug are actual deletions thereafter.

5. LOX dome and gas generator LOX injector purge (60B37600)

The center engine branch of this system consisting of 7 plumbing items weighing a total of 11.2 pounds will be deleted and the branch plugged with a standard as shown in FIGURE 4.2.2.1-30 of the Stage Design Definition and itemized in Section 3.2.18 of Appendix A.

6. Engine cocoon thermal conditioning purge (60B37602)

The line to the center engine, consisting of 7 plumbing items weighing a total of 7.7 pounds is deleted and the branch will be plugged with a standard as illustrated in FIGURE 4.2.2.1-30 of the Stage Design Definition and itemized in Section 3.2.19 of Appendix A.

7. Thrust OK checkout system (60B37600)

The center engine branch line will be omitted as shown in Figure 4.2.2.1-31 of the Stage Design Definition eliminating 9 plumbing items weighing a total of 1 pound and the branch plugged with the standard listed in Section 3.2.20 of Appendix A.

8. Thrust chamber prefill system (60B37550)

The center engine branch consisting of 5 plumbing items weighing a total of 5.3 pounds will be deleted and the branch plugged with the standard listed in Section 3.2.21 of Appendix A . The deletions are illustrated in Figure 4.2.2.1-31.

9. POGO suppression system (60B41840)

The line supplying helium to the center engine LOX prevalve will be deleted. The line consists of the 12 plumbing items weighing 4.7 pounds listed in Appendix A, Section 3.2.22. The branch will be plugged with the listed standard. Figure 4.2.2.1-33 of the S-IC Design Description illustrates this change.

5.3.2.7 d. Flight control subsystem

1. Fluid power subsystem (60B82000)

The center engine ground hydraulic supply and return lines will be deleted as shown in FIGURE 4.2.2.1-34 of the Stage Design Description and the system will be capped using two of the new flanges illustrated as Item 4 of FIGURE 4.2.2.1-21. One hundred fourteen (114) items of plumbing, support bracketry and standards weighing a total of 51.6 pounds are eliminated as itemized in Section 3.2.24 of Appendix A.

2. Thrust vector control system (60B84000)

No changes are required to this system.

e. Engine and related components (60B37450)

The center F-1 engine, support struts and attach hardware are deleted on the INT-20. Section 3.2.26 of Appendix A lists the 49 individual items weighing 19,272 pounds.

5.3.2.8 Electrical/Electronic Equipment

Stage Electrical/Electronic equipment consists of cabling, equipment panels and telemetry. Major telemetry assemblies will be procured from approved commercial sources. All level IV testing will be accomplished utilizing existing facilities. Cable assemblies will be fabricated in the electrical fabrication area and installed while the vehicle is in the final assembly position. The equipment panels will be fabricated utilizing existing honeycomb techniques and facilities. Major equipment panel assemblies will be completed with existing tooling. The electrical distributors will be fabricated. An installation sequence will be developed.

5.3.2.9 Stage Instrumentation

The stage instrumentation consists of strain gages, calorimeter, flow rate, pressure, temperature and vibration transducers and related amplifiers. Instrumentation as defined by a specification control drawing will be procured from approved commercial sources. Minor assemblies and testing will be accomplished utilizing existing facilities. Printed circuit assemblies will be fabricated and installed in the purchased amplifier modules. The measuring rack castings will be purchased and the major assemblies completed in house by installing wiring, connectors and amplifier assemblies. Installation of instrumentation will be accomplished while the vehicle is in the final assembly position. An installation sequence will be developed.

5.3.2.10 Conclusions

a. No significant problems are anticipated in the manufacture of the First Stage INT-20.

b. Modifications to existing tooling will be extremely small.

c. No development of new manufacturing techniques or skills will be required.

d. No problems are forseen resulting from concurrent production of S-1C and INT-20.

e. Manufacturing flow time for INT-20 will probably turn out to be about 2 weeks shorter than S-1C but is considered to be the same for simplicity in the relatively long range projections of this study.

f. Assuming all parts are available, a particular vehicle could be changed from INT-20 to S-1C or vice-versa with very little inconvenience, expense or disruption of flow, up until the point where the oxidizer tank is lowered onto the intertank. At this point the normal sequence of lowering the inboard LOX duct into the inboard LOX tunnel with the overhead crane in the vertical assembly position is blocked by the LOX tank.

5.3.3 S-IVB Stage Manufacturing Plan

5.3.3.1 Flow Plan

The manufacturing plan is based on the assumption that no significant redesign will be required due to expected higher acoustic levels of the INT-20 environment.

The S-IVB manufacturing sequence depicted in Figure 5.3.3-1 has been revised to include a legend identifying those processes affected by S-IVB stage modifications for the INT-20. This legend is coordinated to a typical Position Flow Chart, as shown on Figure 5.3.3-2, to reflect the manufacturing area where the rework or deletion will be made. The Position Flow Chart identifies where each specific item is deleted in the in-line manufacture of the INT-20/S-IVB stage baseline or alternate configuration.

Planning paper will be changed to incorporate the deletions of the selected option. New planning paper will be instituted for components added. New auxiliary tooling will be used if needed to incorporate additions or deletions of components. Present S-IVB Saturn Manufacturing capability will not be affected.

5.3.3.2 Tooling Requirements

No new tooling is anticipated for the S-IVB stage modifications; however, both design options for adaptation of the interstage to the S-IC interface (see Section 4.2.3) require some new tooling. The tooling requirements for the interface bolt hole patterns common to the S-IVB and the S-IC are delineated under each of the two design options, the revised bolt hole pattern, and the new interface adapter ring. These design options and the method of manufacture are depicted in Figure 5.3.3-3.

a. Option No. 1 - Direct Interface (New Bolt Hole Pattern)

   For interface option No. 1, the existing bolt pattern will be modified as follows:

   1. Use 130 1/2-inch bolt holes on 197.17-inch radius - present location.

   2. Add 10 3/8-in dia. bolt holes on 197.17-inch radius.

   3. Add 18 3/8-inch dia. bolt holes on 196.875-inch radius.

   The tooling requirements for option No. 1 will be as follows:

   1. Make new control master as follows:

      (a) Locate 130 1/2-inch dia. bolt holes on 197.17-inch radius - present location per Boeing transfer gage presently located at North American Rockwell Corporation, Seal Beach, Calif.

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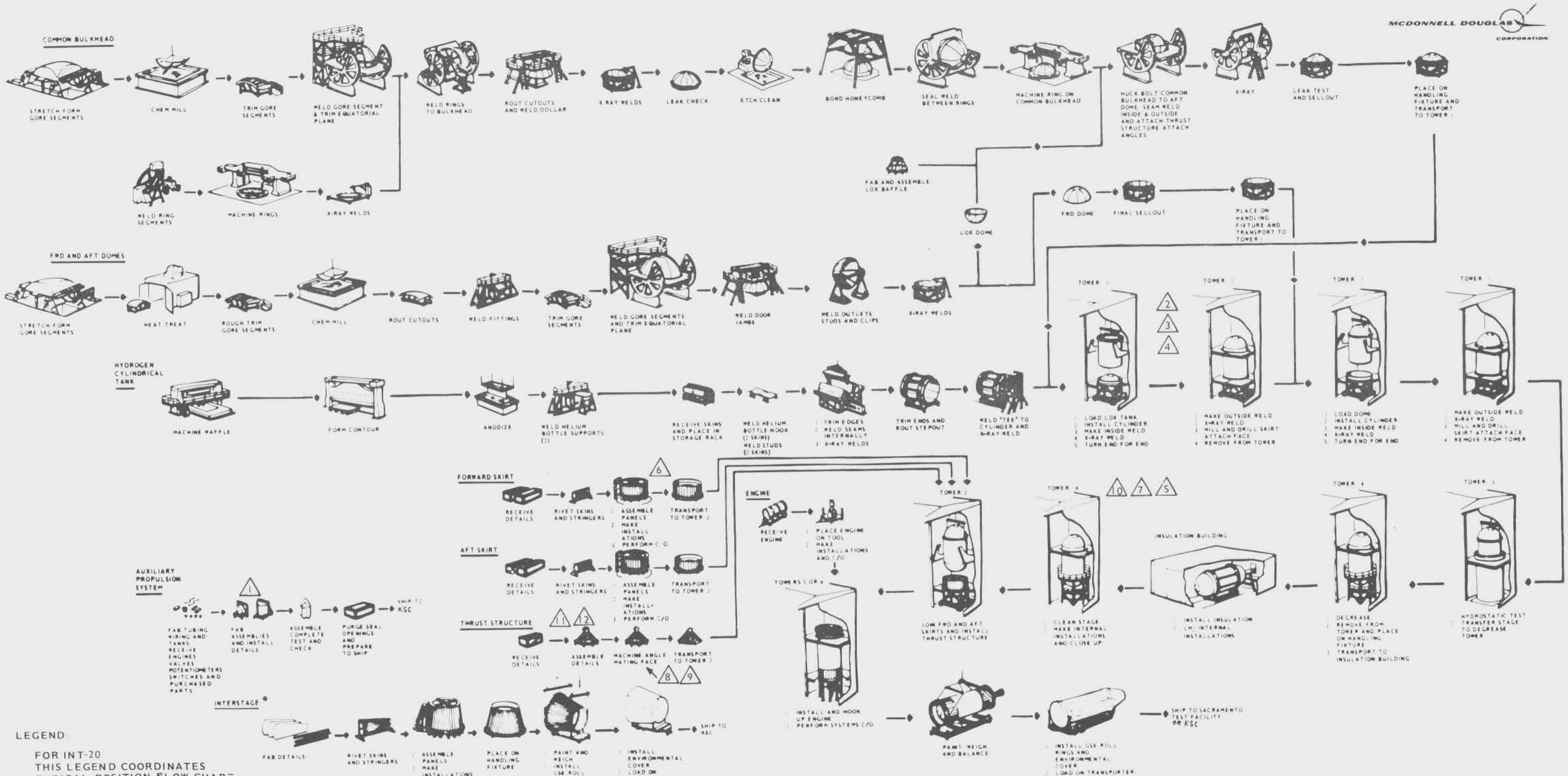
CORPORATION



LEGEND

FOR INT-20
THIS LEGEND COORDINATES TYPICAL POSITION FLOW CHART DELETION (REF. FIGURE 5.3.3-2) TO MFG. SEQUENCE

DELTA	POSITION INDICATED	DELTA	POSITION INDICATED
1	05AP	7	22TA
2	17SA	8	22TS
3	19SA	9	23TS
4	20SA	10	24TA
5	21TA	11	24TS
6	22FS	12	26TS

* INT 20 INTERSTAGE MODIFICATION. SEE FIGURE 5.3.3.-4

Figure 5.3.3-1. INT-20/S-IVB Stage and Interstage Manufacturing Sequence

ONLY AREAS AFFECTED BY INT-20 SIVB VARIATIONS SHOWN

POSITIONS

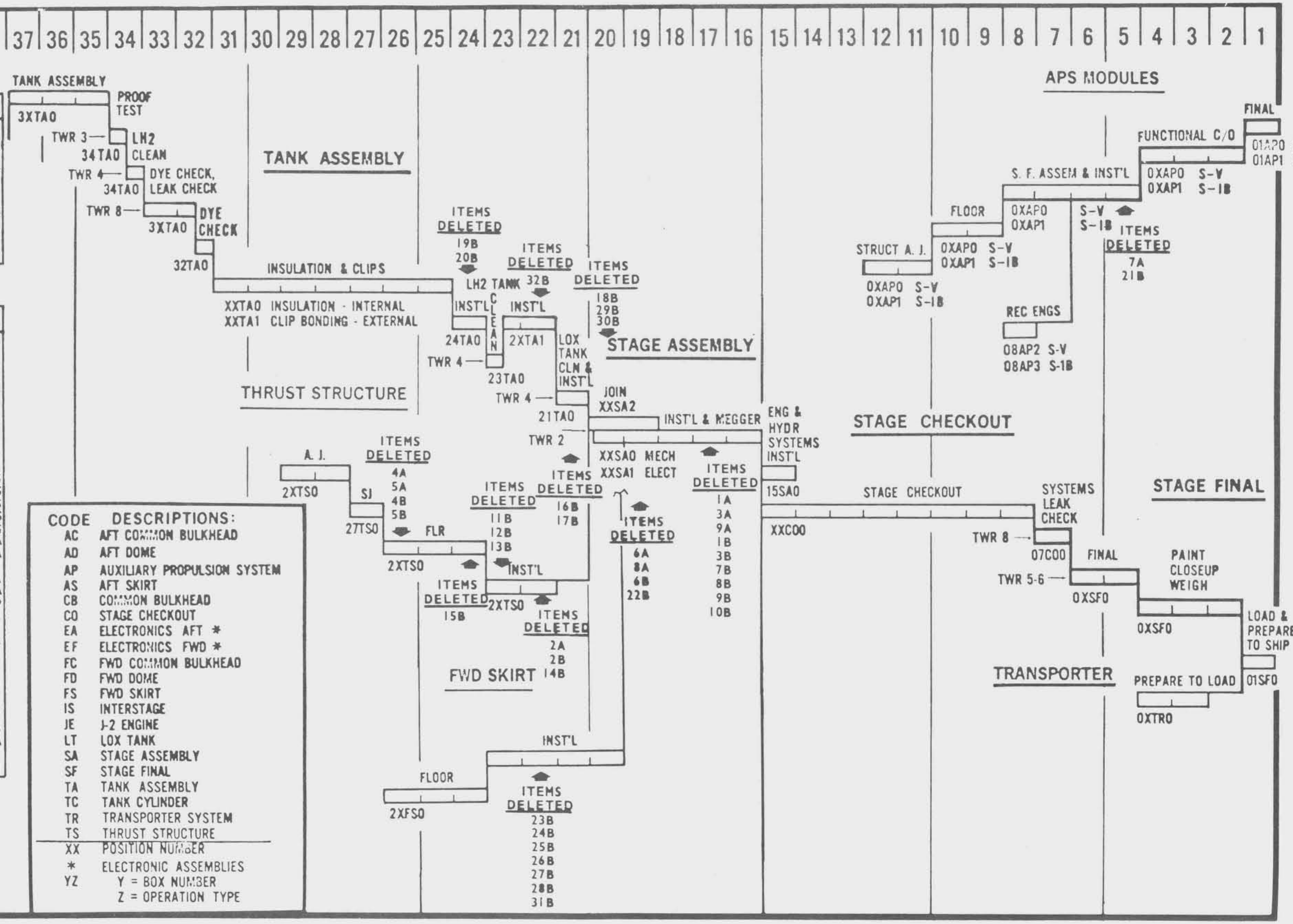
45 44 43 42 41 40 39 38 37 36 35 34 33 32 31 30 29 28 27 26 25 24 23 22 21 20 19 18 17 16 15 14 13 12 11 10 9 8 7 6 5 4 3 2 1

ITEMS DELETED BASELINE

ITEM NO.	PART NUMBER	QTY	POS
1A	1B66868-501	7	17SA
2A	1B56653-513	3	22TS
3A	1B27629-501	7	17SA
4A	1B27632-1	14	26TS
5A	1B27632-2	14	26TS
6A	1B62600-507	1	19SA
7A	15-210001	2	05AP
8A	1A49985-1	1	19SA
9A	DELETE TRANSDUCERS	—	17SA

ITEMS DELETED ALTERNATE

ITEM NO.	PART NUMBER	QTY	POS
1B	1B66868-501	7	17SA
2B	1B56653-513	3	22TS
3B	1B27629-501	7	17SA
4B	1B27632-1	14	26TS
5B	1B27632-2	14	26TS
6B	1B62600-507	1	19SA
7B	1B59005	1	17SA
8B	1B59005-1	1	17SA
9B	1B59009-1	1	17SA
10B	1B59010-1	2	17SA
11B	1B62778-1	2	23TS
12B	1B43659-1	2	23TS
13B	1B40824-1	2	23TS
14B	1B52624	2	22TS
15B	1B66692-1	3	24TS
16B	59021-1-200A	2	21TA
17B	55666-200A1	2	21TA
18B	MVC61772-200A	2	20SA
19B	1A48858-1	3	24TA
20B	1B39297-1	3	24TA
21B	15-210001	2	05AP
22B	1A49985	1	19SA
23B	1A67193	1	22FS
24B	1B53036-1	1	22FS
25B	1B44572	1	22FS
26B	1B44573	2	22FS
27B	1B44574	2	22FS
28B	1B44575-1	2	22FS
29B	1B44557-1	2	20SA
30B	1B56394-1	2	20SA
31B	5145-1	1	22FS
32B	1B56518-1	1	22TA
33B	DELETE TRANSDUCERS	—	17SA



CODE	DESCRIPTIONS:
AC	AFT COMMON BULKHEAD
AD	AFT DOME
AP	AUXILIARY PROPULSION SYSTEM
AS	AFT SKIRT
CB	COMMON BULKHEAD
CO	STAGE CHECKOUT
EA	ELECTRONICS AFT *
EF	ELECTRONICS FWD *
FC	FWD COMMON BULKHEAD
FD	FWD DOME
FS	FWD SKIRT
IS	INTERSTAGE
JE	J-2 ENGINE
LT	LOX TANK
SA	STAGE ASSEMBLY
SF	STAGE FINAL
TA	TANK ASSEMBLY
TC	TANK CYLINDER
TR	TRANSPORTER SYSTEM
TS	THRUST STRUCTURE
XX	POSITION NUMBER
*	ELECTRONIC ASSEMBLIES
YZ	Y = BOX NUMBER Z = OPERATION TYPE

Figure 5.3.3-2. Typical S-IVB Stage Position Flow Chart

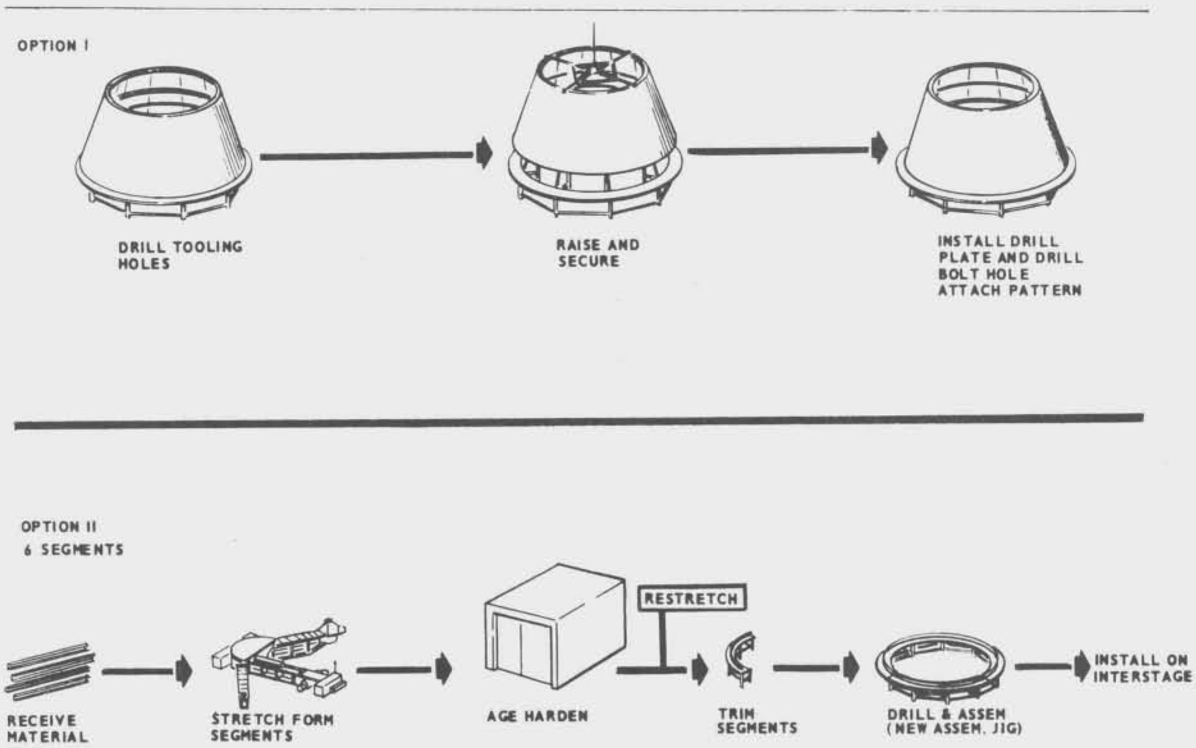


Figure 5.3.3-3. Manufacturing Methods, Interstage Options

(b) Add 10 3/8-inch dia. bolt holes on 197. 17-inch radius.

(c) Add 18 3/8-inch dia. bolt holes on 196. 875-inch radius.

2. Make two new transfer gages, one for MDAC-WD and one for Boeing.

3. Make new drill plate and riser blocks.

b. Option No. 2 - New Adapter Ring

For interface option No. 2, design and build a 5-inch deep adapter ring which employs existing bolt hole pattern of both stages. Manufacture would be as follows:

1. Fabricate a segmented 5-inch deep adapter ring channel. Assemble segments into ring.

2. Drill existing S-IVB interface bolt hole attach pattern in the upper leg of the channel.

3. Drill existing S-IC interface bolt hole attach pattern in the lower leg of the channel.

4. Bolt drilled ring to interstage.

The tooling requirements for option No. 2 will be as follows:

1. Make new stretch form die to form channel adapter ring segments to approximately a 16-1/2 ft ring.

2. Make new trim fixture to trim ends.

3. Make new assembly/drill jig. Drill holes will be established per the existing transfer gages. The S-IVB transfer gage and the S-IC transfer gage are presently located at North American Rockwell Corporation, Seal Beach, California.

4. Make two new mill fixtures for alignment bracket.

5. Make one new drill jig for alignment bracket.

6. Make two new mill fixtures for adapter ring splice plates.

5.3.3.3 Interface Options Tooling/Cost Trade

A trade-off study of the tooling requirements for option No. 1 and option No. 2 was conducted to evaluate both options. The criteria considered in the evaluation were:

1. Retention of the integrity of present S-IVB Saturn Manufacturing capability.

2. Economics.

   (a) Recurring Costs.

   (b) Non-recurring Costs.

3. Logistics.

4. "Fool Proof" manufacturing approach.

Table 5.3.3-I presents the evaluation results. Based on these results, the cost trade-off data presented in Section 5.6.3, and the compatibility for potential retrofit, interface option No. 2, the new attach ring, was selected as the recommended approach.

5.3.3.4 Manufacturing Schedule

The schedule requirements in terms of months from ATP for new and existing fabrication, procurement, assembly, planning and tooling is depicted on Figure 5.3.3-4. It was assumed that long lead time raw material procurement authorization precedes ATP (month 0) by six months. Planning, tooling, and new fabrication requirements are minor and the INT-20/S-IVB stage fabrication and assembly time is essentially the same as for a Saturn V/S-IVB stage. The schedule also indicates the post-manufacturing acceptance checkout and final checkout operations.

Table 5.3.3-I

TOOLING/COST TRADE RESULTS S-IC/S-IVB INTERFACE

Criteria	Option No. 1 Direct Interface (New hole pattern)	Option No. 2 New Adapter Ring
Retain S-IVB Manufacturing	Interferes with present procedures in-line. Concurrent Saturn V production would require intermittent drill plate changes	Off-line operation except for final bolting finished ring in place (no drill plate changes)
Non-Recurring Costs*	Higher due to need to make new control master and transfer gages	Lower since control master and gages exist and other items minor or comparable
Recurring Costs	Lower - except for set up change-over, costs would be essentially same as existing operation	Higher - new item; however cost difference dependent on quantity produced - low quantity total costs can be lower
Logistics	Similar	Similar
"Fool Proof" Manufacturing Approach	Mixed Saturn V and INT-20 operations potential error source	INT-20 operation only - decreases chance of error

*See Section 5.6.3 - S-IVB cost plan for specific costs.

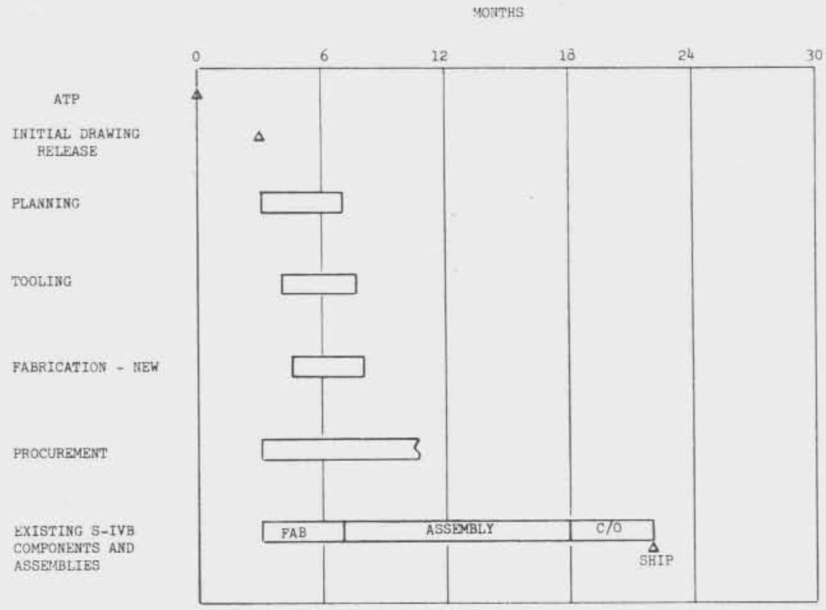


Figure 5.3.3-4. INT-20/S-IVB Manufacturing Plan Schedule

5.3.4 IU Manufacturing Plan

The IU Manufacturing effort can be described in terms of three distinct phases: Fabrication, Assembly, and Preparation for Shipment. Nominal time periods for the completion of each phase are as follows:

Fabrication - 10 weeks including two weeks for receiving inspection of the IU structures segments. This phase involves alignment and splicing of the segment assemblies into an IU structure assembly; painting of the IU structure assembly; and drilling, routing and potting of cutouts for subsequent systems hardware installation.

Assembly - 12 weeks. This phase involves IU structures assembly, alignment and installation of cables, cable tray assembly, thermal conditioning panels and all other component hardware, with the necessary brackets, clamps, tubing and fasteners. All functional component end items are acceptance tested during this phase of the manufacturing operation. All systems are installed in readiness for IU systems checkout at the end of the assembly phase.

Preparation for Shipment - two weeks. This phase is accomplished subsequent to completion of IU systems checkout which requires eight weeks. It involves the removal and packaging of selected flight hardware components and assemblies for separate shipment and otherwise securing the IU stage for shipment.

The Manufacturing effort, generally described above, is controlled by manufacturing routings which outline, step-by-step, the procedure to accomplish all the discrete operations required, including the essential inspections.

5.3.4.1 Tooling

There are no new tooling or fixture requirements for the manufacture of IU assemblies for the INT-20 vehicles. Configuration variations between IU's for the INT-20 vehicle or Saturn V configuration can be handled by the issuance of separate sets of manufacturing instructions (routings) which are unique to a particular IU. In effect, there would be no essential differences from the manner of manufacturing Saturn V on the current program. Further, the nominal times for each of the manufacturing phases, including systems checkout, would be the same.

5.3.4.2 IU Facilities Plan

Existing facilities are designed to satisfy the broad mission requirements of the current program. A single building houses Manufacturing and Test facilities for

5.3.4.2 (Continued)

component acceptance testing, IU fabrication, assembly, systems checkout and packaging for shipment. A Hi-Bay area houses two IU fabrication stations, three assembly stations and one Saturn V IU checkout station. A section of the building contains a component test complex and manufacturing support area. The building also contains office space for the related manufacturing and test support areas. See Figure 5.1.4-1 for modifications to facility Ground Support Equipment required to support a six per year manufacturing schedule.

5.4 FACILITIES PLAN

The Facilities Plan considers new or modified facilities that are needed for the INT-20 vehicle or stages.

5.4.1 Stage Facilities

Design, manufacturing and testing facilities of the present Saturn V program are suitable and adequate for the INT-20 S-IC and S-IVB stages. Present Instrument Unit facilities have a capacity for 10 IU's per year of which five are for Saturn V and five are for Saturn IB. In order to produce six or more IU's per year for Saturn Vs or INT-20s two alternatives exist; either add people to the Saturn V/ INT-20 IU production line by overtime or a second shift, or modify the Saturn IB IU production to produce and check-out INT-20 IU's and Saturn V IU's in addition to Saturn IB IU's. The Saturn IB line modification would provide the capability to produce 10 Saturn V or INT-20 IU's total per year and also retain the capability to produce five Saturn IB's per year. The first alternative would add to the recurring cost of each Saturn V or INT-20 IU over five per year. The second alternative would add $700, 000 to the non-recurring development cost of the INT-20.

5.4.2 KSC Launch Facilities

KSC Launch Facilities must be modified to accept the shorter two-stage INT-20 vehicle. The S-IC stage of the INT-20 fits KSC facilities, but swing arms, platforms, service connections, etc. must be moved downward to the new lower positions of the S-IV stage, the Instrument Unit and the payload. KSC launch facilities were the subject of a separate study. The study, "KSC Facilities and Operations for Saturn MS-IC/MS-IVB (Intermediate-20) Launch Vehicle" (Reference 5.4-1) was conducted by The Boeing Company, Atlantic Test Center, under Contract NAS10-6163. Technical direction and guidance was furnished by the NASA Future Studies Office, John F. Kennedy Space Center. The study is a technical and economic analysis of the impact on complex 39 at the Kennedy Space Center when processing and launching an Intermediate-20 Launch Vehicle. Study results were presented in three volumes:

D5-16785-1, Executive Summary Report
D5-16785-2, Final Technical Report
D5-16785-3, Appendices

The most feasible and economical launch facility modification option would be to modify one complete set of launch facilities to accommodate the INT-20 vehicles and, yet be convertable to a Saturn V configuration when necessary. Such a modification is called the "existing facility, convertable for Saturn V or INT-20". The modification of one Launch Umbilical (LUT), one Mobile Service Structure (MSS), one Launch Control Center (LCC), one VAB High Bay and one Launch Pad for convertable use of the Saturn V or the INT-20 with MLV payload would cost about

5.4.2 (Continued)

$3.2 million. The conversion of these facilities from INT-20 configuration to Saturn V configuration would not be necessary, except for the single Mobile Service Structure; i.e., after the initial modification, the modified LUT, LCC and VAB would remain in the INT-20 configuration. Since there is only one MSS, it must be converted between Saturn V configuration and INT-20 configuration and vice versa according to which launch vehicle is to be launched next. The cost to change the MSS from Saturn V configuration to the INT-20 configuration is $90,200 and the cost to return to the Saturn V configuration is $98,500. The total cost to change from Saturn V to INT-20 and back to Saturn V is $188,700.

The time needed for initial facility modification is 315 working days (8 hours per day) or a total elapsed time of about 15 months. Of the 315 working days, only the last 87 days would be "down time", i.e., facilities "out of commission". The time for structure from Saturn V to INT-20 configuration is 27 working days and from INT-20 to Saturn V configuration is 32 working days (i.e., about 30 working days each way or an elapsed time of 6 weeks.)

The annual Facility Utilization Schedule plan for each Saturn V/INT-20 program is shown on Figures 5.4.2-1 through 5.4.2-3.

General information on the facility modifications and conversions discussed above is given in excerpts from the "KSC Facilities and Operations for Saturn MS-IC/MS-IVB (Intermediate-20) Launch Vehicle, Final Technical Report, D5-16785-2". (Reference 5.4-1). The excerpts follow:

"The facilities at Launch Complex 39 can be modified in various ways to support the checkout and launch of the INT-20 Launch Vehicle. Each of these resulting configurations was examined from the standpoints of technical feasibility and cost effectiveness. The most attractive were then combined into overall "processing concepts" to determine their suitability for the total vehicle processing operation at LC-39. The main body of this report describes and evaluates the various methods of satisfying the checkout and launch requirements.

The primary study objective was to provide LC-39 impact data for the INT-20 vehicle from receipt of hardware through post-launch refurbishment. Specifically, the study accomplished the following:

a) The identification and description of existing, new and/or modified LC-39 facilities and equipment which best satisfy requirements for checkout and launch of both Saturn V and INT-20 vehicles.

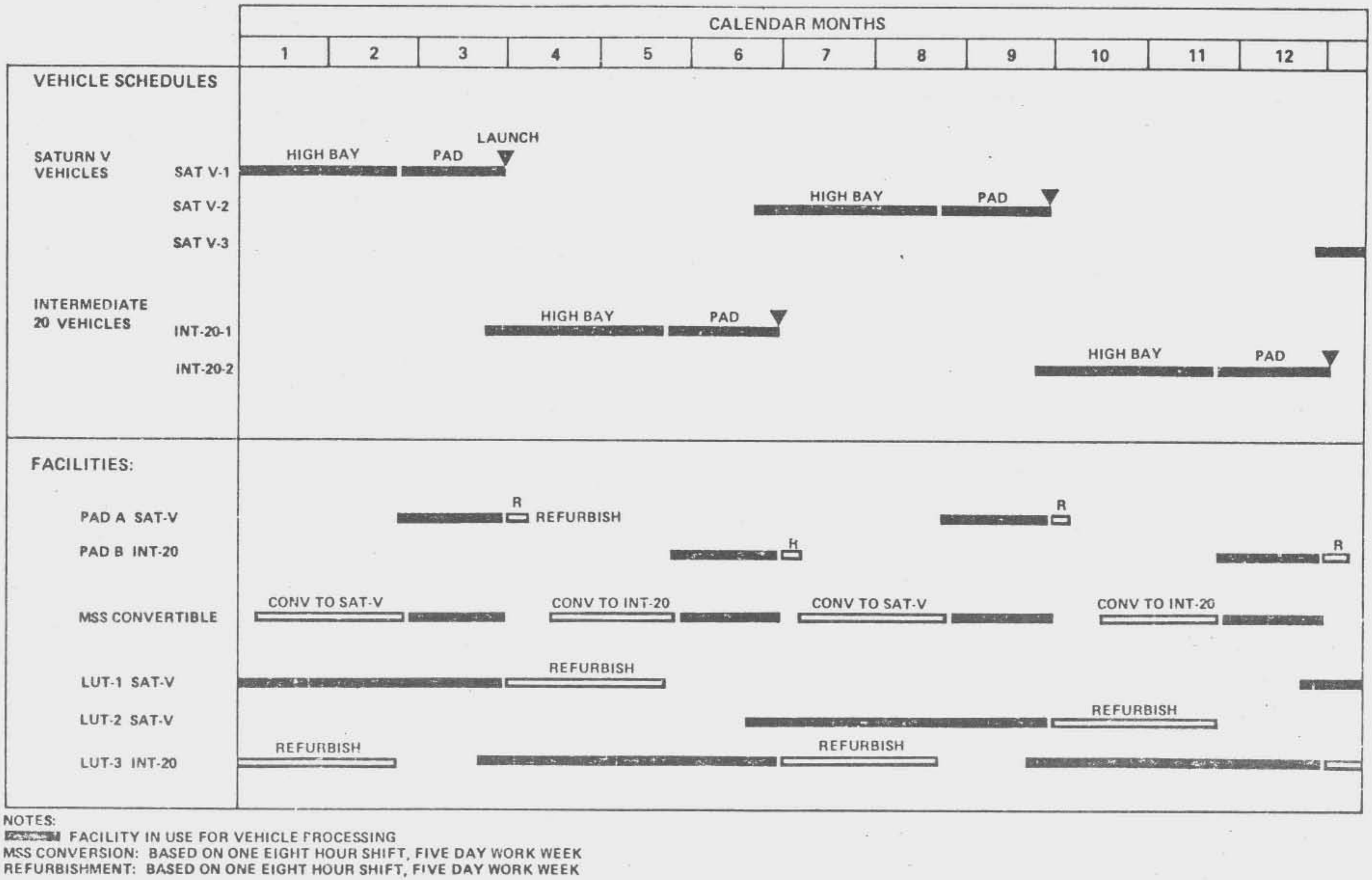


FIGURE 5.4.2-1 FACILITY UTILIZATION SCHEDULE - TWO SATURN V & TWO INT-20 LAUNCHES/YEAR

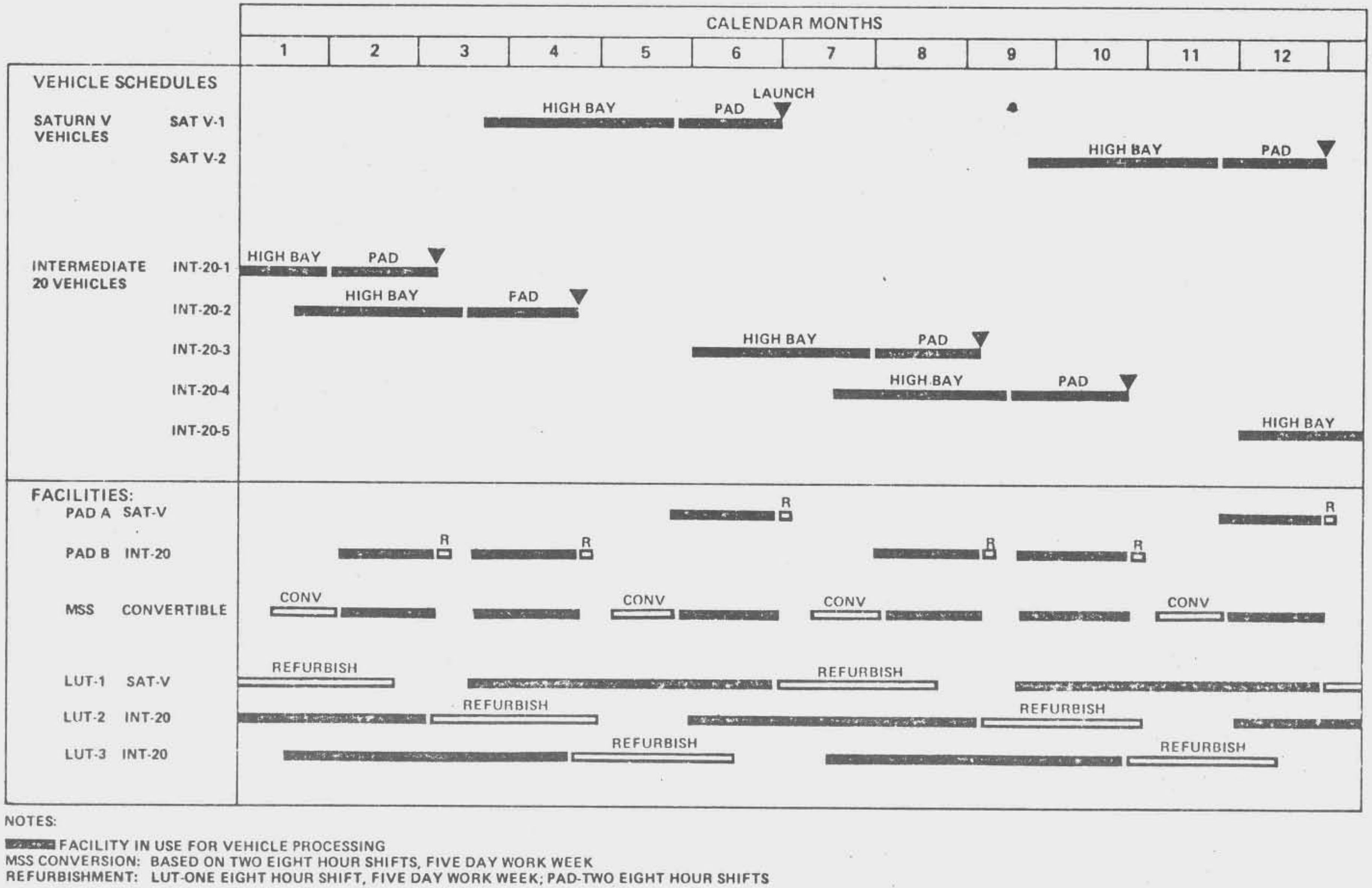
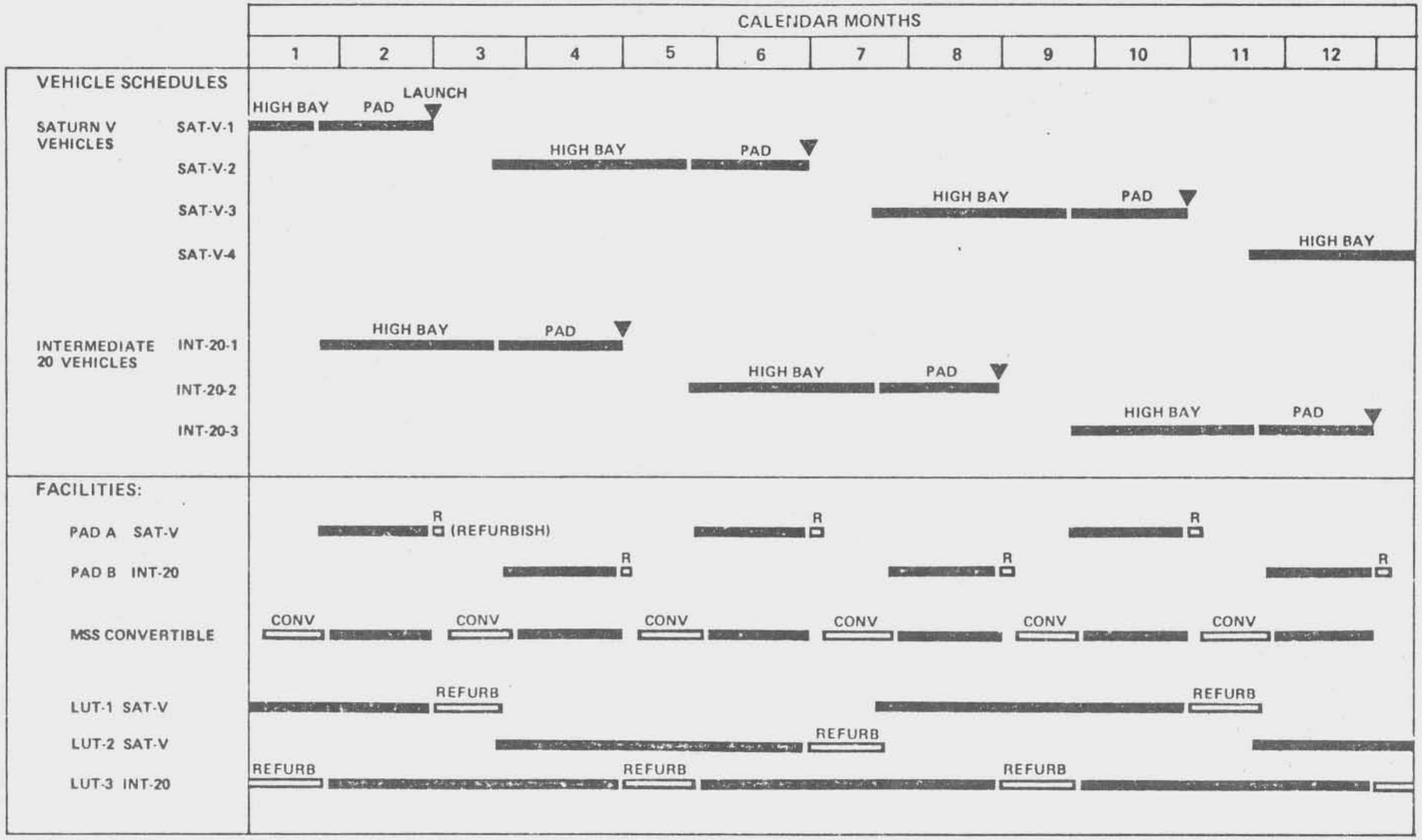


FIGURE 5.4.2-2 FACILITY UTILIZATION SCHEDULE - TWO SATURN V & FOUR INT-20 LAUNCHES/YEAR



NOTES:
FACILITY IN USE FOR VEHICLE PROCESSING
MSS CONVERSION: BASED ON TWO EIGHT HOUR SHIFTS, FIVE DAY WORK WEEK
REFURBISHMENT: BASED ON TWO EIGHT HOUR SHIFTS, SIX DAY WORK WEEK

FIGURE 5.4.2-3 FACILITY UTILIZATION SCHEDULE - THREE SATURN V & THREE INT-20 LAUNCHES/YEAR

5.4.2 (Continued)

b) Schedules, manpower, and cost estimates for the design, construction, and activation phases of each facility configuration.

c) The definition of feasible processing concepts, formulated from the various facility configurations to support the entire checkout flow of an INT-20 vehicle.

d) Schedules, manpower, and cost estimates for converting facilities and equipment from Saturn V to INT-20 and back to Saturn V for each convertible concept, including impact on vehicle processing operations.

The following guidelines and assumptions were adhered to throughout the course of the study.

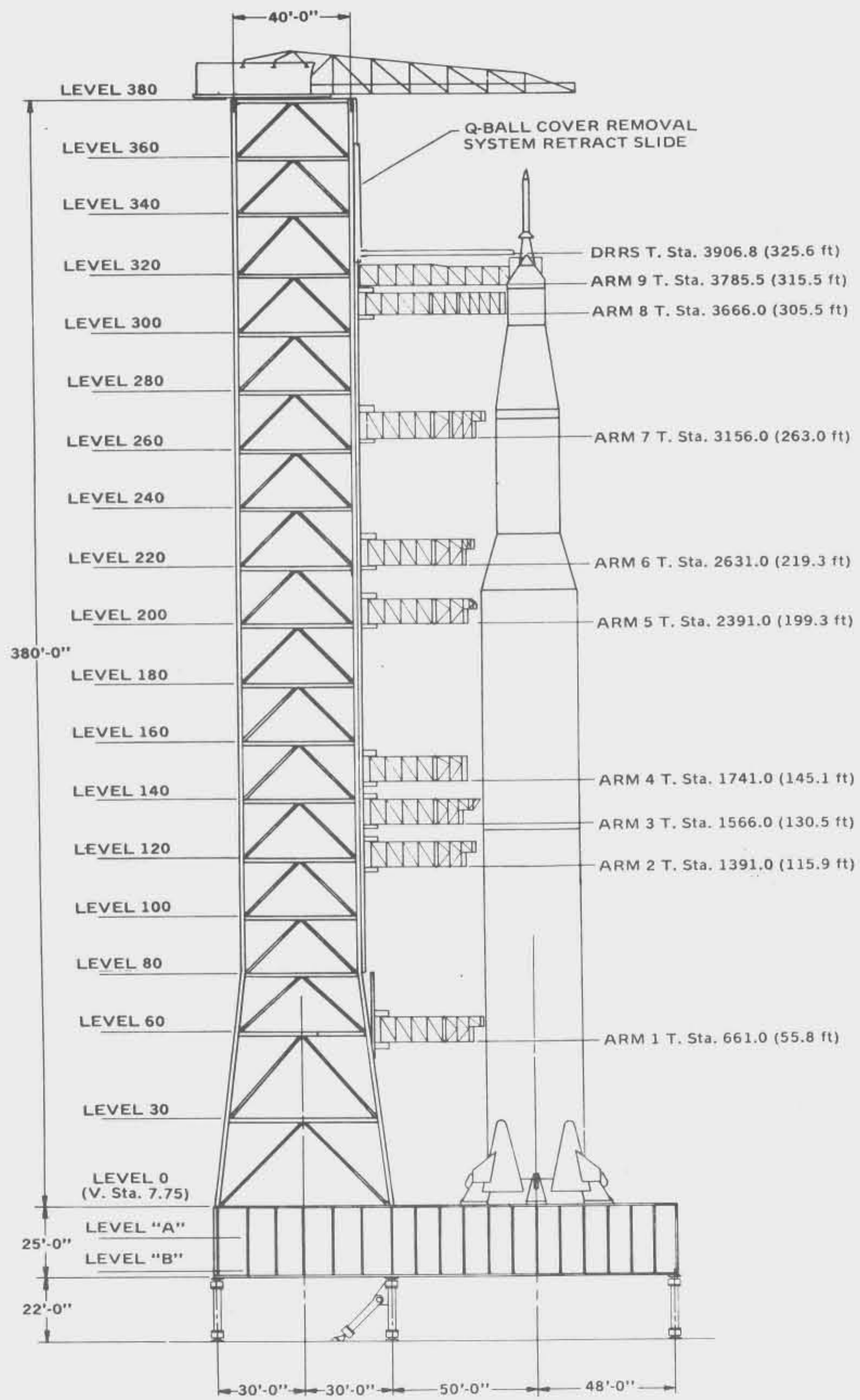
a) Two payloads are considered for the study. The first is a standard Apollo/CSM; the second is a Modified Launch Vehicle (MLV) Payload. Figure 1 indicates the shape and basic dimensions of each. For the MLV Payload, consideration is given only to access provisions, one access point at the nosecone, the other at the cylindrical section. For the Apollo/CSM, both servicing and accesss are considered.

b) The major facilities at LC-39 are defined as the Launcher-Umbilical Tower (LUT), Mobile Service Structure (MSS), Vehicle Assembly Building (VAB), Launch Control Center (LCC), and Pad. Assumed to be presently operational are three LUT's one MSS, three VAB High Bays, three LCC Firing Rooms, and two Pads.

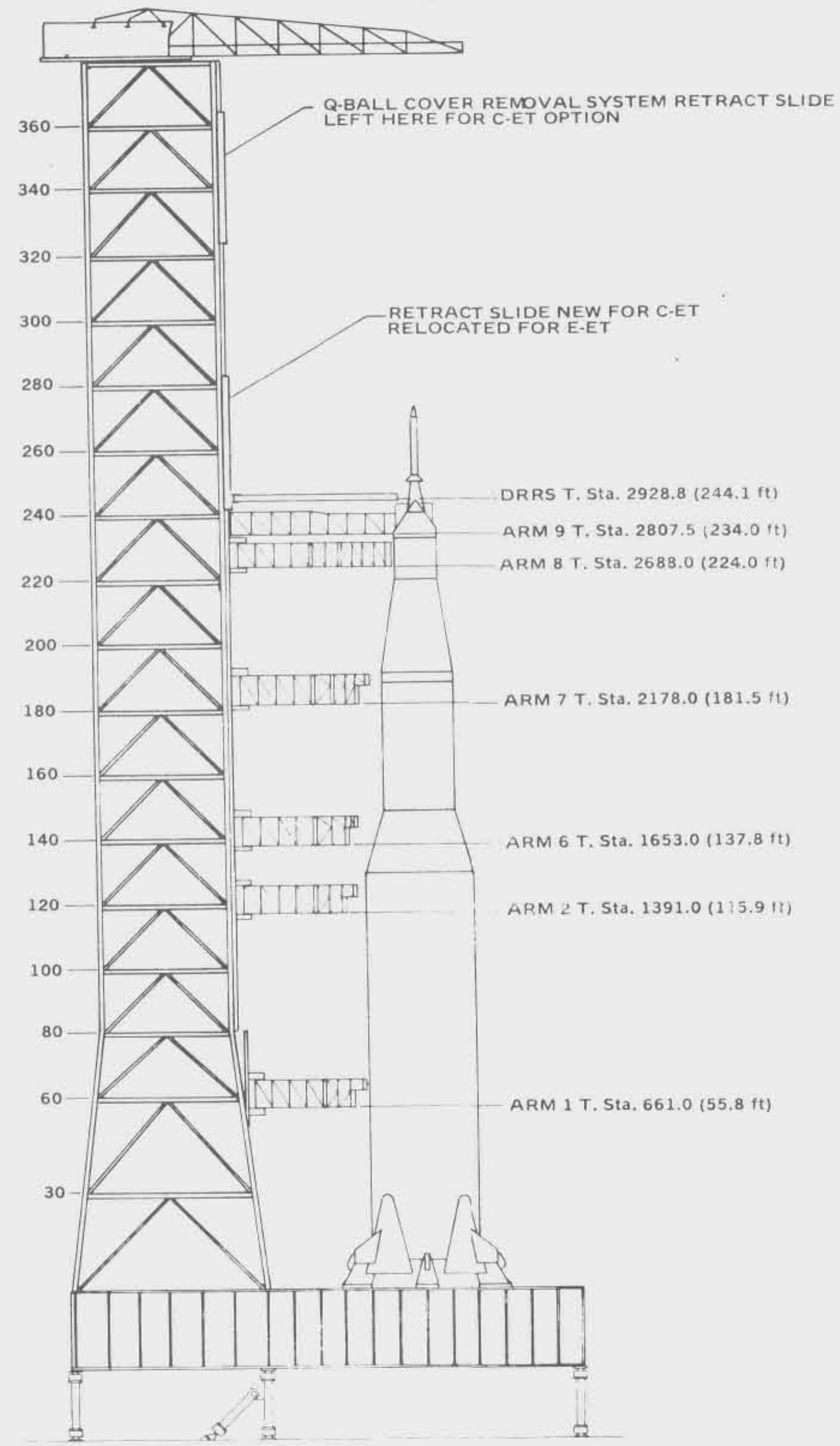
c) Each major facility (LUT, MSS, VAB, LCC, and Pad) was considered from two basic standpoints - a convertible facility and an exclusive INT-20 use facility.

<u>Convertible Launch Umbilical Tower (LUT)</u> (See Figure 5.4.2-4)

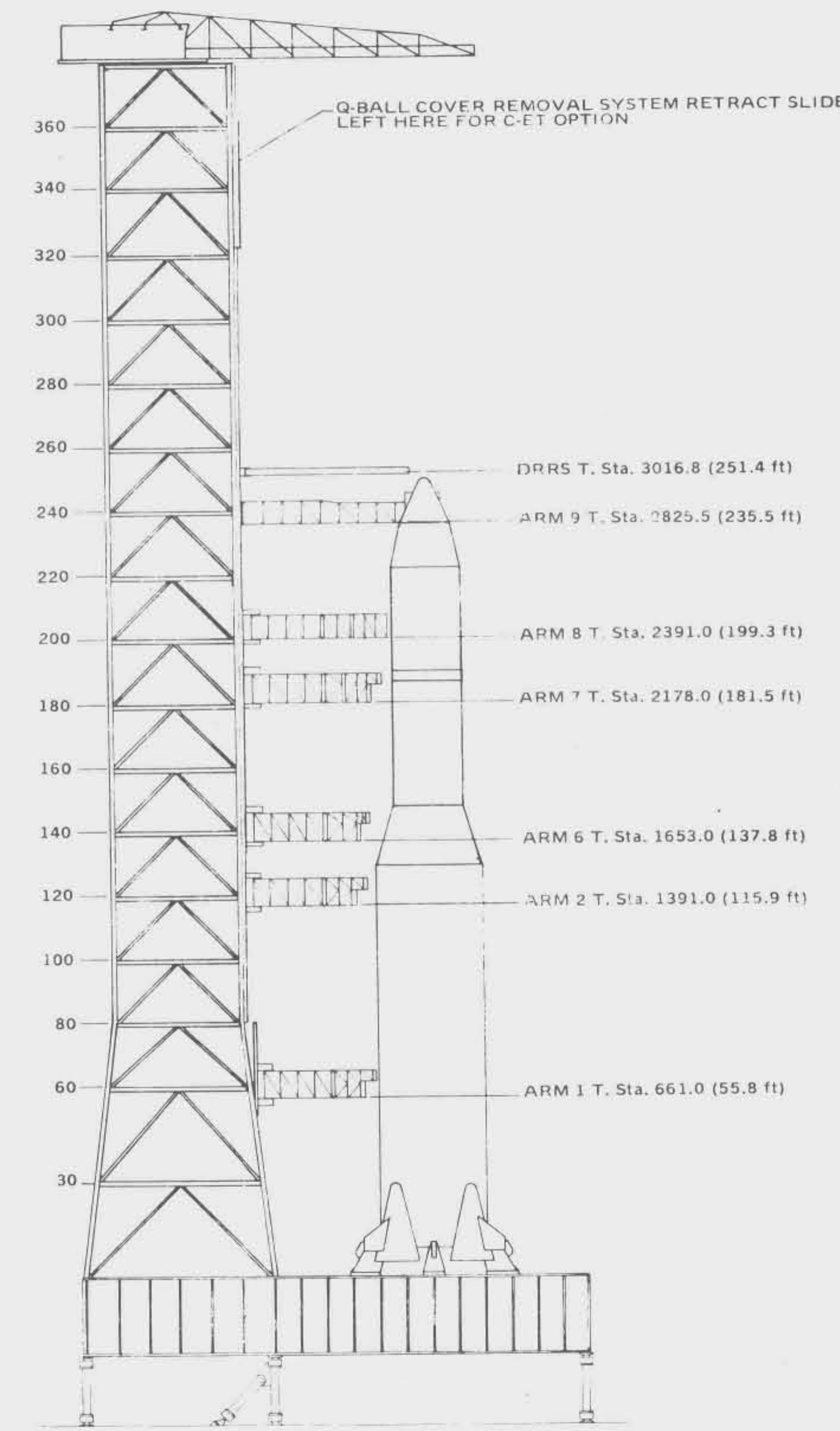
The convertible LUT option utilizes an existing LUT by modifying it to facilitate its conversion from a configuration which will support Saturn V/CSM operations to a configuration which will support INT-20 operations and vice-versa as the operational support requires. The S-IC stage is left in its present physical location (LUT zero level) to take advantage of the existing



SATURN V



INT-20/CSM



INT-20/MLV PAYLOAD

FIGURE 5.4.2-4
CONFIGURATIONS FOR EXISTING LUT
CONVERTIBLE OR EXCLUSIVE-USE

5.4.2 (Continued)

services, such as holddown arms, tail service masts, and service arms. For a CSM payload, this configuration will require an 81.5 foot lower location for the S-IVB, I.U. and CSM service arms and associated equipment in order to maintain the same vehicle interfaces. This is operationally and economically more feasible than attempting to provide duplicate services on the LUT at the lower levels. The service requirements for the MLV payload were not defined, and no provision was made for servicing; however, access is required and is provided by relocating the present CSM service arms to the proper level. The present S-II service arms, associated equipment, and S-II stage-peculiar equipment will be removed from the LUT.

Mobile Service Structure (MSS) Convertible (See Figures 5.4.2-5 and 5.4.2-6)

Two basic design approaches were considered in the development of an MSS for the INT-20 vehicle. These are the modification of the existing MSS for convertible use with either the Saturn V or the INT-20, and the provision of a new MSS for exclusive INT-20 use. Since there is presently only one MSS, exclusive-use modification of the existing MSS could not be considered.

The requirement to develop separate configurations for both the CSM and MLV payloads resulted in four configurations, two for the convertible option and two for the new option. Two additional configurations were developed for handling both payloads. Since only one MSS exists at LC-39, this facility is very critical.

The MLV configuration has many changes that are different from the CSM configuration. The Launch Escape System (LES) platform (Platform 5) is not used and remains in the Saturn V position. Interface disconnects are provided for the other platforms, which are lowered. Platform modifications include convertible annulus sections on Platforms 3-roof, 4A, and 4C to provide compatibility with either the MLV payload diameter for the INT-20 or the CSM diameter for the Saturn V. Also included is an opening in Platform 3 for entry of the MLV Aft Service Arm (S/A 8). Modifications which simplify reconversion operations to reduce costs and time are considered in this configuration as they were with the CSM configuration. Once MLV requirements are defined, additional services will probably be required at the new location of the platforms providing access to the MLV payload.

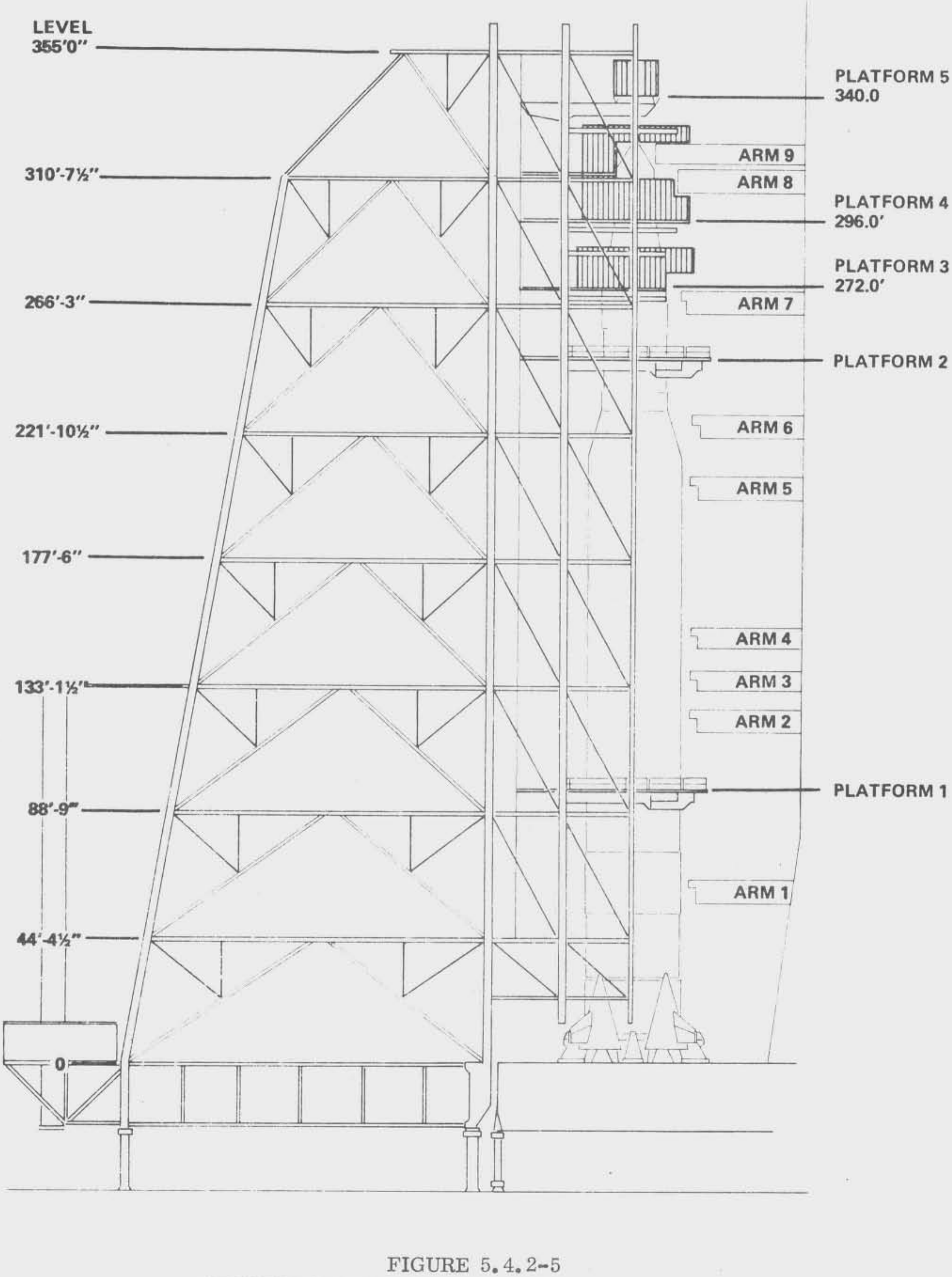


FIGURE 5.4.2-5
EXISTING MSS, SATURN V CONFIGURATION

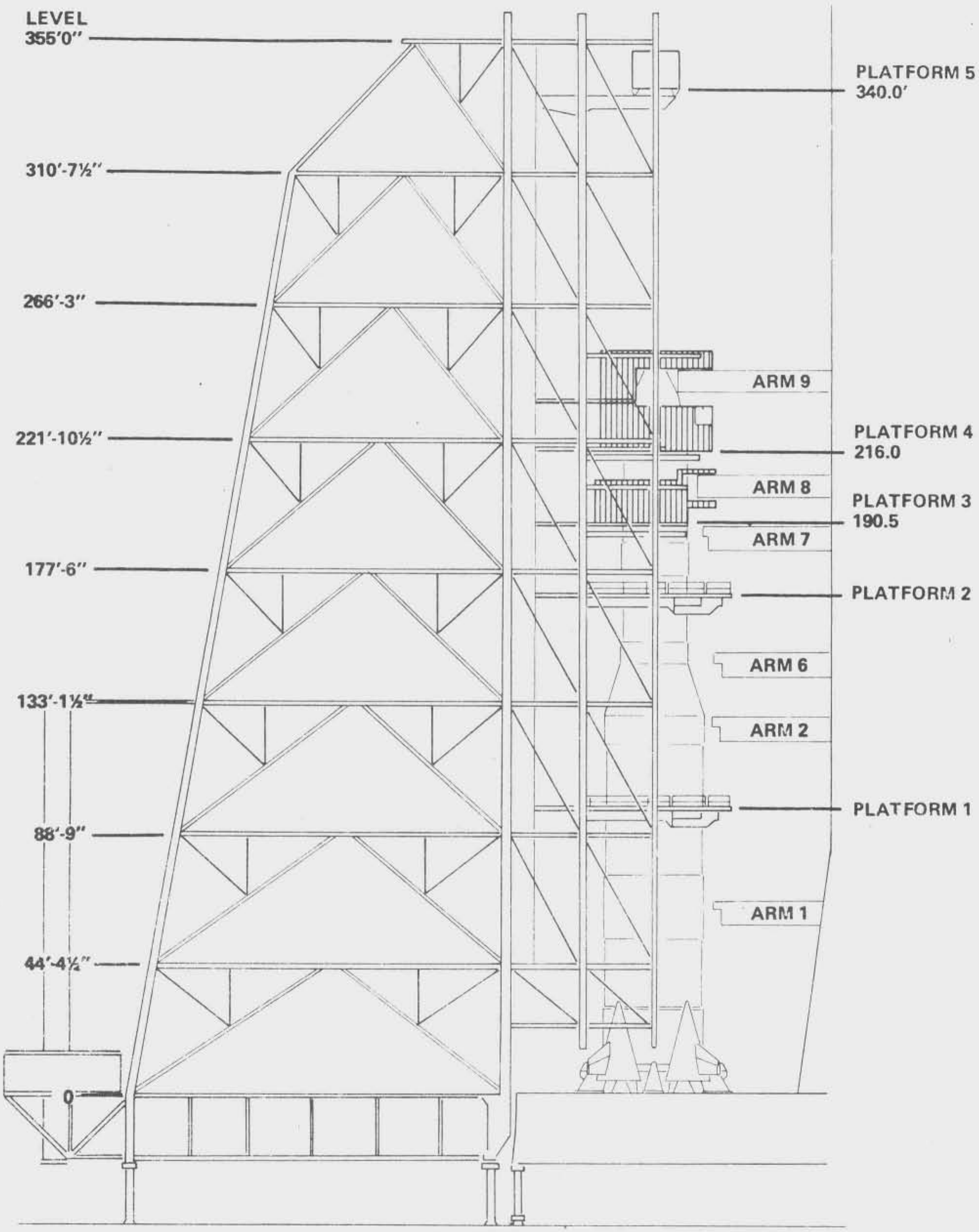


FIGURE 5.4.2-6
EXISTING MSS, CONVERTIBLE SATURN V/INT-20
MLV PAYLOAD

5.4.2 (Continued)

Vehicle Assembly Building (VAB) Convertible, Existing High Bay

Presently, three of the four High Bays in the VAB are equipped to process Saturn V vehicles. This study considered modifications to these equipped High Bays to support INT-20 vehicle processing. In addition, modification of the fourth, unequipped High Bay was studied for minimum services for the LUT with MSS functions. No modifications to the VAB Low Bay facilities are required.

Two basic design approaches were examined in the development of VAB options, a convertible High Bay for use with either the INT-20 or the Saturn V and an exclusive-use High Bay for use with the INT-20 only. Configurations for both the CSM and the MLV payloads were developed for each option.

For the convertible option two basic design approaches were examined. One approach utilized relocated platforms; the second utilized convertible annulus sections plus two additional platform levels. The relocation design provided configurations identical to the exclusive-use option. Modifications to provide additional work levels and convertible annulus sections resulted in higher implementation costs than the relocation design; however, conversion-reconversion costs and times are minimized by this modification approach. After two conversion operations the platform relocation method becomes more costly than the platform addition approach; therefore, the latter was selected. The modifications include the addition of a new platform between Platforms C and D to service the S-IVB forward/IU portion of the INT-20. Also the roof level of existing Platform C will be raised and a third work level (C-3) will be installed to provide access to the Command Module or MLV capsule. Certain platform levels will be provisioned with convertible annulus decking. Extension cables and waveguide will be permanently installed for all CSM, IU and S-IVB measurement, checkout, and RF systems between the existing platform interfaces and the new platform interfaces at the lower elevations for INT-20. Platform utilities for the new platform are acquired from existing vertical runs, by the same methods used for existing platforms.

The CSM configuration will include extension cables for Spacecraft Measurement, Checkout, and RF systems.

5.4.2 (Continued)

The MLV configuration will be the same as the CSM, except that the roof of Platform C will be extended an additional 1-1/2 feet and the new work level C-3 will be installed 1-1/2 feet higher for the MLV nosecone than for the CSM. Extension of Spacecraft cables will not be provided for the MLV configuration; however, special MLV services may be required after the MLV payload is defined.

Launch Control Center (LCC)

Four Firing Rooms presently exist at the LC-39 Launch Control Center, three of which are presently equipped to control Saturn V launch operations. Since the INT-20 launch vehicle stages, (S-IC, S-IVB, IU) are very similar to those of the Saturn V, only minor modifications are necessary to allow use of one or more of the equipped Firing Rooms with the study vehicle. This study considered modification of an equipped Firing Room to either of two configurations, an exclusive-use changeover to INT-20, or convertible, whereby the Firing Room could be used for either vehicle. No distinction between CSM and MLV payloads has been considered, since the Firing Room is primarily used for the launch vehicle. Firing Room changes required to support spacecraft functions are considered negligible.

For the Convertible INT-20/Saturn V, Existing Firing Room option, all cables from the S-II distributors and console panels will be disconnected and/or connected as required for each conversion. The cables will be capped when disconnected, but they will not be stowed. A switch will be added to the L/V Test Conductor's panel to convert from Saturn V to INT-20 or from INT-20 to Saturn V. This switch, through Integration ESE, will simulate the required S-II functions described in the exclusive-use option. Relays and necessary wiring for the signal simulation and switch operation will be installed in the integration racks. The reason for selecting the switch-relay method of conversion is that the rewiring of patch boards and associated circuit checking for each conversion is more costly than installing the switch and relay circuitry.

Pad

The pad will require no hardware modifications to support the INT-20 vehicle for the LUT options which utilize the Mobile Service Structure. All Pad functions can be satisfied by either procedural changes or

5.4.2 (Continued)

hardware changes on the LUT and MSS. When servicing requirements are defined for the MLV payload, it will probably be necessary to provide a new Pad interface with the LUT, plus associated Pad piping modifications to allow spacecraft and APS servicing at a location different from the existing Pad/MSS interface.

Convertible Concept Summary (Baseline II)

The Baseline II processing concept as shown in Table 5.4-I is a low cost approach to the use of a set of existing facilities for either INT-20 or Saturn V processing. An existing LUT and the existing MSS are modified for convertibility by providing for the relocation of service arms and access platforms. Existing work platforms are expanded and a new platform is added to accommodate either INT-20 or Saturn V vehicles in an existing High Bay without relocating work platforms. A convertible LCC Firing Room is used, and no significant modifications to the Pad are necessary.

Conclusions

The following major conclusions are drawn from the results of the study:

a) Concepts consisting of the modification of existing facilities (Baselines I and II) are practical approaches for the support of low launch rates.

b) Concepts involving existing facilities modified for exclusive INT-20 use (Baseline I and Alternate IA) offer no major cost or capability advantages over the convertible concepts (Baseline II and Alternate IIA). They do possess slight operational advantages.

c) Definition of MLV payload checkout and servicing operations will have a significant impact on the facility designs and associated costs and schedules.

d) For variable-length MLV payloads, facility configurations developed in this study may not represent the optimum approach to satisfying the requirements. Configurations were developed for providing equipment and services at specific locations or elevations, and the approach used may not be optimum for providing these services through a range of locations."

5.5 SCHEDULE PLAN

The schedule plan for the INT-20 covers the period from the contractural start of the program definition phase to the delivery of the first production vehicle to KSC. This plan allows for the normal design time, procurement of components and manufacturing time including required testing.

5.5.1 Vehicle Schedule Plan

The INT-20 Development and Delivery Plan is shown for the INT-20 programs on Figure 5.5.1-1.

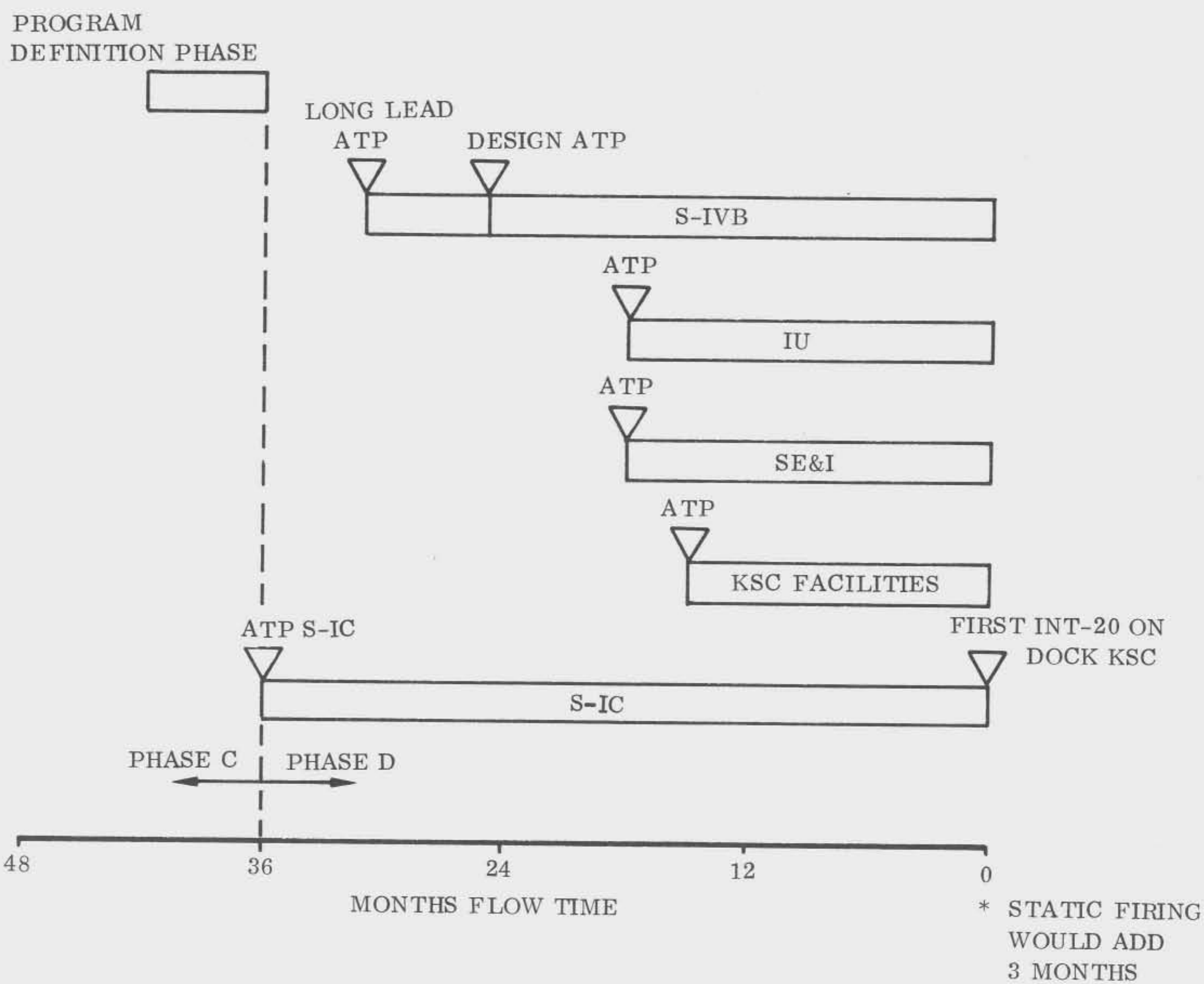


FIGURE 5.5.1-1 INT-20 DEVELOPMENT AND DELIVERY PLAN (NO STATIC FIRING*)

5.5.2 S-IC Schedule Plan

5.5.2.1 Baseline INT-20 Production Plan

The flow schedule shown by FIGURE 5.5.2-1 is based on start up of INT-20 stage procurement in addition to Sat V S-IC follow-on stages assumed to be under contract, in accordance with the study ground rules. The flow period for production start-up for the INT-20 configuration with no follow-on Sat V S-IC stages under contract will be one month longer. This increase in flow time for INT-20 production only,resultsfrom a 40 month procurement lead requirement instead of the 39 month procurement lead shown on FIGURE 5.5.2-1 for mixed production.

These schedules further reflect in-sequence production with resources utilization on a 5 day week with one shift for engineering and two shifts for manufacturing operations. FIGURES 5.5.2-2 through 5.5.2-7 are calendar oriented in-sequence production schedule summaries for the identified delivery dates in response to the study ground rules.

5.5.2.2 Alternate Production Schedule Consideration

Production planning to reduce the flow time from contract go-ahead to delivery of an INT-20 configuration would necessitate reconfiguration of S-IC stages under contract. Such a plan, however, involves an assumed Sat V - S-IC end item delivery obligation and must be considered a contract change for incorporation of the defined INT-20 changes. Such approach would also require the change to be defined by a revision of the S-IC documentation data base as shown in FIGURE 5.5.2-8 and the stage or stages must retain their originally designated Sat V S-IC effectivity identification.

For implementation of this approach, consideration must be given to the pacing long lead items unique to the INT-20 configuration. FIGURE 5.5.2-1 identifies the pacing INT-20 peculiar hardware item, which has been established to be the fuel tank base gore segments. This INT-20 change is defined in paragraph 4.2.2.1.a.4(c) and has been analyzed to be mandatory to meet the baseline INT-20 4.68 g acceleration requirement; see APPENDIX A paragraph 2.1.5.2.c. Other long lead INT-20 peculiar items are listed in order of their schedule impact.

Item	Procurement Lead (In Months)
Lengthened Fuel Loading Probe	28
Instrumented Heat Shiel Panels	19

5.5.2.2 (Continued)

Two design approaches could be used to accommodate reconfiguration of S-IC stages under follow-on (16 and on) contract to INT-20.

(a) Limit the four engine burn acceleration to 3.68 g and tailor the propellant loading to permit use of the existing S-IC fuel loading probe. This approach would reduce the INT-20 peculiar lead flow times to 19 months. By out-of-sequence production incorporation of the remainder of the INT-20 changes, minimum the flow period from contract go-ahead to delivery could approach the 18 months shown for retrofit of S-IC-14 in FIGURE 6.1.5-2.

(b) Another approach would be to change the configuration of the follow-on (18 and subsequent) buy to include the increased base gore thickness (added weight is approximately 300 pounds) and lengthen the fuel loading probe. To use a lengthened loading probe for Sat V S-IC ullage volumes would require a change in sensor length instead of a change in stillwell length only, as defined for INT-20. Revision of the sensor length would also require that the loading electronics would have to be changed to be compatible. All other changes for the 4.68 g baseline INT-20 configuration could then be incorporated, under a contract change, to approach the minimum retrofit flow period.

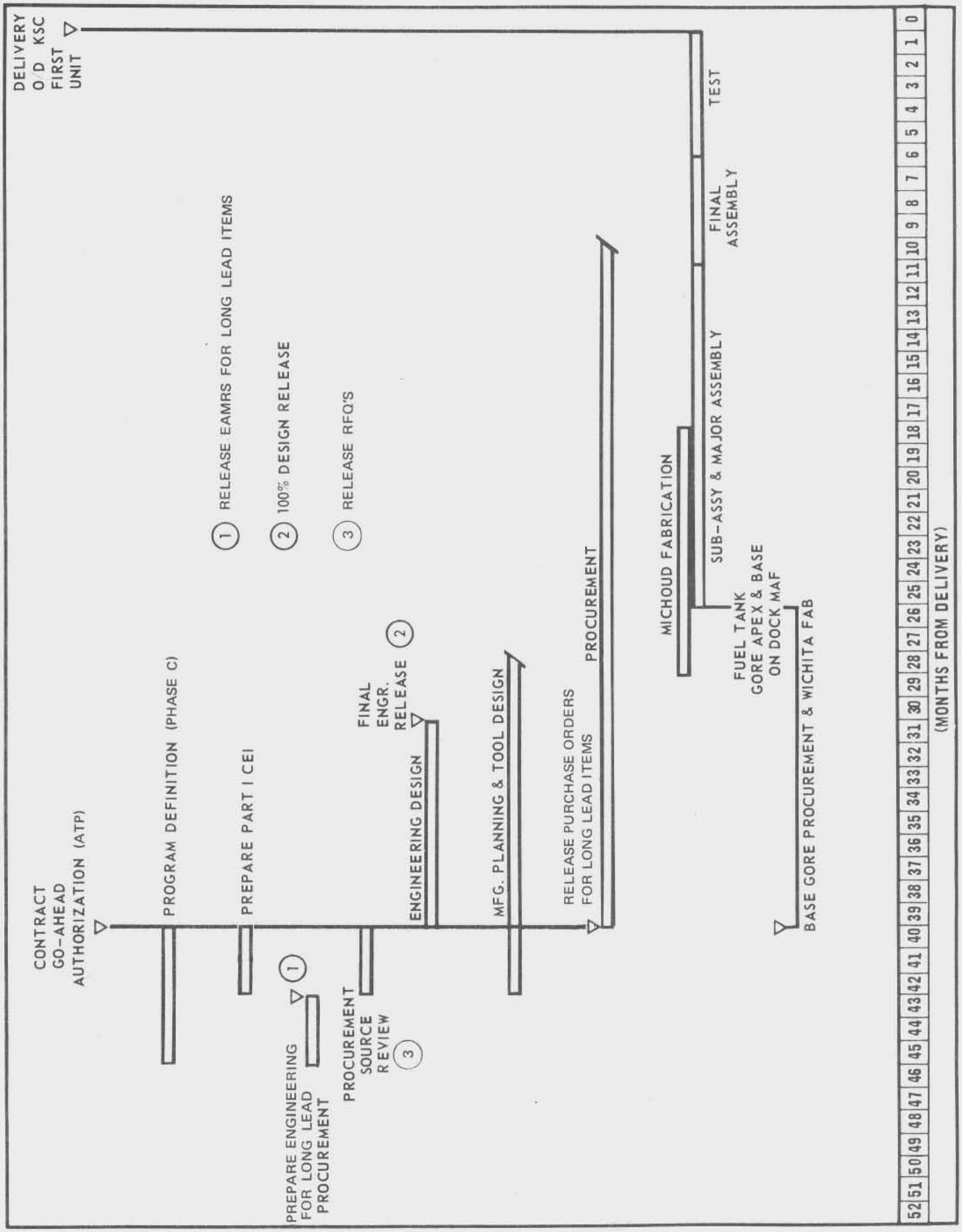


FIGURE 5.5.2-1 INT-20 SCHEDULE

2 S-IC'S PER YEAR - NO INT-20'S

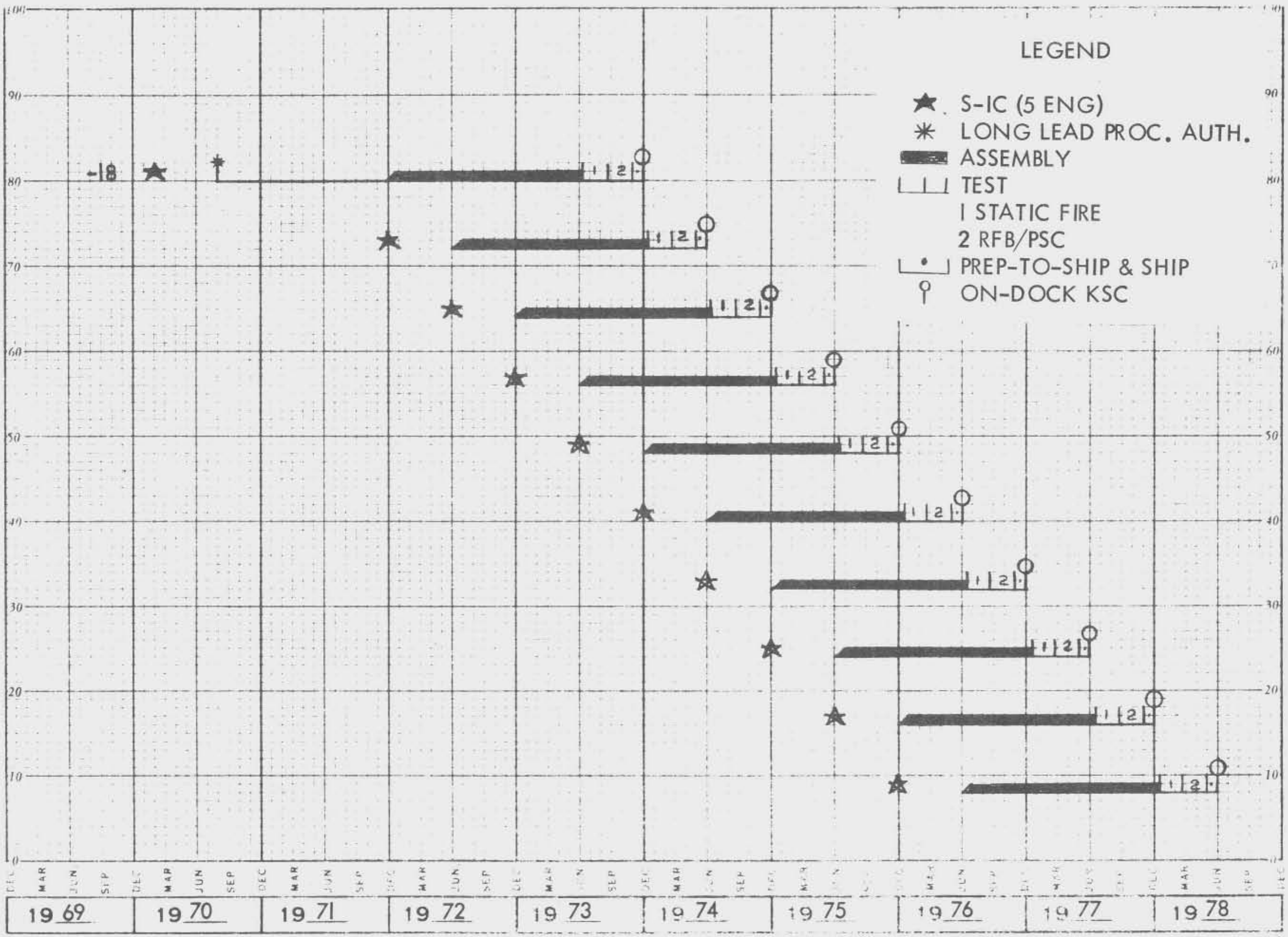


FIGURE 5.5.2-2 PRODUCTION SCHEDULE

PR&TA 5-1230 5-26-9

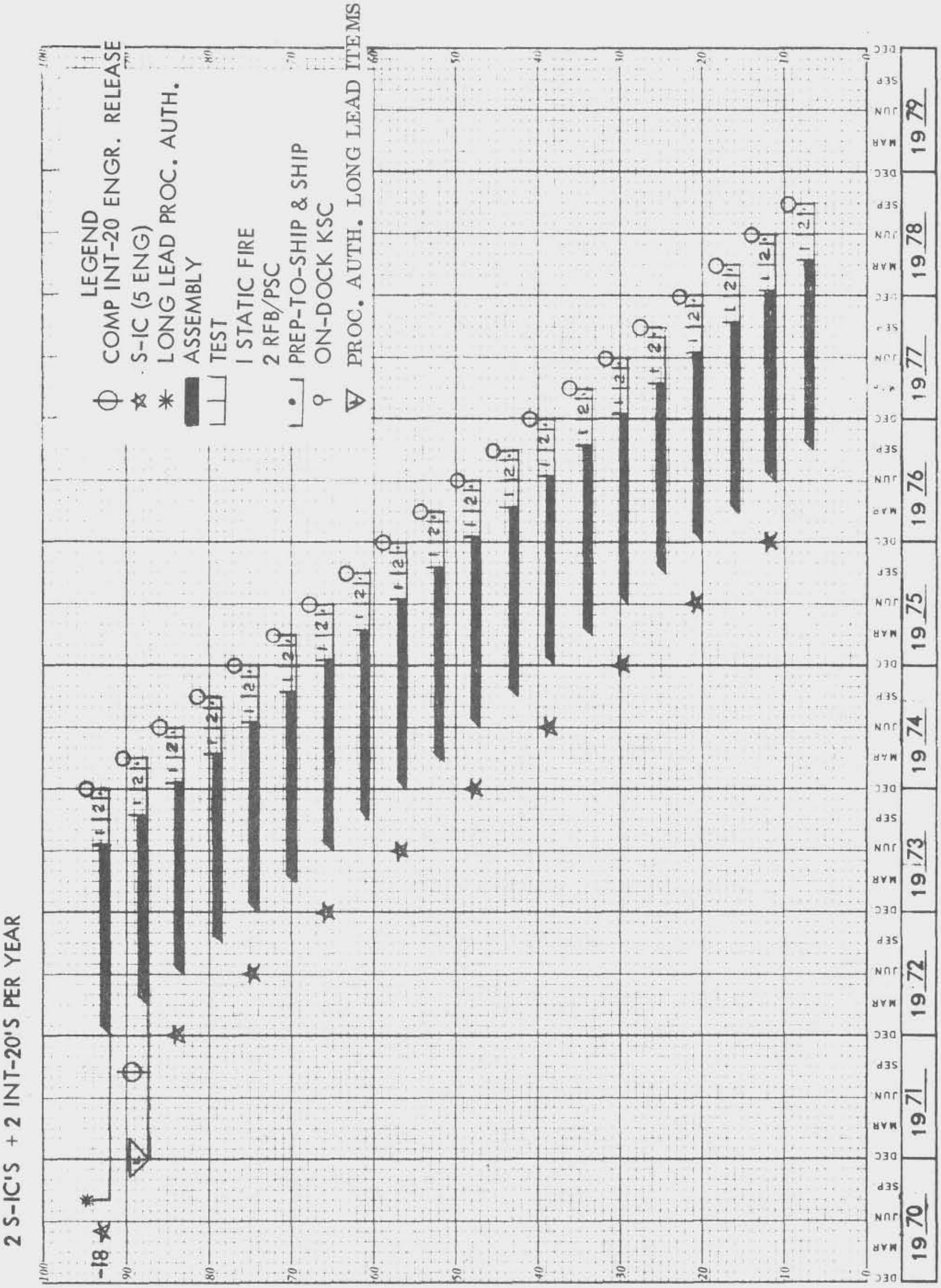


FIGURE 5.5.2-3 PRODUCTION SCHEDULE

PR&TA 5-1230 5-26-9

3 S-IC'S + 3 INT-20'S PER YEAR

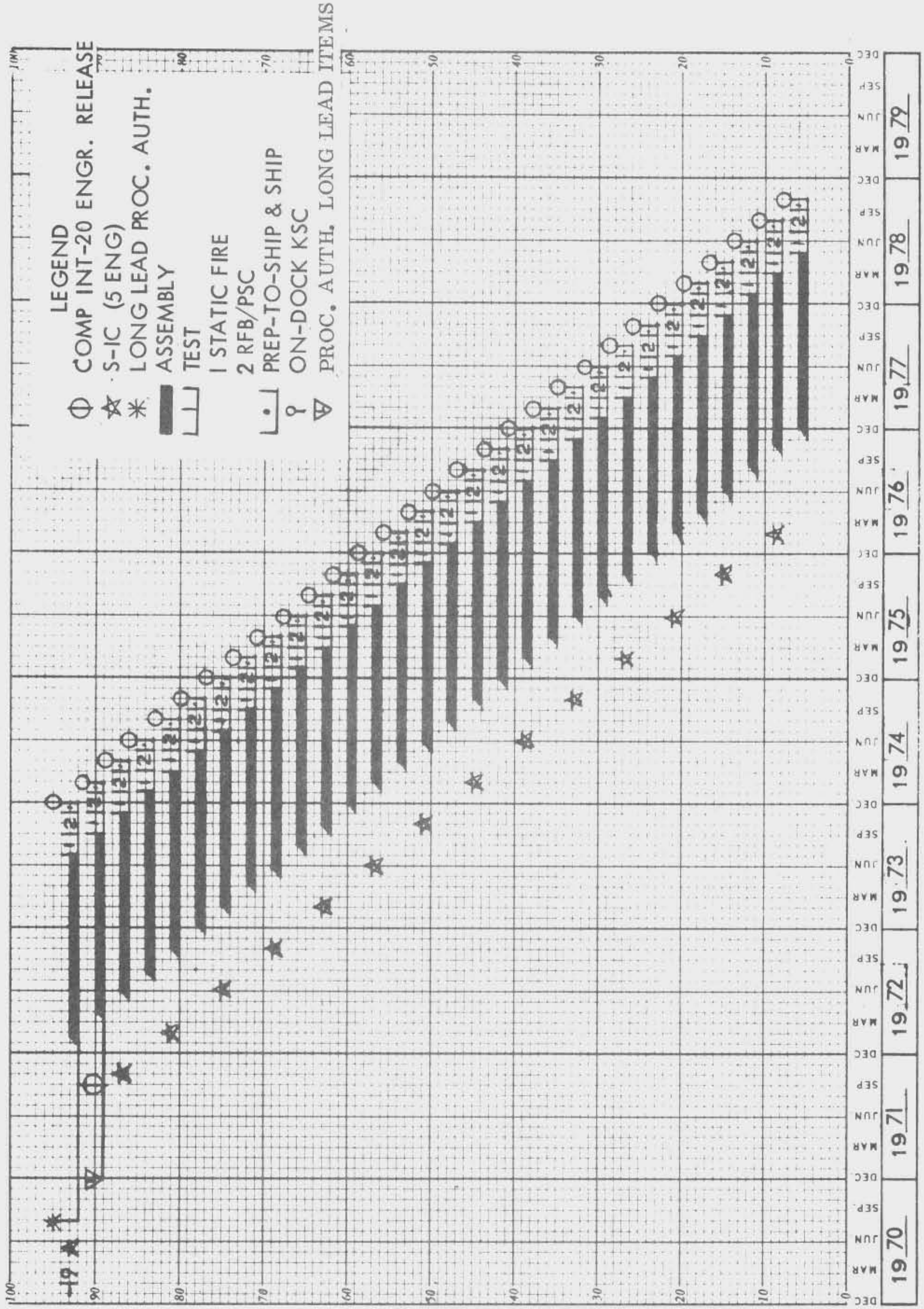


FIGURE 5.5.2-4 PRODUCTION SCHEDULE

2 S-IC'S + 4 INT-20'S PER YEAR

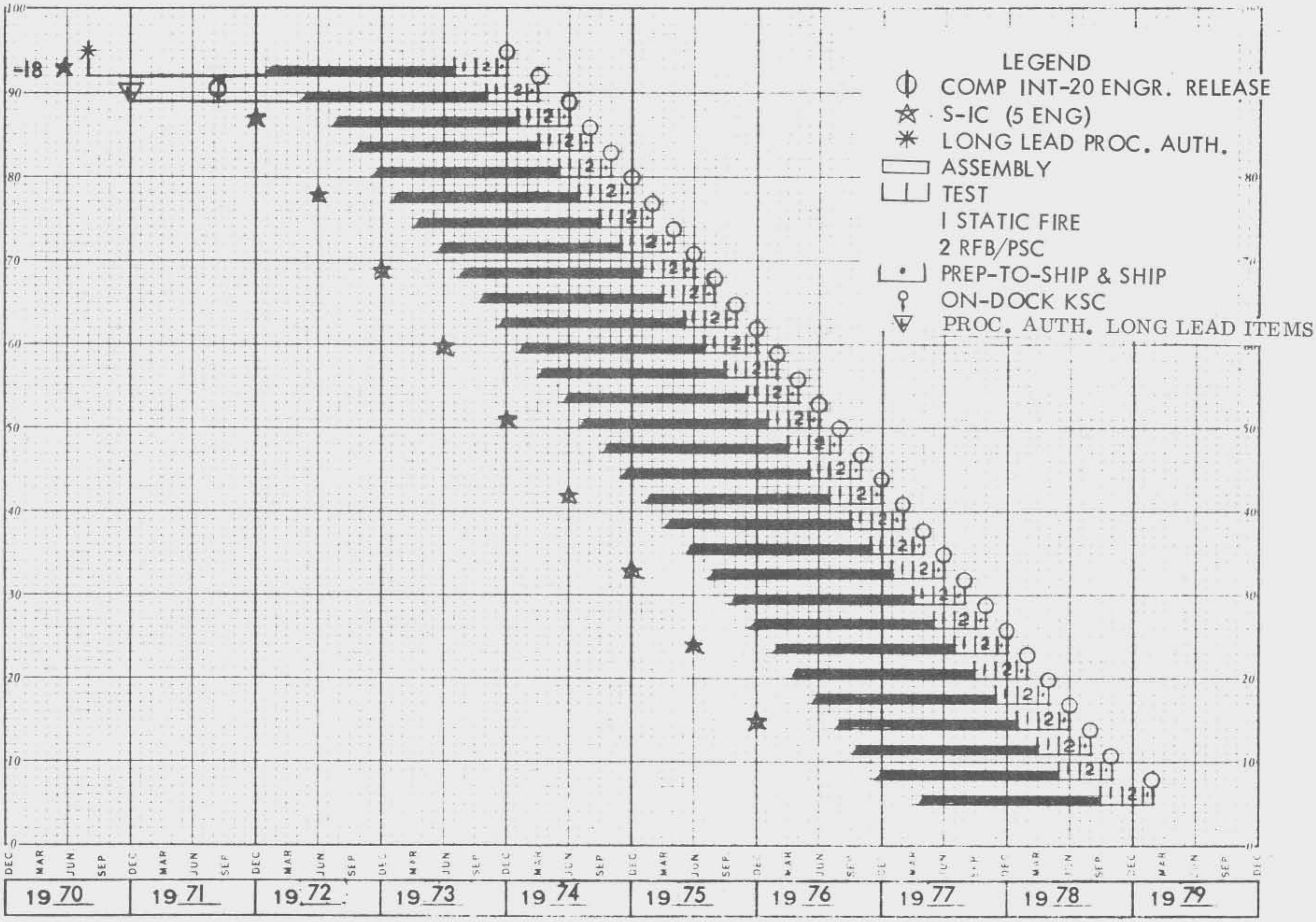
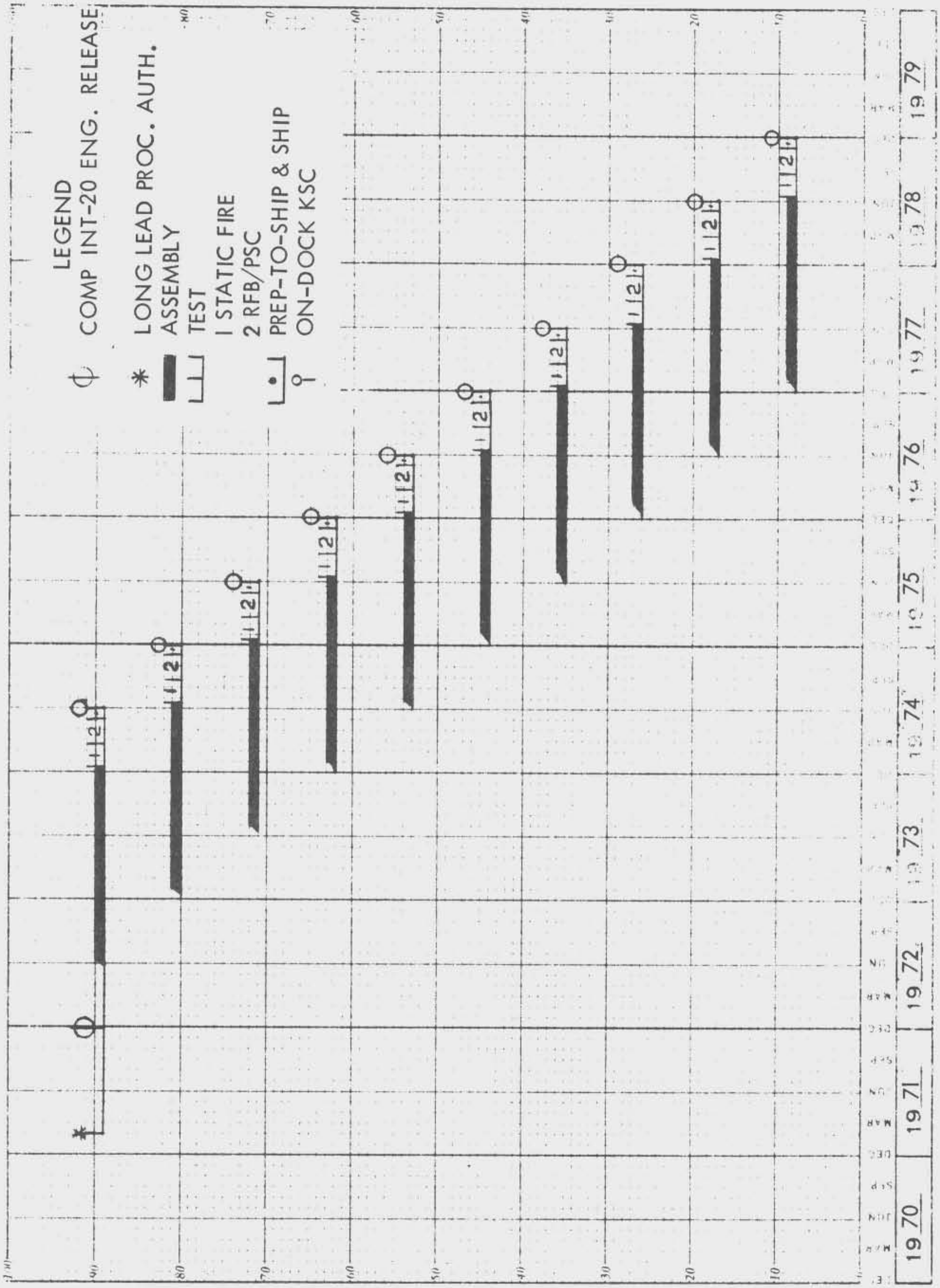


FIGURE 5.5.2-5 PRODUCTION SCHEDULE

PR&TA 5-1230 5-26-9

2 INT-20'S PER YEAR - NO S-IC'S



PR&TA 5-1230 5-26-9

FIGURE 5.5.2-6 PRODUCTION SCHEDULE

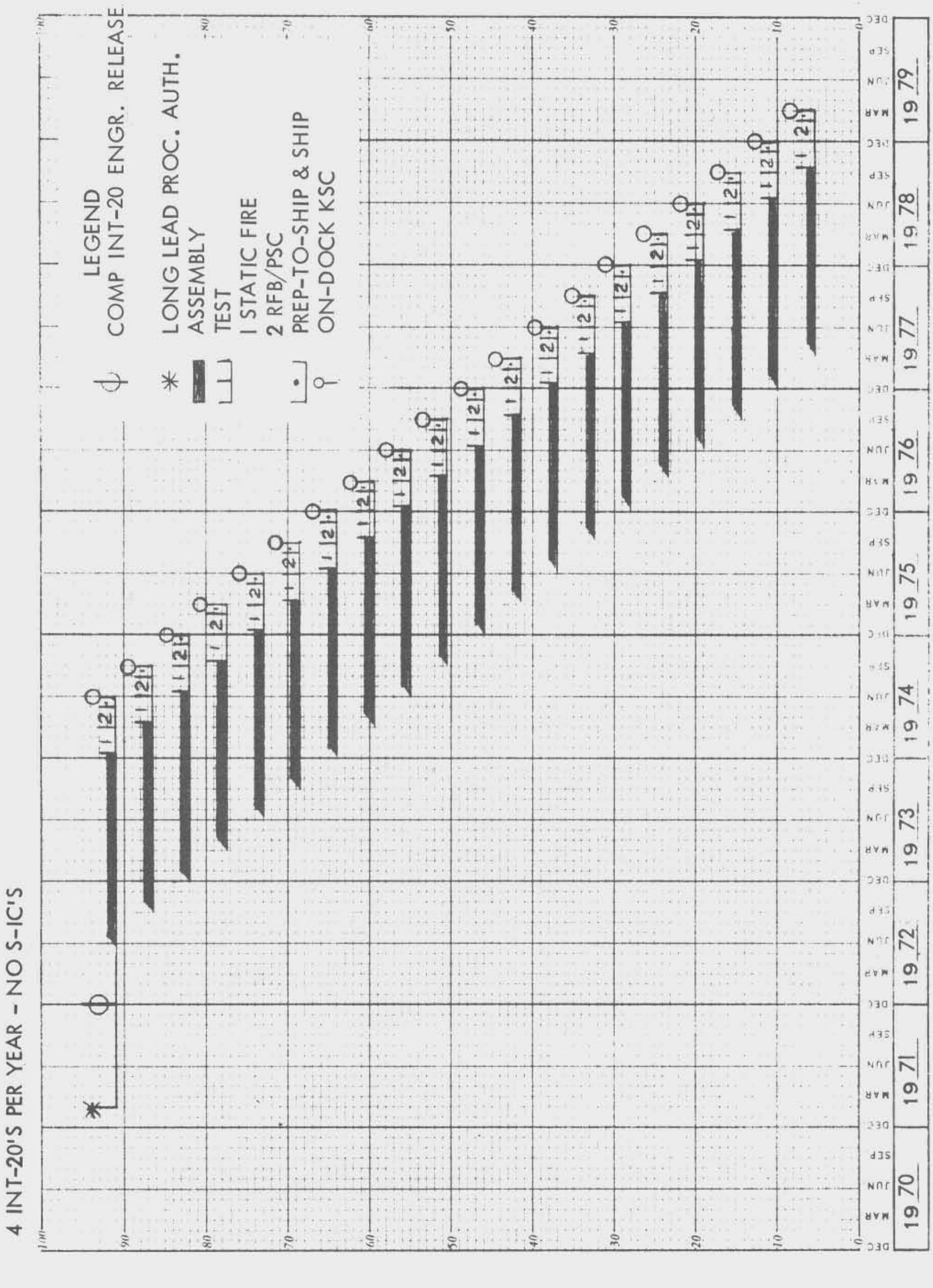


FIGURE 5.5.2-7 PRODUCTION SCHEDULE

PR&TA 5-1230 5-26-9

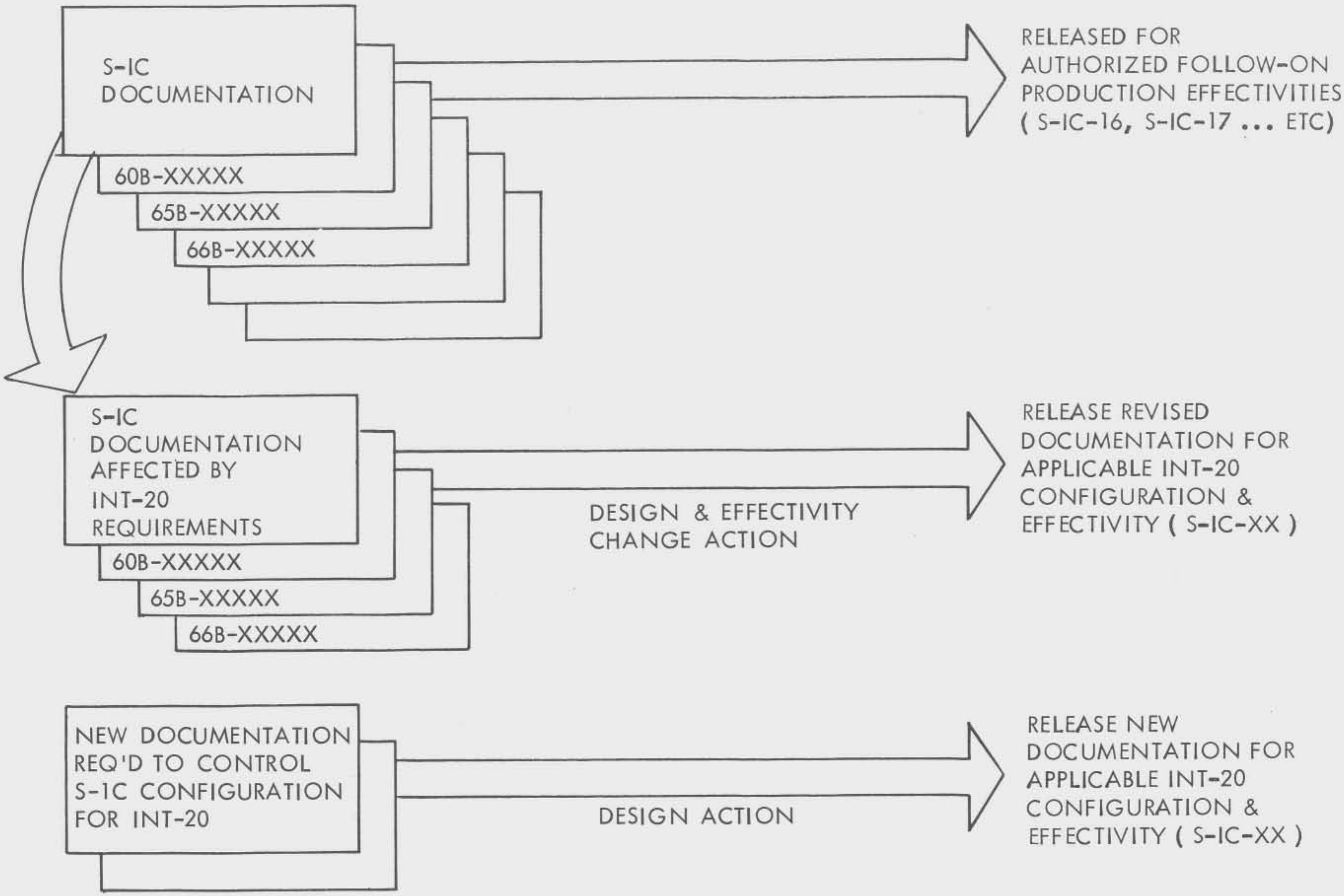


FIGURE 5.5.2-8 INT-20/S-IC STAGE DOCUMENTATION PLAN

5.5.3 S-IVB Stage Schedule Plan

The integrated development schedule for the S-IVB stage of the INT-20 vehicle is presented in Figure 5.5.3-1 in terms of months from Phase D ATP. A six month PDP followed by a 3 month negotiation phase was assumed to precede ATP. Long lead time parts and equipment purchasing is initiated at ATP and fabrication is initiated at three months (raw material purchasing was assumed in process six months prior to ATP). The first unit is ready for J-2 engine installation and stage acceptance checkout 18 months from ATP. Delivery of the first unit at KSC occurs about 22 months after ATP, and first launch then may occur 25 months after ATP. This schedule represents INT-20 availability in the minimum practical time and is compatible with mainline Apollo Program schedules.

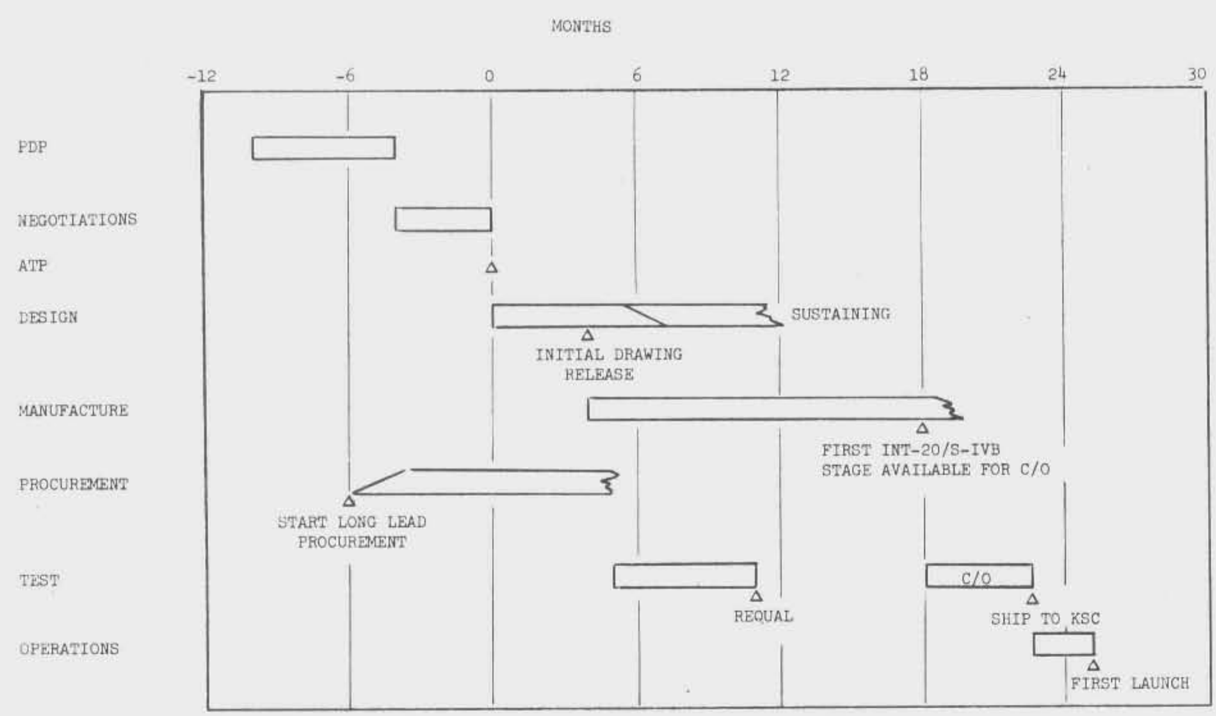


Figure 5.5.3-1. INT-20/S-IVB Development Schedule

5.5.4 IU Schedule Plan

An IU mission cycle is defined as that period of time between the first issue of an Instrumentation Program and Components (IP&C) List for a given IU and the launch of a vehicle with that IU. This cycle is 24 months, established by scheduling availability of the IP&C List 18 months prior to an IU delivery and an IU delivery which is scheduled against an arbitrary launch date to occur six months later. Expressing schedules in terms of mission cycles is not necessarily useful if the first and all subsequent IU's are alike. The use of a mission cycle provides planning visibility when considering that each IU is released in a unique configuration to satisfy peculiar requirements of a particular mission/vehicle configuration; a situation which is appropriate to this study.

From experience, the availability date of the IP&C List is a meaningful point of departure. At this point in time, through the IP&C List, the measurements requirements are established and the configuration baseline for instrumentation hardware is established. Together, they constitute a major portion of the IU electrical network design. Dependent on the degree to which mission objectives change, or are different, IU electrical-network design is normally subject to change with each IU. Normally, such changes are easily incorporated into hardware design within manufacturing and production control schedules for IU fabrication and assembly. Further, the lead times for all hardware procurements are between a point in time 18 months prior to IU delivery and the start of IU assembly, a period 12 months. Since all the lead times considered in this study fell within this period, Authority to Proceed (ATP) for any of the IU programs for INT-20/Saturn V application could be coincidental with issuance of the IP&C List for any given IU.

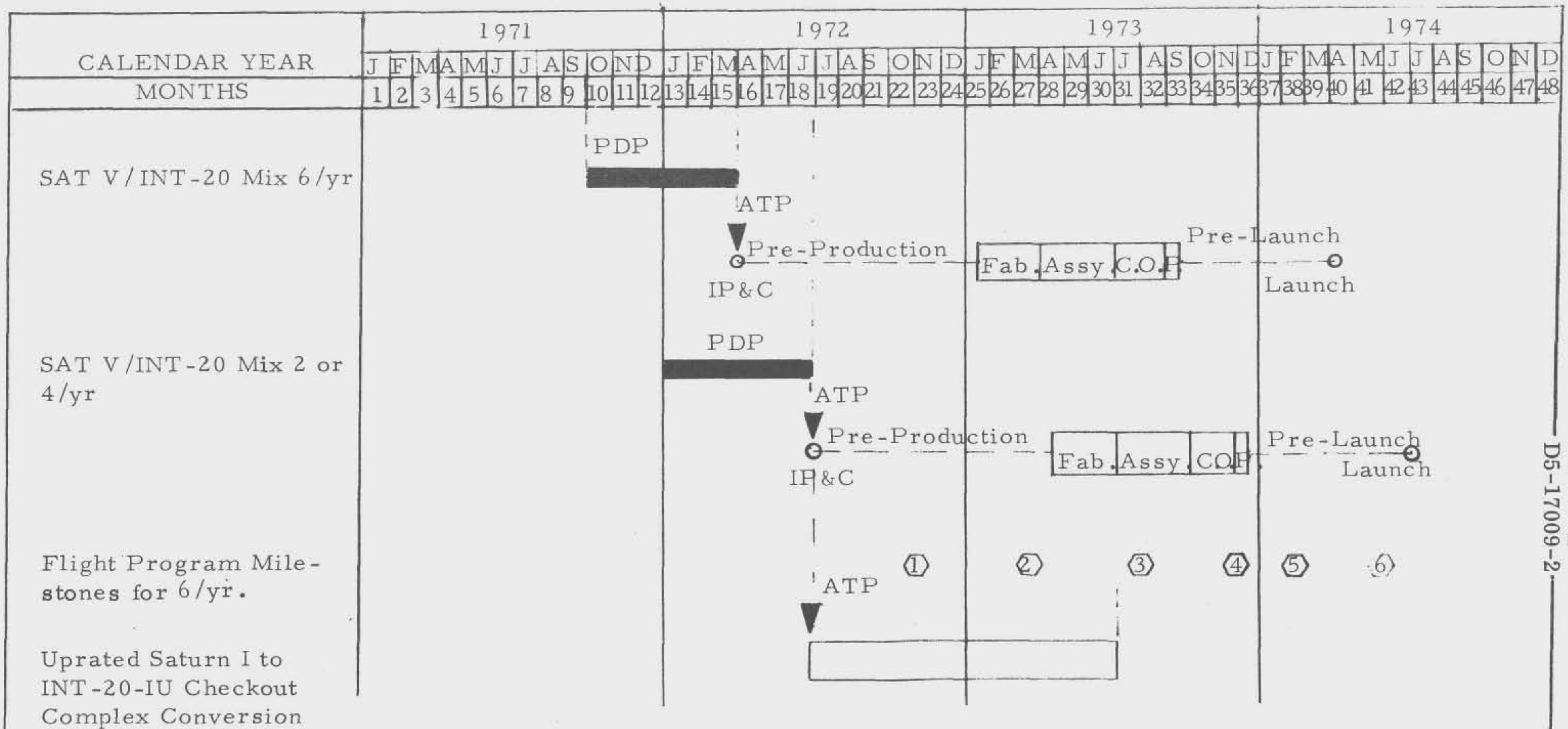
5.5.4.1 IU Delivery Schedule

The master schedule for production and delivery of completed IU assemblies as shown in Figure 5.5.4-1, was arranged to be compatible with schedules for the total vehicle delivery dates and launch readiness dates for contracted vehicles (AS-507 through AS-515) and proposed delivery dates for the follow-on Saturn V's (through AS-527). Launch dates for the follow-on vehicles assumed to be about six months after the delivery dates.

Two distinct ATP's are shown on the master schedule due to the production rates.

The ATP of April 1, 1972, reflecting the definition phase for INT-20 first vehicle (INT-20 (1)) production schedule with Saturn V mixed rate, and the ATP of July 1, 1972, depicting the definition phase for INT-20 first article (INT-20 (1)) production without Saturn V mix.

These production schedules encompassing five separate program developments for yearly production rates of:



LEGEND: IP&C, Instrumentation Program and Components; ATP, Authority to Proceed; PDP, Program Definition Phase; Fab, Fabrication; Assy, Assembly; C.O., IU Systems Test; P, Preparation for Shipment;

Flight Program Milestones ⬡ : 1, Mission Defining Document (MDD); 2, Equation Defining Document (EDD); 3, Final Mission Defining Document (FMDD); 4, Level II Program; 5, Verification Completed; 6, Post-Flight Data Reduction Complete.

FIGURE 5.5.4-1. MASTER PHASING SCHEDULE

5.5.4.1 (Continued)

Two Saturn V plus two INT-20.

Two Saturn V plus four INT-20.

Three Saturn V plus three INT-20.

Two INT-20 only.

Four INT-20 only.

Note that INT-20 and Saturn V's may be mixed arbitrarily in combinations resulting in rates of two, four, and six per year because of the insensitivity of the manufacturing cycle to the minor differences between the IU's.

5.5.4.2 Schedule Ground Rules and Assumptions

Facility modification of Uprated Saturn I Ground Support Equipment to support six per year rate will not interfere with the assembly or redelivery of stored Uprated Saturn I IU's.

There will be no interference between assembly and final checkout of 518 or 519 and the fabrication assembly of INT-20 (1) IU.

A Saturn V and INT-20 IU can be arbitrarily intermixed in scheduling onto the assembly floor.

Time between delivery of the last standard Saturn V IU and introduction of the first Saturn V IU with modifications for INT-20 capability is time phased for efficient transition without loss of continuity in facility utilization.

Program Definition Phase of six months is dictated as a study ground rule but not necessarily required for contractural implementation.

Schedules are based on one shift, five day week for Manufacturing and Engineering.

Air transportation is assumed for IU delivery to the KSC.

5.6 COST PLAN

The cost plan provides budgetary estimates to implement a five year program of Intermediate-20 (S-IC/S-IVB/IU) launch vehicles. Five different INT-20 programs have been analyzed. The development cost for the INT-20 vehicle and the hardware delta cost (difference between INT-20 hardware procurement and corresponding Saturn V hardware procurement) for each program is listed in Table 5.6-I.

TABLE 5.6-I INT-20 DEVELOPMENT COST AND HARDWARE DELTA COST

INT-20 PROGRAM ANNUAL LAUNCH RATE	INT-20 DEVELOPMENT COST	* INT-20 HARDWARE DELTA COST
2 Saturn Vs + 2 INT-20s	$7.49M	-$.81M
2 Saturn Vs + 4 INT-20s	$9.79M	-$.73M
3 Saturn Vs + 3 INT-20s	$8.19M	-$.73M
2 INT-20s (No Saturn Vs)	$7.49M	-$.95M
4 INT-20s (No Saturn Vs)	$7.49M	-$.82M

* Comparison of Saturn V component cost with INT-20 cost.

5.6.1 The INT-20 Vehicle Cost Plan

The cost plan is in accordance with the Resource Ground Rules listed in Paragraph 5.0.

The integrated vehicle cost analysis is based on the Saturn V cost data of the "National Space Booster Study, Part One, Cost Analysis of Current Launch Systems, Saturn Systems Presentation, Contract NASW-1740, October 3, 1968, by Chrysler Corporation Space Division" (Reference 1.3.3-1). When a Saturn V Cost Reduced baseline is established, the INT-20 development costs and the INT-20 hardware delta costs of this study may be applied to the new baseline to obtain the cost of a "Cost Reduced INT-20". Launch costs and launch support cost were estimated by The Boeing Company.

Delta costs, which state the difference between the cost of INT-20 stages and Instrument Unit and the corresponding hardware for a Saturn V, have been determined by the respective stage and IU contractors. All delta costs are negative, i.e., cost reduction. The delta costs are subtracted from INT-20 hardware costs based on the "National Space Booster Study" (Reference 1.3.3-1).

5.6.1 (Continued)

Development costs for the INT-20 consist of (1) establishing a new data base and coding drawings for the four-engine S-IC stage, requalifying the S-IVB stage for its new environment nearer the S-IC stage and reprogramming the I.U., (2) modifying KSC launch facilities to accommodate the shorter two-stage INT-20, (3) qualifying the F-1 engines by a static firing program for longer firing duration and, (4) reprogramming SE&I flight analysis computers. Development costs are higher for the IU for a 6 per year rate because the Saturn V IU production line is equipped for 5 IU's per year and to handle the sixth IU, the Saturn IB line would need modification.

Facilities costs are zero, except for KSC Launch Facilities. The development cost to modify one set of KSC Launch Facilities for use by either a Saturn V or an INT-20 vehicle with an MLV payload is $3,200,000. Facilities modified are the Launch Umbilical Tower, the Mobile Service Structure, a high bay of the Vehicle Assembly Building and a Launch Control Center. The Launch Pad does not need modification. Each modified facility may remain in the INT-20 configuration except the Mobile Service Structure of which there is only one. The Mobile Service Structure must be converted from the Saturn V configuration to the INT-20 configuration and then returned to the Saturn V configuration as the launch schedule dictates at a cost of $188,700 per round-trip conversion. The cost of MSS conversions is added to the operation cost of each INT-20 program, however, the cost may become zero if the conversions were made by a launch support contractor.

The annual cost to operate the Mississippi Test Facility (per Reference 1.3.3-1) is $30.0 million for a 2 Saturn V per year rate and $32.6 million for a 4 per year rate. The S-IC portion of these static firing costs is $3.74 million per S-IC at a rate of 4 per year, and $7.13 million per S-IC at a rate of 2 per year. The Sacramento Test Facility cost to static fire each S-IVB stage is $.5 million per S-IVB stage. The costs in this paragraph are to be deducted from the INT-20 vehicle basic cost for calculation of programs without stage static firing.

Total program cost and total operational cost for each 5 year program is calculated. Total costs include hardware procurement support, SE&I and launch and are calculated both with stage static firing and without stage static firing. The total costs are divided between the Saturn Vs and the INT-20 vehicles by the "incremental cost method" and the "distributed cost method" to obtain INT-20 average unit cost. By the "incremental cost method", the total cost of INT-20 vehicles is the difference between the total cost of the combined Saturn V/INT-20 program and the total cost of the Saturn V program alone. By the "distributed cost method", the total cost of the INT-20 vehicles is obtained by adding the INT-20 proportionate share of each Saturn V/INT-20 cost element (i.e., hardware, support, SE&I and launch). For each method, the total cost of the combined Saturn V/INT-20 program is the same but the cost attributed to each of the two vehicles is different.

COST TABLES ARE LISTED:

TABLE 5.6.1-I	INT-20 Delta Recurring Costs
TABLE 5.6.1-II	INT-20 Development Costs
TABLE 5.6.1-III	Comparison of "Incremental Cost Method" and "Distributed Cost Method", No Static Firing
TABLE 5.6.1-IV	Comparison of "Incremental Cost Method" and "distributed Cost Method", With Static Firing
TABLE 5.6.1-V	Average Unit Costs (Operational)
TABLE 5.6.1-VI	Cost Summary, 2 Saturn Vs/Yr., No Static Firing
TABLE 5.6.1-VII	Cost Summary, 2 Saturn Vs + 2 INT-20s/Yr., No Static Firing, Incremental Cost Method
TABLE 5.6.1-VIII	Cost Summary, 2 Saturn Vs + 4 INT-20s/Yr., No Static Firing, Incremental Cost Method
TABLE 5.6.1-IX	Cost Summary, 3 Saturn Vs/Yr., No Static Firing
TABLE 5.6.1-X	Cost Summary, 3 Saturn Vs + 3 INT-20s/Yr., No Static Firing, Incremental Cost Method
TABLE 5.6.1-XI	Cost Summary, 2 INT-20s (No Saturn Vs)/Yr., No Static Firing
TABLE 5.6.1-XII	Cost Summary, 4 INT-20s (No Saturn Vs)/Yr., No Static Firing
TABLE 5.6.1-XIII	Cost Summary , 2 Saturn Vs/Yr., With Static Firing
TABLE 5.6.1-XIV	Cost Summary, 2 Saturn Vs + 2 INT-20s/Yr., With Static Firing, Incremental Cost Method
TABLE 5.6.1-XV	Cost Summary, 2 Saturn Vs + 4 INT-20s/Yr., With Static Firing, Incremental Cost Method
TABLE 5.6.1-XVI	Cost Summary, 3 Saturn Vs/Yr., With Static Firing

TABLE 5.6.1-I
INT-20 DELTA RECURRING COSTS

(The Recurring Cost Difference is to be subtracted from Saturn V stage costs as reported in Reference 1.3.3-1 to give the cost of a corresponding INT-20 stage.)

	Delta Cost Per INT-20 Vehicle Dollars in millions				
Rate (Sat V/INT-20)	2 + 2	2 + 4	3 + 3	0 + 2	0 + 4
S-IC Stage	-.61	-.53	-.53	-.72	-.61
S-IVB Stage	-.20	-.20	-.20	-.23	-.21
I.U.	None	None	None	None	None
F-1 Engine	None	None	None	None	None
J-2 Engine	None	None	None	None	None
Total Delta Per INT-20 Vehicle	-.81	-.73	-.73	-.95	-.82

TABLE 5.6.1-II

INT-20 DEVELOPMENT COST

(1968 DOLLARS IN MILLIONS)

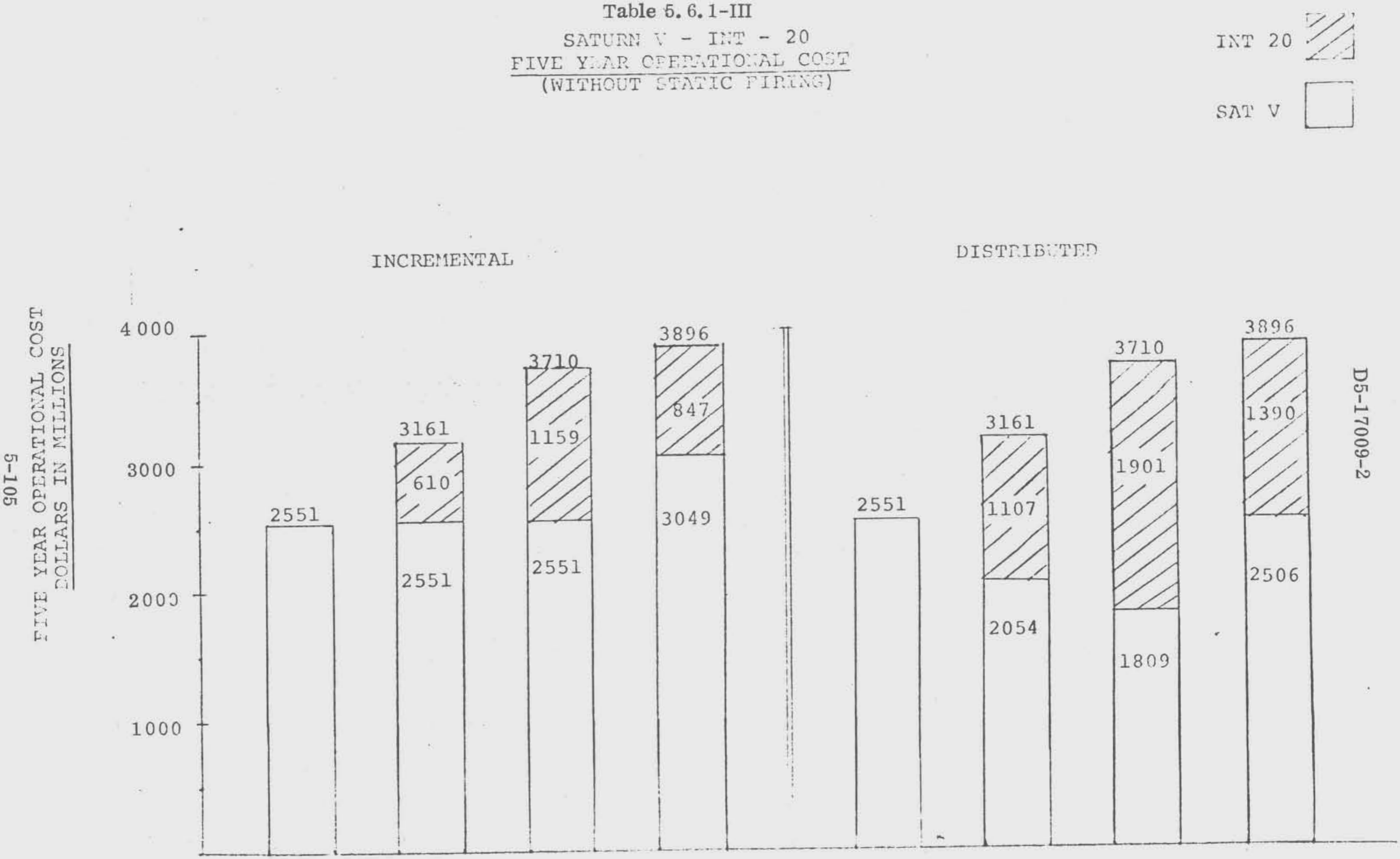
S-IC	1.00
S-IVB	2.94
I. U.	.01 (Above 5 Unit/Yr. Add 0.7)
F-1 Engine	.23
J-2 Engine	None
SE&I	.11
Total	4.29
KSC Facilities	3.20 (For 2 Saturn Vs + 4 INT-20s/ Yr. Add 1.6)
TOTAL	$7.49M

Table 5. 6. 1-III

SATURN V - INT - 20

FIVE YEAR OPERATIONAL COST

(WITHOUT STATIC FIRING)



	INCREMENTAL				DISTRIBUTED			
ANNUAL RATE (SAT V-INT 20)	2 SAT V	2+2	2+4	3+3	2 SAT V	2+2	2+4	3+3
AVG UNIT COST INT-20	-0-	61.0	58.0	56.[illegible]	-0-	110.7	95.1	92.7
AVG UNIT COST SAT V	255.1	255.1	255.1	203.3	255.1	205.5	190.9	167.1

Table 5.6.1-IV
SATURN V - INT - 20
FIVE YEAR OPERATIONAL COST
(WITH STATIC FIRING)

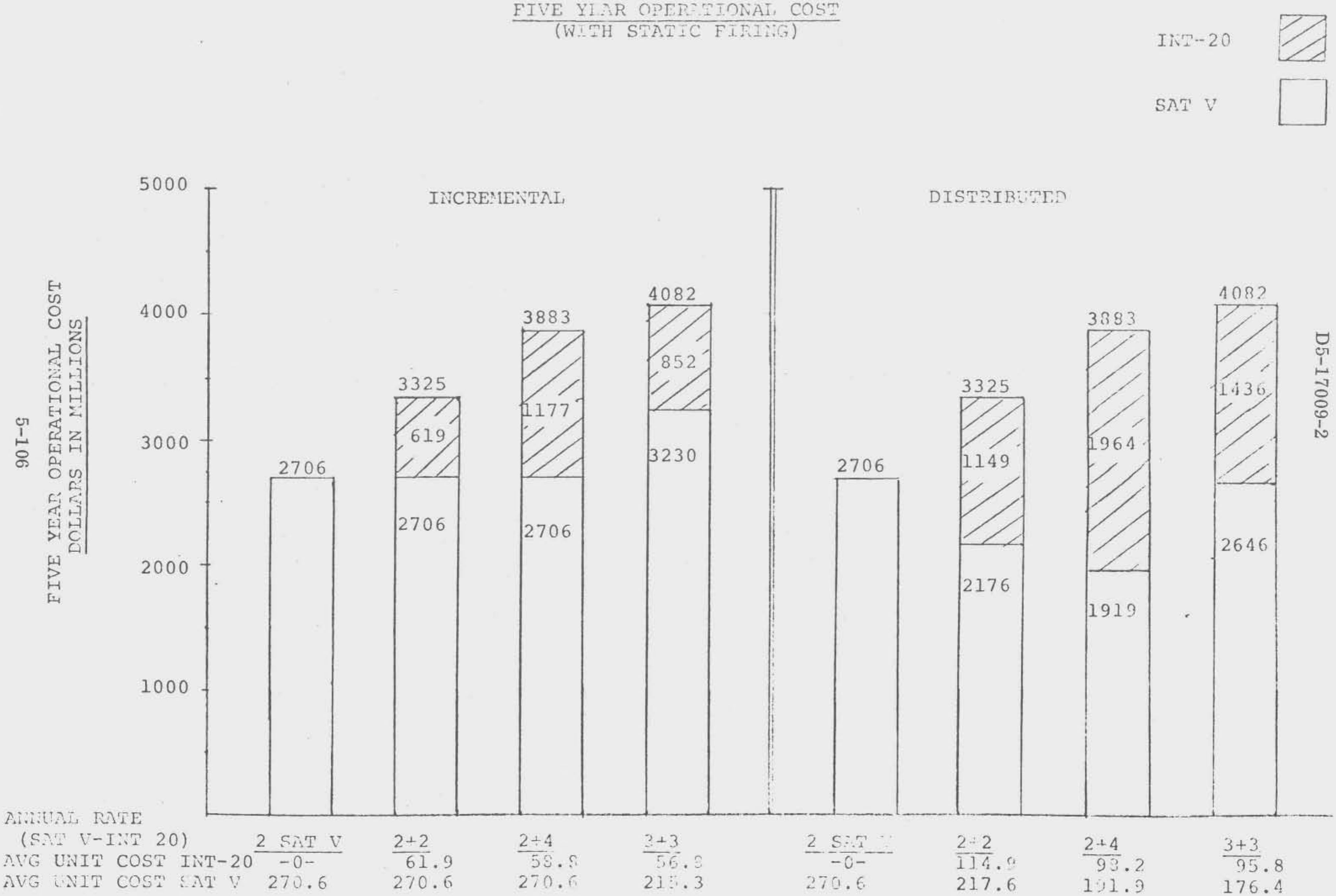


TABLE 5.6.1-V
AVERAGE UNIT COSTS (OPERATIONAL)

INCREMENTAL COST METHOD

5 Year Program Annual Launch Rate	With Static Firing		Without Static Firing	
	Saturn V Avg. Unit Cost	INT-20 Avg. Unit Cost	Saturn V Avg. Unit Cost	INT-20 Avg. Unit Cost
2 Saturn Vs	270.6	----	255.1	----
2 Saturn Vs+2 INT-20s	270.6	61.9	255.1	61.0
2 Saturn Vs+4 INT-20s	270.6	58.8	255.1	58.0
3 Saturn Vs	215.3	----	203.3	
3 Saturn Vs+3 INT-20s	215.3	56.8	203.3	56.5
2 INT-20s (No Saturn Vs)	----	192.6	- - - -	185.0
4 INT-20s (No Saturn Vs)	----	131.0	- - - -	126.8

DISTRIBUTED COST METHOD

5 Year Program Annual Launch Rate	With Static Firing		Without Static Firing	
	Saturn V Avg. Unit Cost	INT-20 Avg. Unit Cost	Saturn V Avg. Unit Cost	INT-20 Avg. Unit Cost
2 Saturn Vs	270.6	- - - -	255.1	- - - -
2 Saturn Vs+2 INT-20s	217.6	114.9	205.5	110.7
2 Saturn Vs+4 INT-20s	191.9	98.2	180.9	95.1
3 Saturn Vs	215.3	----	203.3	----
3 Saturn Vs+3 INT-20s	176.4	95.8	167.1	92.7
2 INT-20 (No Saturn Vs)	----	192.6	----	185.0
4 INT-20 (No Saturn Vs)	----	131.0	----	126.8

TABLE 5.6.1-VI
COST SUMMARY
S-IC/S-IVB/IU LAUNCH VEHICLES
(1968 DOLLARS IN MILLIONS)
WITHOUT STATIC FIRING

5 YEAR PROGRAM
2 SATURN V'S/YEAR

COST BREAKDOWN	DEVELOPMENT		OPERATIONAL		TOTAL
	STAGE	ENGINE	STAGE	ENGINE	
LAUNCH VEHICLE HARDWARE					
S-IC Stage			314.26	104.61	
S-II Stage			382.58	87.52	
S-IVB Stage			284.11	17.51	
Instrument Unit			93.46		
LAUNCH VEHICLE HARDWARE TOTAL			1074.41	209.64	1284.05
HARDWARE SUPPORT					
S-IC Stage			116.45	93.22	
S-II Stage			34.78	88.86	
S-IVB Stage			13.12	17.80	
Instrument Unit			7.09		
HARDWARE SUPPORT TOTAL			171.44	199.88	371.32
GROUND SUPPORT EQUIPMENT					
S-IC Stage					
GSE TOTAL					
FACILITIES					
Launch Vehicle - KSC					
FACILITIES TOTAL					
LAUNCH OPERATIONS			382.20		382.20
LAUNCH SUPPORT			460.32		460.32
INTEGRATION			53.01		53.01
SUB-TOTAL			2141.38	409.52	
LAUNCH SYSTEMS TOTAL			2550.90		2550.90
SATURN V PROGRAM					
TOTAL COST OF VEHICLES	(10 SAT V's)		2550.90		
AVERAGE UNIT COST	(10 OPER SAT V's)		255.09		255.09

COST DATA BASED ON NATIONAL SPACE BOOSTER STUDY, REF. PARA. 1.3.3-1

TABLE 5.6.1-VII
COST SUMMARY
S-IC/S-IVB/IU LAUNCH VEHICLES
(1968 DOLLARS IN MILLIONS)
WITHOUT STATIC FIRING

5 YEAR PROGRAM
2 SATURN V'S
2 INT'S/YR
INCREMENTAL

COST BREAKDOWN	DEVELOPMENT		OPERATIONAL		TOTAL
	STAGE	ENGINE	STAGE	ENGINE	
LAUNCH VEHICLE HARDWARE					
S-IC Stage	.78		488.04	179.25	
S-II Stage			382.58	99.17	
S-IVB Stage	2.94		404.55	19.83	
Instrument Unit	.01		165.50		
LAUNCH VEHICLE HARDWARE TOTAL	3.73		1440.67	298.25	1742.65
HARDWARE SUPPORT					
S-IC Stage	.16	.23	139.55	121.18	
S-II Stage			34.78	88.86	
S-IVB Stage			16.32	24.79	
Instrument Unit			7.44		
HARDWARE SUPPORT TOTAL	.16	.23	198.09	234.82	433.30
GROUND SUPPORT EQUIPMENT					
S-IC Stage	.06				
GSE TOTAL	.06				.06
FACILITIES					
Launch Vehicle - KSC	3.20		1.62		
FACILITIES TOTAL	3.20		1.62		4.82
LAUNCH OPERATIONS			411.64		411.64
LAUNCH SUPPORT			509.73		509.73
INTEGRATION	.11		66.36		66.47
SUB-TOTAL	7.26	.23	2628.11	533.07	
LAUNCH SYSTEMS TOTAL	7.49		3161.18		3168.67
SATURN V PROGRAM (2/YR)			2550.92		2550.92
TOTAL COST OF VEHICLES	(10 INT-20'S)		610.26		617.75
AVERAGE UNIT COST	(10 INT-20'S)		61.03		61.78

COST DATA BASED ON NATIONAL SPACE BOOSTER STUDY, REF. PARA. 1.3.3-1

TABLE 5.6.1-VIII

COST SUMMARY
S-IC/S-IVB/IU LAUNCH VEHICLES
(1968 DOLLARS IN MILLIONS)
WITHOUT STATIC FIRING

5 YEAR PROGRAM
2 SATURN V'S/YR
4 INT'S/YR
INCREMENTAL

COST BREAKDOWN	DEVELOPMENT		OPERATIONAL		TOTAL
	STAGE	ENGINE	STAGE	ENGINE	
LAUNCH VEHICLE HARDWARE					
S-IC Stage	.78		617.18	252.50	
S-II Stage			382.58	108.33	
S-IVB Stage	2.94		521.09	21.67	
Instrument Unit	.71		231.00		
LAUNCH VEHICLE HARDWARE TOTAL	4.43		1751.85	382.50	2138.78
HARDWARE SUPPORT					
S-IC Stage	.16	.23	162.66	149.12	
S-II Stage			34.78	88.86	
S-IVB Stage			19.52	31.78	
Instrument Unit			7.80		
HARDWARE SUPPORT TOTAL	.16	.23	224.76	269.76	494.91
GROUND SUPPORT EQUIPMENT					
S-IC Stage	.06				
GSE TOTAL	.06	-			.06
FACILITIES					
Launch Vehicle - KSC	4.80		1.62		
FACILITIES TOTAL	4.80		1.62		6.42
LAUNCH OPERATIONS			441.08		441.08
LAUNCH SUPPORT			559.14		559.14
INTEGRATION	.11		79.71		79.82
SUB-TOTAL	9.56	.23	3058.16	652.26	
LAUNCH SYSTEMS TOTAL	9.79		3710.42		3720.21
SATURN V PROGRAM (2/YR)			2550.92		2550.92
TOTAL COST OF VEHICLES	(20 INT-20's)		1159.50		1169.29
AVERAGE UNIT COST	(20 INT 20'S)		57.98		58.46

COST DATA BASED ON NATIONAL SPACE BOOSTER STUDY, REF. PARA. 1.3.3-1

TABLE 5.6.1-IX

COST SUMMARY
S-IC/S-IVB/IU LAUNCH VEHICLES
(1968 DOLLARS IN MILLIONS)
WITHOUT STATIC FIRING

5 YEAR PROGRAM
3 SATURN V'S/YEAR

COST BREAKDOWN	DEVELOPMENT		OPERATIONAL		TOTAL
	STAGE	ENGINE	STAGE	ENGINE	
LAUNCH VEHICLE HARDWARE					
S-IC Stage			415.75	152.50	
S-II Stage			462.50	118.75	
S-IVB Stage			346.00	23.75	
Instrument Unit			130.00		
LAUNCH VEHICLE HARDWARE TOTAL			1354.25	295.00	1649.25
HARDWARE SUPPORT					
S-IC Stage			119.26	109.83	
S-II Stage			39.16	104.09	
S-IVB Stage			14.92	20.90	
Instrument Unit			7.36		
HARDWARE SUPPORT TOTAL			180.70	234.82	415.52
GROUND SUPPORT EQUIPMENT					
S-IC Stage					
GSE TOTAL					
FACILITIES					
Launch Vehicle - KSC					
FACILITIES TOTAL					
LAUNCH OPERATIONS			411.64		411.64
LAUNCH SUPPORT			509.73		509.73
INTEGRATION			62.55		62.55
SUB-TOTAL			2518.87	529.82	
LAUNCH SYSTEMS TOTAL			3048.69		3048.69
SATURN V PROGRAM					
TOTAL COST OF VEHICLES	(15 SAT V's)		3048.69		
AVERAGE UNIT COST	(15 OPER SAT V's)		203.25		203.25

COST DATA BASED ON NATIONAL SPACE BOOSTER STUDY, REF. PARA. 1.3.3-1

TABLE 5.6.1-X
COST SUMMARY
S-IC/S-IVB/IU LAUNCH VEHICLES
(1968 DOLLARS IN MILLIONS)
WITHOUT STATIC FIRING

5 YEAR PROGRAM
3 SATURN V'S/YR
3 INT'S/YEAR
INCREMENTAL

COST BREAKDOWN	DEVELOPMENT		OPERATIONAL		TOTAL
	STAGE	ENGINE	STAGE	ENGINE	
LAUNCH VEHICLE HARDWARE					
S-IC Stage	.78		619.71	260.00	
S-II Stage			462.50	133.33	
S-IVB Stage	2.94		522.07	26.67	
Instrument Unit	.71		231.00		
LAUNCH VEHICLE HARDWARE TOTAL	4.43		1835.28	420.00	2259.71
HARDWARE SUPPORT					
S-IC Stage	.16	.23	162.66	151.36	
S-II Stage			39.16	104.09	
S-IVB Stage			19.52	31.78	
Instrument Unit			7.80		
HARDWARE SUPPORT TOTAL	.16	.23	229.14	287.23	516.76
GROUND SUPPORT EQUIPMENT					
S-IC Stage	.06				
GSE TOTAL	.06				.06
FACILITIES					
Launch Vehicle - KSC	3.20		2.53		
FACILITIES TOTAL	3.20		2.53		5.73
LAUNCH OPERATIONS			455.78		455.78
LAUNCH SUPPORT			583.85		583.85
INTEGRATION	.11		82.57		82.68
SUB-TOTAL	7.96	.23	3189.15	707.23	
LAUNCH SYSTEMS TOTAL	8.19		3896.38		3904.57
SATURN V PROGRAM (3/YR)			3048.69		3048.69
TOTAL COST OF VEHICLES	(15 INT-20'S)		847.69		855.88
AVERAGE UNIT COST	(15 INT-20'S)		56.51		57.06

COST DATA BASED ON NATIONAL SPACE BOOSTER STUDY, REF. PARA. 1.3.3-1

TABLE 5.6.1-XI
COST SUMMARY
S-IC/S-IVB/IU LAUNCH VEHICLES
(1968 DOLLARS IN MILLIONS)
(WITHOUT STATIC FIRING)

5 YEAR PROGRAM
2 INT-20'S/YEAR

COST BREAKDOWN	DEVELOPMENT		OPERATIONAL		TOTAL
	STAGE	ENGINE	STAGE	ENGINE	
LAUNCH VEHICLE HARDWARE					
S-IC Stage	.78		307.24	83.69	
S-IVB Stage	2.94		281.84	20.59	
Instrument Unit	.01		93.46		
LAUNCH VEHICLE HARDWARE TOTAL	3.73		682.54	104.28	790.55
HARDWARE SUPPORT					
S-IC Stage	.16	.23	116.45	93.22	
S-IVB Stage			13.12	17.80	
Instrument Unit			7.09		
HARDWARE SUPPORT TOTAL	.16	.23	136.66	111.02	248.07
GROUND SUPPORT EQUIPMENT					
S-IC Stage	.06				
GSE TOTAL	.06				.06
FACILITIES					
Launch Vehicle - KSC	3.20				
FACILITIES TOTAL	3.20				3.20
LAUNCH OPERATIONS			336.44		336.44
LAUNCH SUPPORT			441.60		441.60
INTEGRATION	.11		37.11		37.22
SUB-TOTAL	7.26	.23	1634.35	215.30	
LAUNCH SYSTEMS TOTAL	7.49		1849.65		1857.14
SATURN V PROGRAM					
TOTAL COST OF VEHICLES	(10 INT-20/YR)		1849.65		1857.14
AVERAGE UNIT COST	(10 INT-20's)		184.97		185.71

COST DATA BASED ON NATIONAL SPACE BOOSTER STUDY, REF. PARA. 1.3.3-1

TABLE 5.6.1-XII
COST SUMMARY
S-IC/S-IVB/IU LAUNCH VEHICLES
(1968 DOLLARS IN MILLIONS)
(WITHOUT STATIC FIRING)

5 YEAR PROGRAM
4 INT-20'S/YEAR

COST BREAKDOWN	DEVELOPMENT		OPERATIONAL		TOTAL
	STAGE	ENGINE	STAGE	ENGINE	
LAUNCH VEHICLE HARDWARE					
S-IC Stage	.78		482.13	161.25	
S-IVB Stage	2.94		402.51	41.18	
Instrument Unit	.01		165.50		
LAUNCH VEHICLE HARDWARE TOTAL	3.73		1050.14	202.43	1256.30
HARDWARE SUPPORT					
S-IC Stage	.16	.23	139.55	121.18	
S-IVB Stage			16.32	24.79	
Instrument Unit			7.44		
HARDWARE SUPPORT TOTAL	.16	.23	163.31	145.97	309.67
GROUND SUPPORT EQUIPMENT					
S-IC Stage	.06				
GSE TOTAL	.06				.06
FACILITIES					
Launch Vehicle - KSC	3.20				
FACILITIES TOTAL	3.20				3.20
LAUNCH OPERATIONS			392.26		392.26
LAUNCH SUPPORT			530.94		530.94
INTEGRATION	.11		50.46		50.57
SUB-TOTAL	7.26	.23	2187.11	348.40	
LAUNCH SYSTEMS TOTAL	7.49		2535.51		2543.00
SATURN V PROGRAM					
TOTAL COST OF VEHICLES	(4 INT-20's/YR)		2535.51		2543.00
AVERAGE UNIT COST	(20 INT-20'S)		126.78		127.15

COST DATA BASED ON NATIONAL SPACE BOOSTER STUDY, REF. PARA. 1.3.3-1

TABLE 5.6.1-XIII

COST SUMMARY

S-IC/S-IVB/IU LAUNCH VEHICLES

(1968 DOLLARS IN MILLIONS)

WITH STATIC FIRING

5 YEAR PROGRAM
2 SATURN V'S/YEAR

COST BREAKDOWN	DEVELOPMENT		OPERATIONAL		TOTAL
	STAGE	ENGINE	STAGE	ENGINE	
LAUNCH VEHICLE HARDWARE					
S-IC Stage			314.26	104.61	
S-II Stage			382.58	87.52	
S-IVB Stage			284.11	17.51	
Instrument Unit			93.46		
LAUNCH VEHICLE HARDWARE TOTAL			1074.41	209.64	1284.05
HARDWARE SUPPORT					
S-IC Stage			187.71	93.22	
S-II Stage			113.60	88.86	
S-IVB Stage			18.12	17.80	
Instrument Unit			7.09		
HARDWARE SUPPORT TOTAL			326.52	199.88	526.40
GROUND SUPPORT EQUIPMENT					
S-IC Stage					
GSE TOTAL					
FACILITIES					
Launch Vehicle - KSC					
FACILITIES TOTAL					
LAUNCH OPERATIONS			382.20		382.20
LAUNCH SUPPORT			460.32		460.32
INTEGRATION			53.03		53.03
SUB-TOTAL			2296.48	409.52	
LAUNCH SYSTEMS TOTAL			2706.00		2706.00
SATURN V PROGRAM					
TOTAL COST OF VEHICLES	(10 SAT V)		2706.00		
AVERAGE UNIT COST	(10 OPER SAT V)		270.60		270.60

COST DATA BASED ON NATIONAL SPACE BOOSTER STUDY, REF. PARA. 1.3.3-1

TABLE 5.6.1-XIV
COST SUMMARY
S-IC/S-IVB/IU LAUNCH VEHICLES
(1968 DOLLARS IN MILLIONS)
WITH STATIC FIRING

5 YEAR PROGRAM
2 SATURN V'S
2 INT'S/YEAR
INCREMENTAL

COST BREAKDOWN	DEVELOPMENT		OPERATIONAL		TOTAL
	STAGE	ENGINE	STAGE	ENGINE	
LAUNCH VEHICLE HARDWARE					
S-IC Stage	.78		488.04	179.25	
S-II Stage			382.58	99.17	
S-IVB Stage	2.94		404.55	19.83	
Instrument Unit	.01		165.50		
LAUNCH VEHICLE HARDWARE TOTAL	3.73		1440.67	298.25	1742.65
HARDWARE SUPPORT					
S-IC Stage	.16	.23	219.38	121.18	
S-II Stage			113.60	88.86	
S-IVB Stage			21.31	24.79	
Instrument Unit			7.44		
HARDWARE SUPPORT TOTAL	.16	.23	361.73	234.82	596.94
GROUND SUPPORT EQUIPMENT					
S-IC Stage	.06				
GSE TOTAL	.06				.06
FACILITIES					
Launch Vehicle - KSC	3.20		1.62		
FACILITIES TOTAL	3.20		1.62		4.82
LAUNCH OPERATIONS			411.64		411.64
LAUNCH SUPPORT			509.73		509.73
INTEGRATION	.11		66.36		66.47
SUB-TOTAL	7.26	.23	2791.75	533.07	
LAUNCH SYSTEMS TOTAL	7.49		3324.82		3332.31
SATURN V PROGRAM (2/YR)			2706.00		2706.00
TOTAL COST OF VEHICLES	(10 INT-20's)		618.82		626.31
AVERAGE UNIT COST	(10 INT-20's)		61.88		62.63

COST DATA BASED ON NATIONAL SPACE BOOSTER STUDY, REF. PARA. 1.3.3-1

TABLE 5.6.1-XV
COST SUMMARY
S-IC/S-IVB/IU LAUNCH VEHICLES
(1968 DOLLARS IN MILLIONS)
WITH STATIC FIRING

5 YEAR PROGRAM
2 SATURN V'S/YEAR
4 INT'S/YEAR
INCREMENTAL

COST BREAKDOWN	DEVELOPMENT		OPERATIONAL		TOTAL
	STAGE	ENGINE	STAGE	ENGINE	
LAUNCH VEHICLE HARDWARE					
S-IC Stage	.78		617.18	252.50	
S-II Stage			382.58	108.33	
S-IVB Stage	2.94		521.09	21.67	
Instrument Unit	.71		231.00		
LAUNCH VEHICLE HARDWARE TOTAL	4.43		1751.85	382.50	2138.78
HARDWARE SUPPORT					
S-IC Stage	.16	.23	251.07	149.12	
S-II Stage			113.60	88.86	
S-IVB Stage			24.49	31.78	
Instrument Unit			7.80		
HARDWARE SUPPORT TOTAL	.16	.23	396.96	269.76	667.11
GROUND SUPPORT EQUIPMENT					
S-IC Stage	.06				
GSE TOTAL	.06				.06
FACILITIES					
Launch Vehicle - KSC	4.80		1.62		
FACILITIES TOTAL	4.80		1.62		6.42
LAUNCH OPERATIONS			441.08		441.08
LAUNCH SUPPORT			559.14		559.14
INTEGRATION	.11		79.71		79.82
SUB-TOTAL	9.56	.23	3230.36	652.26	
LAUNCH SYSTEMS TOTAL	9.79		3882.62		3892.41
SATURN V PROGRAM (2/YR)			2706.00		2706.00
TOTAL COST OF VEHICLES	(20 INT-20's)		1176.62		1186.41
AVERAGE UNIT COST	(20 INT-20's)		58.83		59.32

COST DATA BASED ON NATIONAL SPACE BOOSTER STUDY, REF. PARA. 1.3.3-1

TABLE 5.6.1-XVI

COST SUMMARY

S-IC/S-IVB/IU LAUNCH VEHICLES

(1968 DOLLARS IN MILLIONS)

WITH STATIC FIRING

5 YEAR PROGRAM

3 SATURN V'S/YEAR

COST BREAKDOWN	DEVELOPMENT		OPERATIONAL		TOTAL
	STAGE	ENGINE	STAGE	ENGINE	
LAUNCH VEHICLE HARDWARE					
S-IC Stage			415.75	152.50	
S-II Stage			462.50	118.75	
S-IVB Stage			346.00	23.75	
Instrument Unit			130.00		
LAUNCH VEHICLE HARDWARE TOTAL			1354.25	295.00	1649.25
HARDWARE SUPPORT					
S-IC Stage			203.37	109.83	
S-II Stage			131.38	104.09	
S-IVB Stage			19.91	20.90	
Instrument Unit			7.26		
HARDWARE SUPPORT TOTAL			361.92	234.82	596.74
GROUND SUPPORT EQUIPMENT					
S-IC Stage					
GSE TOTAL					
FACILITIES					
Launch Vehicle - KSC					
FACILITIES TOTAL					
LAUNCH OPERATIONS			411.64		411.64
LAUNCH SUPPORT			509.73		509.73
INTEGRATION			62.55		62.55
SUB-TOTAL			2700.09	529.82	
LAUNCH SYSTEMS TOTAL			3229.91		3229.91
SATURN V PROGRAM					
TOTAL COST OF VEHICLES	(15 SAT V's)		3229.91		
AVERAGE UNIT COST	(15 OPER SAT V's)		215.33		215.33

COST DATA BASED ON NATIONAL SPACE BOOSTER STUDY, REF. PARA. 1.3.3-1

TABLE 5.6.1-XVII
COST SUMMARY
S-IC/S-IVB/IU LAUNCH VEHICLES
(1968 DOLLARS IN MILLIONS)
WITH STATIC FIRING

5 YEAR PROGRAM
3 SATURN V'S/YEAR
3 INT-S/YEAR
INCREMENTAL

COST BREAKDOWN	DEVELOPMENT		OPERATIONAL		TOTAL
	STAGE	ENGINE	STAGE	ENGINE	
LAUNCH VEHICLE HARDWARE					
S-IC Stage	.78		619.71	260.00	
S-II Stage			462.50	133.33	
S-IVB Stage	2.94		522.07	26.67	
Instrument Unit	.71		231.00		
LAUNCH VEHICLE HARDWARE TOTAL	4.43		1835.28	420.00	2259.71
HARDWARE SUPPORT					
S-IC Stage	.16	.23	251.07	151.36	
S-II Stage			131.38	104.09	
S-IVB Stage			24.49	31.78	
Instrument Unit			7.80		
HARDWARE SUPPORT TOTAL	.16	.23	414.74	287.23	702.36
GROUND SUPPORT EQUIPMENT					
S-IC Stage	.06				
GSE TOTAL	.06				.06
FACILITIES					
Launch Vehicle - KSC	3.20		2.53		
FACILITIES TOTAL	3.20		2.53		5.73
LAUNCH OPERATIONS			455.78		455.78
LAUNCH SUPPORT			583.85		583.85
INTEGRATION	.11		82.57		82.68
SUB-TOTAL	7.96	.23	3374.75	707.23	
LAUNCH SYSTEMS TOTAL	8.19		4081.98		4090,17
SATURN V PROGRAM (3/YR)			3229.91		3229.91
TOTAL COST OF VEHICLES	(15 INT-20's)		852.07		860.26
AVERAGE UNIT COST	(15 INT-20's)		56.80		57.35

COST DATA BASED ON NATIONAL SPACE BOOSTER STUDY, REF. PARA. 1.3.3-1

TABLE 5.6.1-XVIII
COST SUMMARY
S-IC/S-IVB/IU LAUNCH VEHICLES
(1968 DOLLARS IN MILLIONS)
WITH STATIC FIRING

5 YEAR PROGRAM
2 INT-20's/YEAR

COST BREAKDOWN	DEVELOPMENT		OPERATIONAL		TOTAL
	STAGE	ENGINE	STAGE	ENGINE	
LAUNCH VEHICLE HARDWARE					
S-IC Stage	.78		307.24	83.69	
S-IVB Stage	2.94		281.84	20.59	
Instrument Unit	.01		93.46		
LAUNCH VEHICLE HARDWARE TOTAL	3.73		682.54	104.28	790.55
HARDWARE SUPPORT					
S-IC Stage	.16	.23	187.71	93.22	
S-IVB Stage			18.12	17.80	
Instrument Unit			7.09		
HARDWARE SUPPORT TOTAL	.16	.23	212.92	111.02	324.33
GROUND SUPPORT EQUIPMENT					
S-IC Stage	.06				
GSE TOTAL	.06				.06
FACILITIES					
Launch Vehicle - KSC	3.20				
FACILITIES TOTAL	3.20				3.20
LAUNCH OPERATIONS			336.44		336.44
LAUNCH SUPPORT			441.60		441.60
INTEGRATION	.11		37.11		37.22
SUB-TOTAL	7.26	.23	1710.61	215.30	
LAUNCH SYSTEMS TOTAL	7.49		1925.91		1933.40
SATURN V PROGRAM					
TOTAL COST OF VEHICLES	(2 INT-20/YR)		1925.91		1933.40
AVERAGE UNIT COST	(10 INT-20's)		192.59		193.34

COST DATA BASED ON NATIONAL SPACE BOOSTER STUDY, REF. PARA. 1.3.3-1

TABLE 5.6.1-XIX

COST SUMMARY

S-IC/S-IVB/IU LAUNCH VEHICLES

(1968 DOLLARS IN MILLIONS)

WITH STATIC FIRING

5 YEAR PROGRAM
4 INT-20's/YEAR

COST BREAKDOWN	DEVELOPMENT		OPERATIONAL		TOTAL
	STAGE	ENGINE	STAGE	ENGINE	
LAUNCH VEHICLE HARDWARE					
S-IC Stage	.78		482.13	161.25	
S-IVB Stage	2.94		402.51	41.18	
Instrument Unit	.01		165.50		
LAUNCH VEHICLE HARDWARE TOTAL	3.73		1050.14	202.43	1256.30
HARDWARE SUPPORT					
S-IC Stage	.16	.23	219.38	121.18	
S-IVB Stage			21.31	24.79	
Instrument Unit			7.44		
HARDWARE SUPPORT TOTAL	.16	.23	248.13	145.97	394.49
GROUND SUPPORT EQUIPMENT					
S-IC Stage	.06				
GSE TOTAL	.06				.06
FACILITIES					
Launch Vehicle - KSC	3.20				
FACILITIES TOTAL	3.20				3.20
LAUNCH OPERATIONS			392.26		392.26
LAUNCH SUPPORT			530.94		530.94
INTEGRATION	.11		50.46		50.57
SUB-TOTAL	7.26	.23	2271.93	348.40	
LAUNCH SYSTEMS TOTAL	7.49		2620.33		2627.82
SATURN V PROGRAM					
TOTAL COST OF VEHICLES	(4 INT-20's/YR)		2620.33		2627.82
AVERAGE UNIT COST	(20 INT-20's)		131.02		131.39

COST DATA BASED ON NATIONAL SPACE BOOSTER STUDY, REF. PARA. 1.3.3-1

TABLE 5.6.1-XX

COST SUMMARY
S-1C/S-1VB/IU LAUNCH VEHICLES
(1968 DOLLARS IN MILLIONS)
(WITHOUT STATIC FIRING)

5 Year Program
2 Saturn V's/Year
2 INT-20's/Year

DISTRIBUTED

COST BREAKDOWN	DEVELOPMENT STAGE	DEVELOPMENT ENGINE	OPERATIONAL *SAT V'S	OPERATIONAL *INT-20'S	TOTAL OPER
LAUNCH VEHICLE HARDWARE					
S-1C Stage	.78		246.98	241.06	
S-11 Stage			382.58	-0-	
S-1VB Stage	2.94		203.30	201.25	
Instrument Unit	.01		82.75	82.75	
			201.59	96.66	
Total Hardware	3.73		1117.20	621.72	1738.92
HARDWARE SUPPORT					
Engine		.23	108.98	61.59	
Stage	.16		167.61	94.73	
Total Hardware Support	.16	.23	276.59	156.32	432.91
GROUND SUPPORT EQUIPMENT					
Stage	.06				
Total GSE	.06				
FACILITIES					
Launch Vehicle - KSC	3.20			1.62	
Total Facilities	3.20			1.62	1.62
LAUNCH OPERATIONS			274.42	137.22	411.64
LAUNCH SUPPORT			339.82	169.91	509.73
INTEGRATION	.11		46.46	19.90	66.36
LAUNCH SYSTEMS TOTAL	7.26	.23			
	7.49		2054.48	1106.70	3161.18
AVERAGE UNIT COST (OPER)			205.45	110.67	
AVERAGE UNIT COST INT-20 (PROGRAM)				111.42	

COST DATA BASED ON NATIONAL SPACE BOOSTER STUDY, REFERENCE 1.3.3-1.

* TWO

TABLE 5.6.1-XXI

COST SUMMARY
S-1C/S-1VB/IU LAUNCH VEHICLES
(1968 DOLLARS IN MILLIONS)
(WITHOUT STATIC FIRING)

5 Year Program
2 Saturn V's/Year
4 INT-20's/Year

DISTRIBUTED

COST BREAKDOWN	DEVELOPMENT		OPERATIONAL		TOTAL OPER
	STAGE	ENGINE	*SAT V'S	*INT-20'S	
LAUNCH VEHICLE HARDWARE					
S-1C Stage	.78		209.13	408.05	
S-11 Stage			382.58	-0-	
S-1VB Stage	2.94		175.00	346.09	
Instrument Unit	.71		77.00	154.00	
			194.61	187.89	
Total Hardware	4.43		1038.32	1096.03	2134.35
HARDWARE SUPPORT					
Engine		.23	90.33	109.46	
Stage	.16		133.26	161.47	
Total Hardware Support	.16	.23	223.59	270.93	494.52
GROUND SUPPORT EQUIPMENT					
Stage	.06				
Total GSE	.06				
FACILITIES					
Launch Vehicle - KSC	4.80			1.62	
Total Facilities	4.80			1.62	1.62
LAUNCH OPERATIONS			220.54	220.54	441.08
LAUNCH SUPPORT			279.57	279.57	559.14
INTEGRATION	.11		46.89	32.82	79.71
	9.56	.23			
LAUNCH SYSTEMS TOTAL	9.79		1808.92	1901.50	3710.42
AVERAGE UNIT COST (OPER)			180.89	95.08	
AVERAGE UNIT COST INT-20 (PROGRAM)				95.56	

COST DATA BASED ON NATIONAL SPACE BOOSTER STUDY, REFERENCE 1.3.3-1.

* TWO

TABLE 5.6.1-XXII
COST SUMMARY
S-1C/S-1VB/IU LAUNCH VEHICLES
(1968 DOLLARS IN MILLIONS)
(WITHOUT STATIC FIRING)

5 Year Program
3 Saturn V's/Year
3 INT-20's/Year

COST BREAKDOWN	DEVELOPMENT STAGE	DEVELOPMENT ENGINE	OPERATIONAL *SAT V'S	OPERATIONAL *INT-20'S	DISTRIBUTED TOTAL OPER
LAUNCH VEHICLE HARDWARE					
S-1C Stage	.78		313.69	306.01	
S-11 Stage			462.50	-0-	
S-1VB Stage	2.94		262.50	259.57	
Instrument Unit	.71		115.50	115.50	
			281.58	138.41	
Total Hardware	4.43		1435.78	819.50	2255.28
HARDWARE SUPPORT					
Engine		.23	134.66	76.53	
Stage	.16		194.59	110.59	
Total Hardware Support	.16	.23	329.25	187.12	516.37
GROUND SUPPORT EQUIPMENT					
Stage	.06				
Total GSE	.06				
FACILITIES					
Launch Vehicle - KSC	3.20			2.53	
Total Facilities	3.20			2.53	2.53
LAUNCH OPERATIONS			303.85	151.93	455.78
LAUNCH SUPPORT			389.23	194.62	583.85
INTEGRATION	.11		48.57	34.00	82.57
LAUNCH SYSTEMS TOTAL	7.96	.23			
	8.19		2506.68	1389.70	3896.38
AVERAGE UNIT COST (OPER)			167.11	92.65	
AVERAGE UNIT COST INT-20 (PROGRAM)				93.19	

COST DATA BASED ON NATIONAL SPACE BOOSTER STUDY, REFERENCE 1.3.3-1.

* THREE

TABLE 5.6.1-XXIII

COST SUMMARY
S-1C/S-1VB/IU LAUNCH VEHICLES
(1968 DOLLARS IN MILLIONS)
(WITH STATIC FIRING)

5 Year Program
2 Saturn V's/Year
2 INT-20's/Year

DISTRIBUTED

COST BREAKDOWN	DEVELOPMENT STAGE	DEVELOPMENT ENGINE	OPERATIONAL *SAT V'S	OPERATIONAL *INT-20'S	TOTAL OPER
LAUNCH VEHICLE HARDWARE					
S-1C Stage	.78		246.98	241.06	
S-11 Stage			382.58		
S-1VB Stage	2.94		203.30	201.25	
Instrument Unit	.01		82.75	82.75	
Engines			201.59	96.66	
Total Hardware	3.73		1117.20	621.72	1738.92
HARDWARE SUPPORT					
Engine		.23	156.54	78.28	
Stage	.16		241.28	120.45	
Total Hardware Support	.16	.23	397.82	198.73	596.55
GROUND SUPPORT EQUIPMENT					
Stage	.06				
Total GSE	.06				
FACILITIES					
Launch Vehicle - KSC	3.20			1.62	
Total Facilities	3.20			1.62	1.62
LAUNCH OPERATIONS			274.42	137.22	411.64
LAUNCH SUPPORT			339.82	169.91	509.73
INTEGRATION	.11		46.46	19.90	66.36
	7.26	.23			
LAUNCH SYSTEMS TOTAL	7.49		2175.72	1149.10	3324.82
AVERAGE UNIT COST (OPER)			217.57	114.91	
AVERAGE UNIT COST INT-20 (PROGRAM)				115.66	

COST DATA BASED ON NATIONAL SPACE BOOSTER STUDY, REFERENCE 1.3.3-1
* Two

TABLE 5.6.1-XXIV

COST SUMMARY
S-1C/S-1VB/IU LAUNCH VEHICLES
(1968 DOLLARS IN MILLIONS)
(WITH STATIC FIRING)

5 Year Program
2 Saturn V's/Year
4 INT-20's/Year

DISTRIBUTED

COST BREAKDOWN	DEVELOPMENT STAGE	DEVELOPMENT ENGINE	OPERATIONAL *SAT V'S	OPERATIONAL **INT-20'S	TOTAL OPER
LAUNCH VEHICLE HARDWARE					
S-1C Stage	.78		209.13	408.05	
S-11 Stage			382.58		
S-1VB Stage	2.94		175.00	346.09	
Instrument Unit	.71		77.00	154.00	
Engines			194.61	187.89	
Total Hardware	4.43		1038.32	1096.03	2134.35
HARDWARE SUPPORT					
Engine		.23	134.88	134.70	
Stage	.16		198.66	198.48	
Total Hardware Support	.16	.23	333.54	333.18	666.72
GROUND SUPPORT EQUIPMENT					
Stage	.06				
Total GSE	.06				
FACILITIES					
Launch Vehicle - KSC	4.80			1.62	
Total Facilities	4.80			1.62	1.62
LAUNCH OPERATIONS			220.54	220.54	441.08
LAUNCH SUPPORT			279.57	279.57	559.14
INTEGRATION	.11		46.89	32.82	79.71
LAUNCH SYSTEMS TOTAL	9.56	.23			
	9.79		1918.86	1963.76	3882.62
AVERAGE UNIT COST (OPER)			191.89	98.19	
AVERAGE UNIT COST INT-20 (PROGRAM)				98.68	

COST DATA BASED ON NATIONAL SPACE BOOSTER STUDY, REFERENCE 1.3.3-1

* Two
** Four

TABLE 5.6.1-XXV

COST SUMMARY
S-1C/S-1VB/IU LAUNCH VEHICLES
(1968 DOLLARS IN MILLIONS)
(WITH STATIC FIRING)

5 Year Program
3 Saturn V's/Year
3 INT-20's/Year

DISTRIBUTED

COST BREAKDOWN	DEVELOPMENT STAGE	DEVELOPMENT ENGINE	OPERATIONAL *SAT V'S	OPERATIONAL *INT-20'S	TOTAL OPER
LAUNCH VEHICLE HARDWARE					
S-1C Stage	.78		313.69	306.02	
S-11 Stage			462.50		
S-1VB Stage	2.94		262.50	259.57	
Instrument Unit	.71		115.50	115.50	
Engines			281.59	138.41	
Total Hardware	4.43		1435.78	819.50	2255.28
HARDWARE SUPPORT					
Engine		.23	191.40	95.70	
Stage	.16		276.57	138.30	
Total Hardware Support	.16	.23	467.97	234.00	701.97
GROUND SUPPORT EQUIPMENT					
Stage	.06				
Total GSE	.06				
FACILITIES					
Launch Vehicle - KSC	3.20			2.53	
Total Facilities	3.20			2.53	2.53
LAUNCH OPERATIONS			303.85	151.93	455.78
LAUNCH SUPPORT			389.23	194.62	583.85
INTEGRATION	.11		48.57	34.00	82.57
	7.96	.23			
LAUNCH SYSTEMS TOTAL	8.19		2645.59	1436.66	4081.98
AVERAGE UNIT COST (OPER)			176.37	95.78	
AVERAGE UNIT COST INT-20 (PROGRAM)				96.32	

COST DATA BASED ON NATIONAL SPACE BOOSTER STUDY, REFERENCE 1.3.3-1
* Three

TABLE 5.6.1-XXVI FISCAL FUNDING DISTRIBUTION, 2 SATURN Vs + 2 INT-20s/YEAR
INCREMENTAL COST METHOD

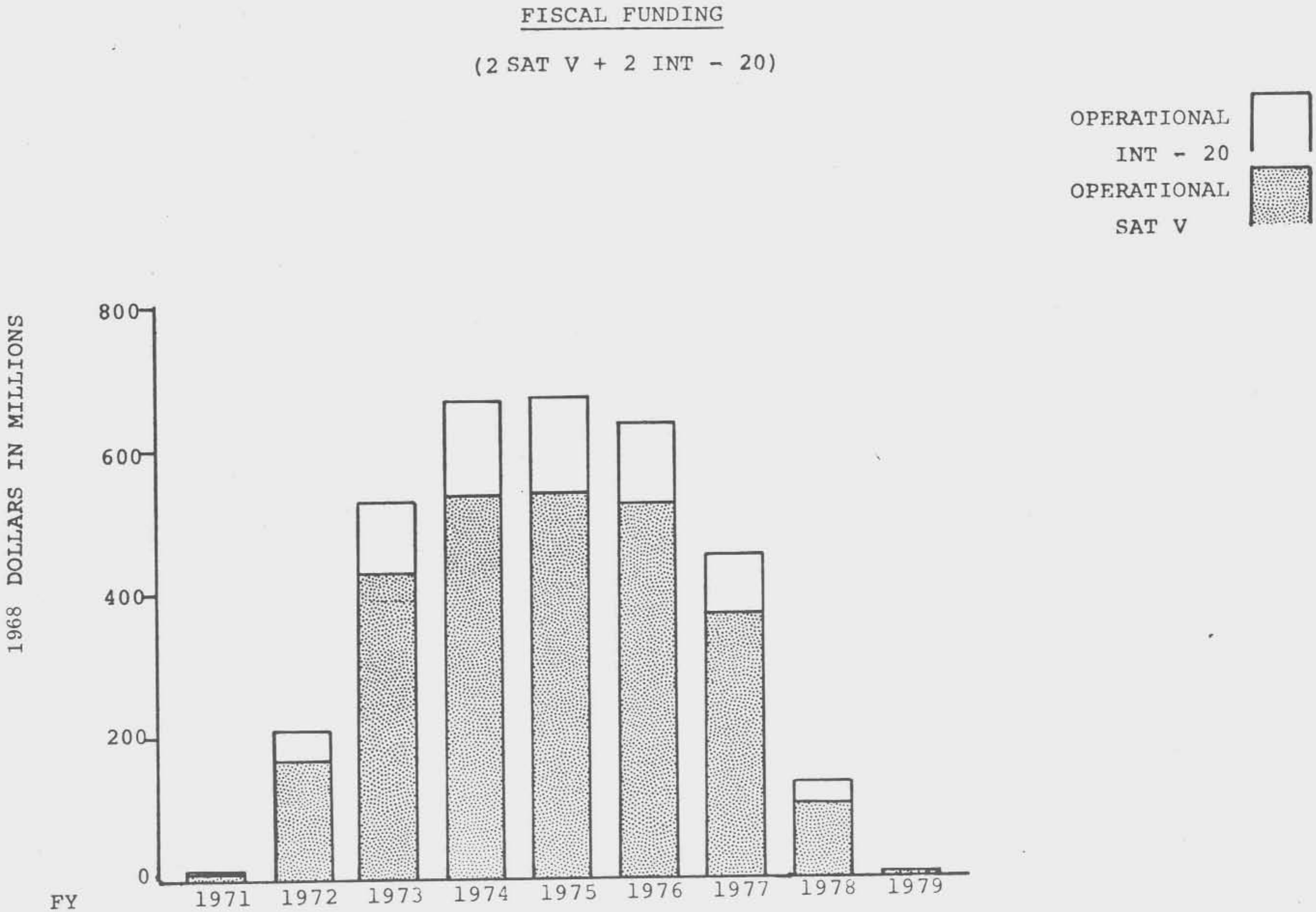


TABLE 5.6.1-XXVII FISCAL FUNDING DISTRIBUTION, 2 SATURN Vs + 4 INT-20s/YEAR
INCREMENTAL COST METHOD

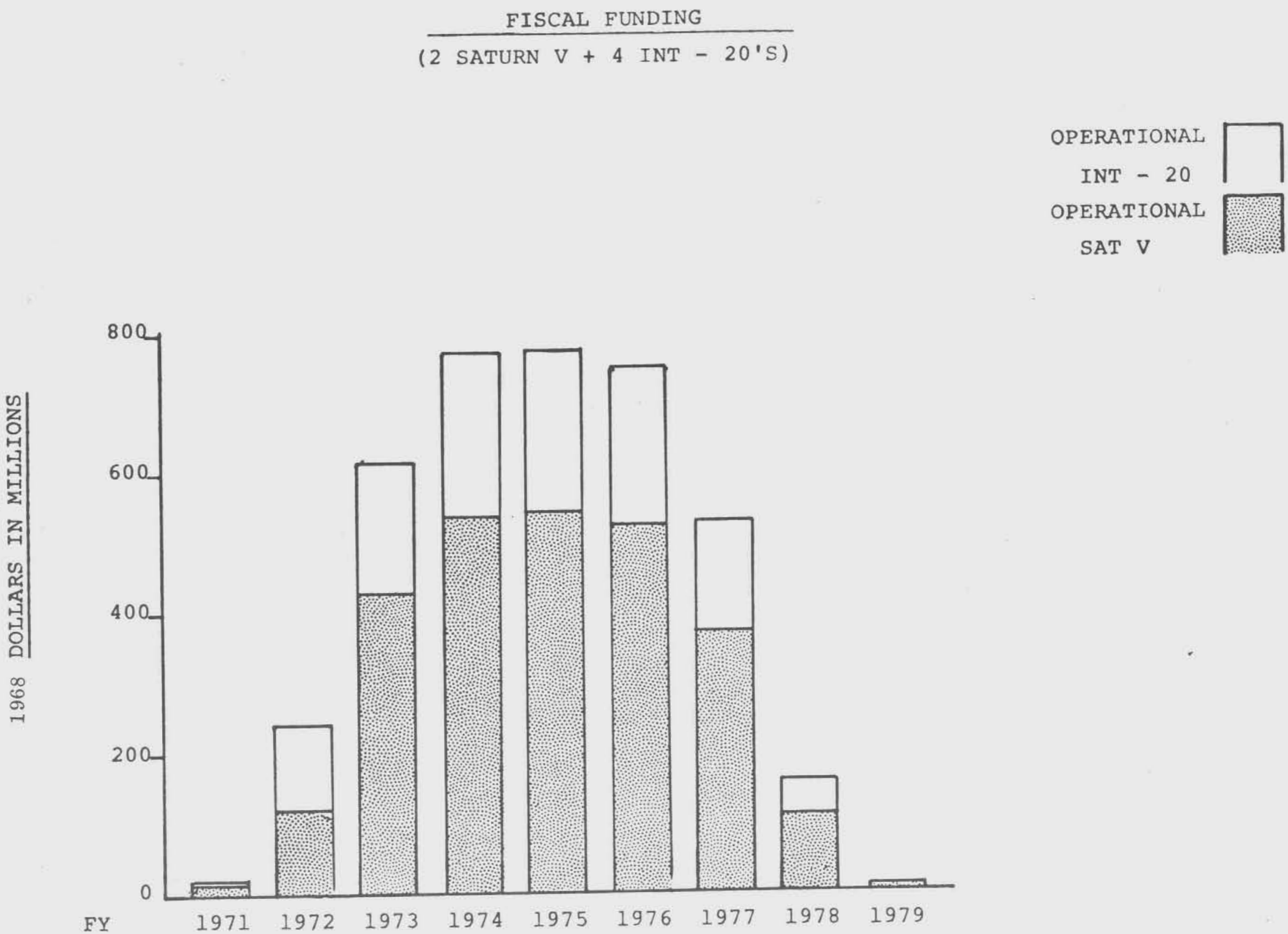


TABLE 5.6.1-XXVIII FISCAL FUNDING DISTRIBUTION, 3 SATURN Vs + 3 INT-20s/YEAR
INCREMENTAL COST METHOD

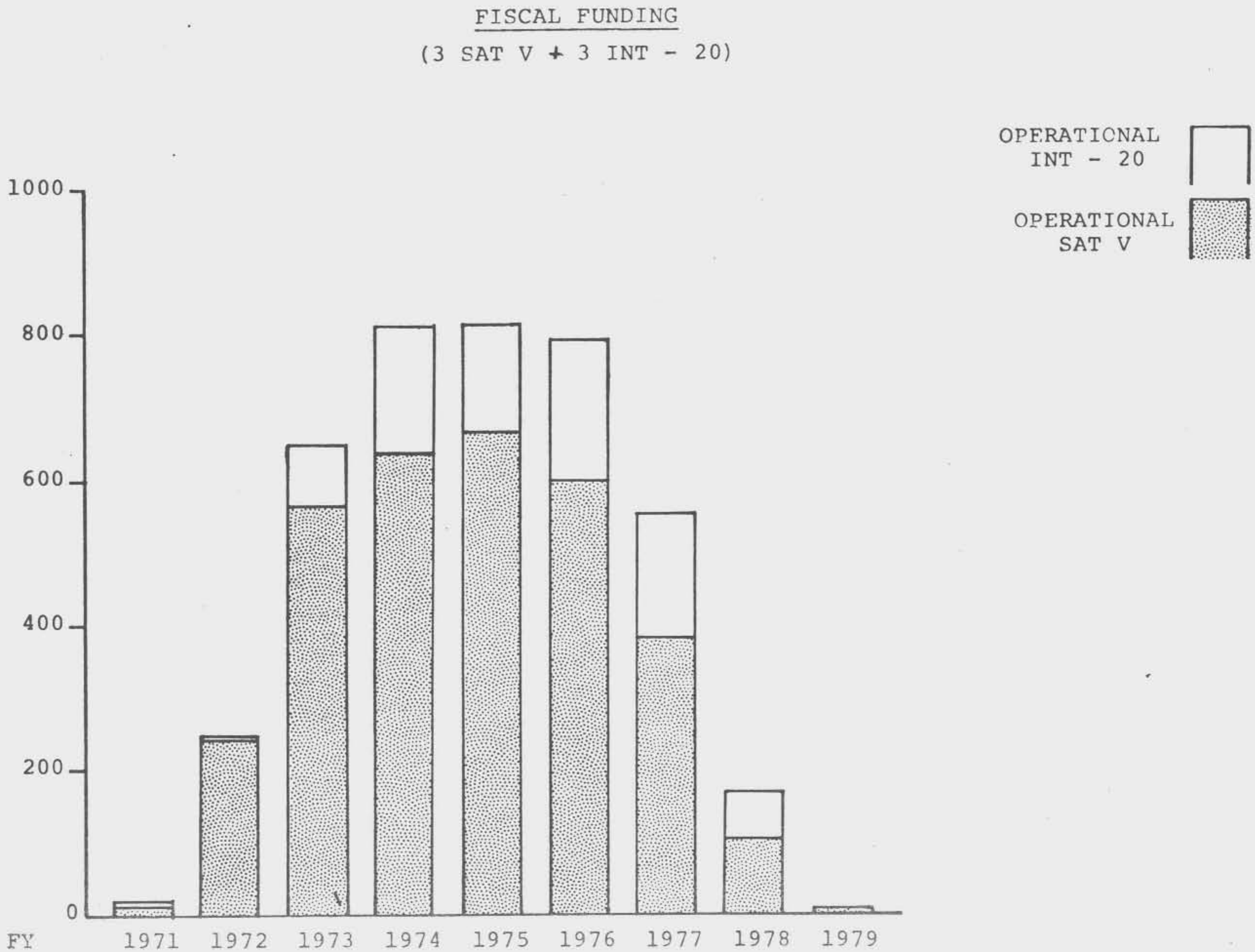


TABLE 5.6.1-XXIX FISCAL FUNDING DISTRIBUTION 2 INT-20s (NO SATURN Vs)/YEAR

FISCAL FUNDING
(2 INT 20 + 0 SAT V's)

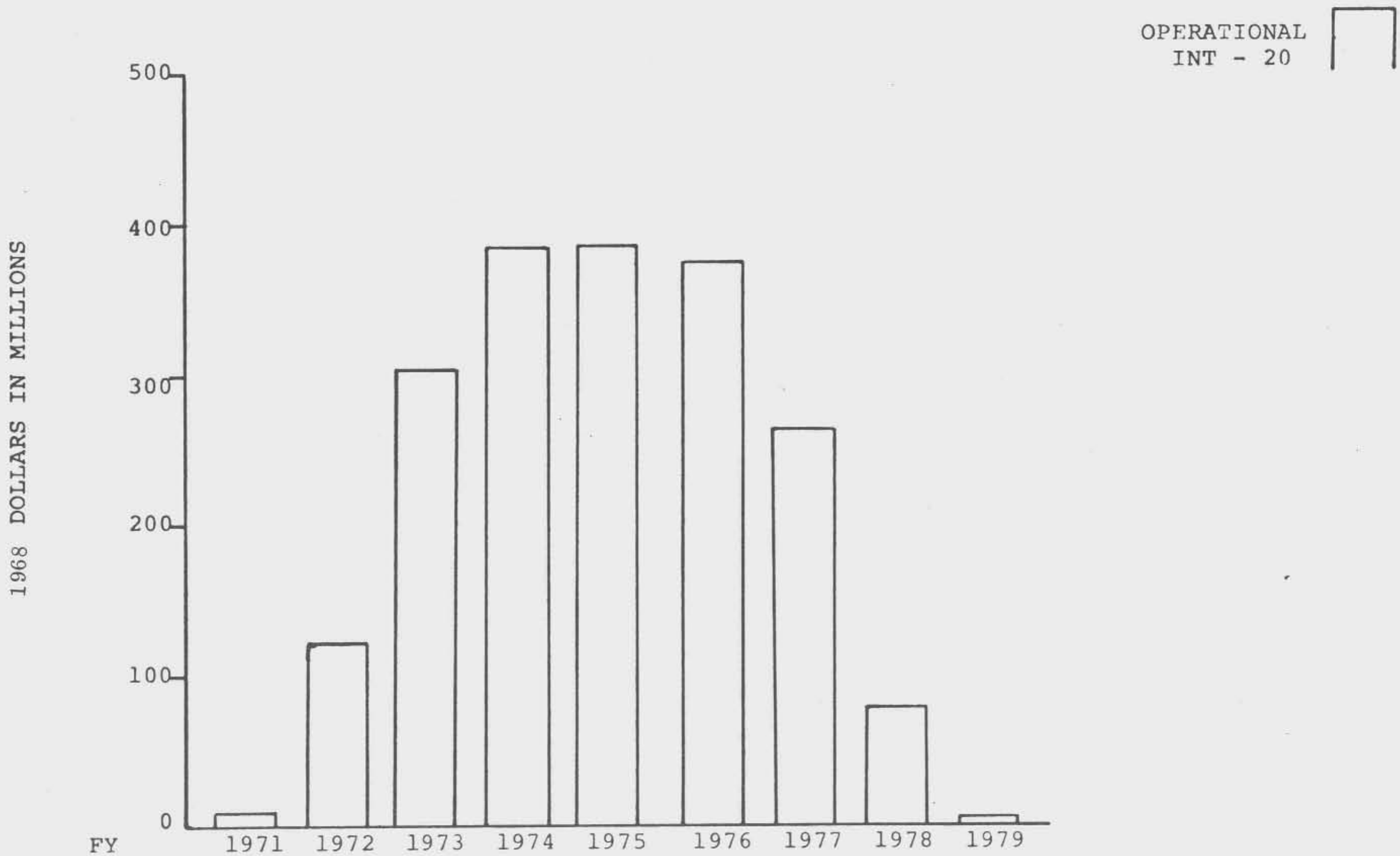
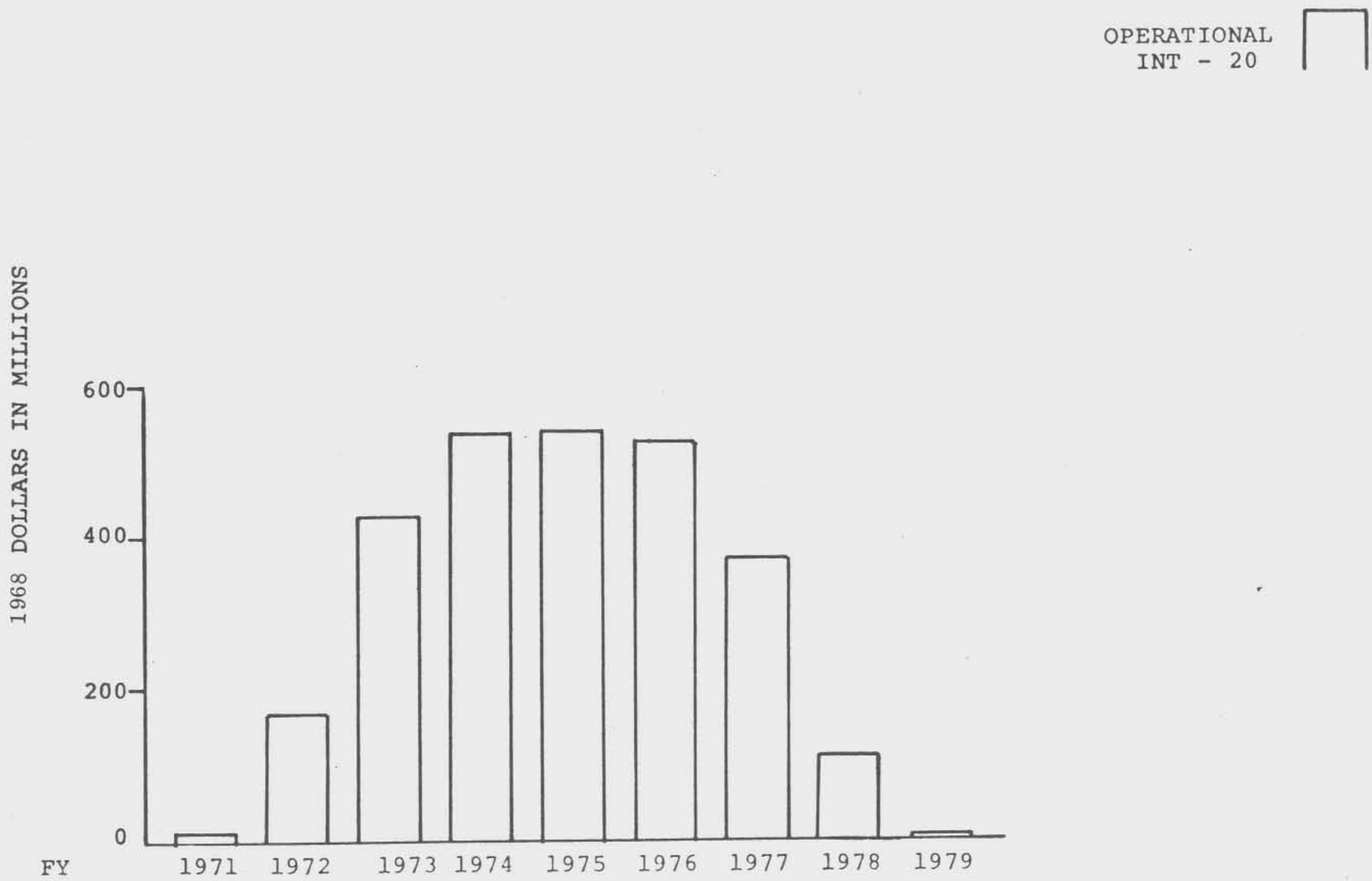


TABLE 5.6.1-XXX FISCAL FUNDING DISTRIBUTION, 4 INT-20s (NO SATURN Vs)/YEAR

FISCAL FUNDING
(4 INT-20'S + NO SAT V'S)



5.6.2 S-IC Cost Plan

5.6.2.1 Non-Recurring or Developmental Price

The development cost estimates are shown on Table 5.6.2-I and were based on 1968 dollars and rates.

5.6.2.2 Recurring - Delta Reduction Price

This planning estimate is based upon the following:

a. 1968 dollars and rates were used in the preparation of this estimate.

b. Estimated delta reductions were measured from the prices contained in the National Space Booster Study (contract NASW-1740, October 3, 1968) for various delivery rates for the S-IC/SAT V.

c. Estimated delta reductions were based on the following programs:

   1. 10 S-IC/SAT V's and 10 S-IC/INT-20's at a four-per-year delivery rate.

   2. 10 S-IC/SAT V's and 20 S-IC/INT-20's at a six-per-year delivery rate.

   3. 15 S-IC/SAT V's and 15 S-IC/INT-20's at a six-per-year delivery rate.

   4. 10 S-IC/INT-20's and no S-IC/SAT V's at a two-per-year delivery rate.

   5. 20 S-IC/INT-20's and no S-IC/SAT V's at a four-per-year delivery rate.

d. Assume that all startup and reactivation costs will be absorbed by the follow-on program for two S-IC/SAT V's, Stages S-IC-16 and S-IC-17.

e. Assume that the various delivery rates would have been attained on Stages S-IC-16 and S-IC-17.

Tables 5.6.2-II through -VI reflect the estimated delta reductions for the five programs listed in paragraph 5.6.2.2.

TABLE 5.8.2-1 DEVELOPMENT COSTS

S-IC/S-IVB/IU LAUNCH VEHICLES 5 YEAR PROGRAM

S-IC/INT-20 STAGE

(DOLLARS IN THOUSANDS)

COST BREAKDOWN	ENGRG.	TOOLING	MANUFACTURING	TEST	Q&RA	MAT'L	MISC.	TOTAL
STAGE HARDWARE								
Structure	$ 21	$ 6	$--	$ --	$ 1	$26	$ 1	$ 55
Sub-Sys. Instal.	108	22	--	251	67	3	23	474
Electrical	35	--	--	5	1	4	2	47
Environmental Control	--	--	--	--	--	--	--	--
Flight Control	--	--	--	--	--	--	--	--
Guidance & Navigation	--	--	--	--	--	--	--	--
Instrumentation	76	--	--	12	3	8	5	104
Ordnance Subsystem	--	--	--	--	--	--	--	--
Pressurization Sys.	17	--	--	3	1	3	1	25
Propulsion Sys. (Less Engine)	15	--	--	9	2	50	1	77
Stage GSE	--	--	--	--	--	--	--	--
STAGE TOTAL	$272	$28	$--	$280	$75	$94	$33	$782
Engines & Acces. Total	--	--	--	--	--	--	--	--
HARDWARE SUPPORT								
* Itemize:								
- Attach Separate List								
HARDWARE SUPPORT TOTAL	$ 58	$ 6	$--	$ 58	$15	$15	$ 8	$160
GROUND SUPPORT EQUIPMENT								
Test and Checkout	$ 20	$--	$--	$ 19	$ 4	$13	$ 2	$ 58
Transp. and Handling	2	--	--	--	--	--	--	2
Other								
GSE TOTAL	$ 22	$--	$--	$ 19	$ 4	$13	$ 2	$ 60
FACILITIES								
Test								
Manufacturing								
KSC								
FACILITIES TOTAL								

Note: *Itemize such items as - Transportation, Communications, Computer Services, Propellants & Pressurants, Range & Base Services, Etc.

Costs should include burden and fee

TABLE 5.6.2-II

DELTA OPERATIONAL COSTS*
S-IC/S-IVB/IU LAUNCH VEHICLES
S-IC STAGE
(DOLLARS IN THOUSANDS)

5 YR PROGRAM
2 SATURN V/S/YR 518 THROUGH 527
2 INT'S/YR 1 THROUGH 10

COST BREAKDOWN	ENGRG.	TOOLING	MANUFACTURING	TEST	CAPE	MATL	MISC.	TOTAL
STAGE HARDWARE								
Structure	$--	$--	$ 6	$--	$ 2	$ 80	$ 1	$ 89
Sub-Sys. Instal.	--	19	498	49	148	--	32	748
Electrical	--	--	6	--	2	50	1	59
Environmental Control	--	--	--	--	--	--	--	--
Flight Control	--	--	--	--	--	--	--	--
Guidance & Navigation	--	--	--	--	--	--	--	--
Instrumentation	--	--	12	--	3	50	2	67
Ordnance Subsystem	--	--	--	--	--	--	--	--
Pressurization Sys.	--	--	6	--	2	100	1	109
Propulsion Sys. (Less Engine)	--	--	58	5	18	4,761	4	4,846
Stage GSE	--	--	--	--	--	--	--	--
STAGE TOTAL	$--	$19	$586	$54	$175	$5,041	$41	$5,916
Engines & Acces. Total	--	--	--	--	--	--	--	--
HARDWARE SUPPORT								
* Itemize:								
Attach Separate List								
HARDWARE SUPPORT TOTAL	$--	$ 4	$129	$ 9	$ 36	$ --	$ 7	$ 185
GROUND SUPPORT EQUIPMENT								
Test and Checkout								
Transp. and Handling								
Other								
GSE TOTAL	$--	$--	$ --	$--	$ --	$ --	$--	$ --
FACILITIES								
Test								
Manufacturing								
KSC								
FACILITIES TOTAL	$--	$--	$ --	$--	$ --	$ --	$--	$ --

Note: * Itemize such items as - Transportation, Communications, Computer Services, Propellant & Pressurants, Range Base Services, Etc.

Costs should include burden and fee.

TABLE 5.6.2-III

DELTA OPERATIONAL COSTS
S-IC/S-IVB/IU LAUNCH VEHICLES
S-IC STAGE
(DOLLARS IN THOUSANDS)

5 YEAR PROGRAM
2 SATURN V'S/YEAR 518 THROUGH 527
4 INT'S/YR 1 THROUGH 20

COST BREAKDOWN	ENGRG.	TOOLING	MANUFACTURING	TEST	Q&RA	MATL	MISC.	TOTAL
STAGE HARDWARE								
Structure	$--	$--	$ 11	$ --	$ 3	$ 160	$ 1	$ 175
Sub-Sys. Instal.	--	38	967	97	290	--	66	1,458
Electrical	--	--	11	--	3	85	1	100
Environmental Control	--	--	--	--	--	--	--	--
Flight Control	--	--	--	--	--	--	--	--
Guidance & Navigation	--	--	--	--	--	--	--	--
Instrumentation	--	--	22	--	7	85	2	116
Ordnance Subsystem	--	--	--	--	--	--	--	--
Pressurization Sys.	--	--	11	--	3	200	1	215
Propulsion Sys. (Less Engine)	--	--	114	10	34	7,980	8	8,146
Stage GSE	--	--	--	--	--	--	--	--
STAGE TOTAL	$--	$38	$1,136	$107	$340	$8,510	$79	$10,210
Engines & Acces. Total	--	--	--	--	--	--	--	--
HARDWARE SUPPORT								
* Itemize:								
Attach Separate List								
HARDWARE SUPPORT TOTAL	$--	$ 8	$ 248	$ 19	$ 69	$ --	$13	$ 357
GROUND SUPPORT EQUIPMENT								
Test and Checkout								
Transp. and Handling								
Other								
GSE TOTAL	$--	$--	$ --	$ --	$ --	$ --	$--	$ --
FACILITIES								
Test								
Manufacturing								
KSC								
FACILITIES TOTAL	$--	$--	$ --	$ --	$ --	$ --	$--	$ --

Note: *Itemize such items as - Transportation, Communications Computer Services, Propellant & Pressurants, Range & Base Services, Etc.

Costs should include burden and fee.

TABLE 5.6.2-IV

DELTA OPERATIONAL COSTS
S-IC/S-IVB/IU LAUNCH VEHICLES
S-IC STAGE
(DOLLARS IN THOUSANDS)

5 YEAR PROGRAM
3 SATURN V/S/YR 519 THROUGH 533
3 INT'S/YEAR 1 THROUGH 15

COST BREAKDOWN	ENGRG.	TOOLING	MANUFACTURING	TEST	Q&RA	MATL	MISC.	TOTAL
STAGE HARDWARE								
Structure	$--	$--	$ 9	$--	$ 3	$ 120	$ 1	$ 133
Sub-Sys. Instal.	--	29	741	71	220	--	49	1,110
Electrical	--	--	9	--	3	64	1	77
Environmental Control	--	--	--	--	--	--	--	--
Flight Control	--	--	--	--	--	--	--	--
Guidance & Navigation	--	--	--	--	--	--	--	--
Instrumentation	--	--	17	--	5	64	2	88
Ordnance Subsystem	--	--	--	--	--	--	--	--
Pressurization Sys.	--	--	9	--	3	150	1	163
Propulsion Sys. (Less Engine)	--	--	87	8	26	5,982	6	6,109
Stage GSE	--	--	--	--	--	--	--	--
STAGE TOTAL	$--	$29	$872	$79	$260	$6,380	$60	$7,680
Engines & Acces. Total	--	--	--	--	--	--	--	--
HARDWARE SUPPORT								
* Itemize:								
Attach Separate List								
HARDWARE SUPPORT TOTAL	$--	$ 6	$191	$15	$ 52	$ --	$11	$ 275
GROUND SUPPORT EQUIPMENT								
Test and Checkout								
Transp. and Handling								
Other								
GSE TOTAL	$--	$--	$ --	$--	$ --	$ --	$--	$ --
FACILITIES								
Test								
Manufacturing								
KSC								
FACILITIES TOTAL	$--	$--	$ --	$--	$ --	$ --	$--	$ --

Note: *Itemize such items as - Transportation, Communications, Computer Services, Propellant & Pressurants, Range & Base Services, Etc.

Costs should include burden and fee.

TABLE 5.6.2-V

DELTA OPERATIONAL COSTS
S-IC/S-IVB/IU LAUNCH VEHICLES
S-IC STAGE
(DOLLARS IN THOUSANDS)

5 YEAR PROGRAM
2 INT-20'S/YEAR
1 THROUGH 10

COST BREAKDOWN	ENGRG.	TOOLING	MANUFACTURING	TEST	Q&RA	MAT'L	MISC.	TOTAL
STAGE HARDWARE								
Structure	$--	$--	$ 6	$--	$ 2	$ 80	$ 1	$ 89
Sub-Sys. Instal.	--	20	544	49	166	--	36	815
Electrical	--	--	6	--	2	60	1	69
Environmental Control	--	--	--	--	--	--	--	--
Flight Control	--	--	--	--	--	--	--	--
Guidance & Navigation	--	--	--	--	--	--	--	--
Instrumentation	--	--	12	--	4	60	2	78
Ordnance Subsystem	--	--	--	--	--	--	--	--
Pressurization Sys.	--	--	6	--	2	100	1	109
Propulsion Sys. (Less Engine)	--	--	64	5	20	5,767	4	5,860
Stage GSE	--	--	--	--	--	--	--	--
STAGE TOTAL	$--	$20	$638	$54	$196	$6,067	$45	$7,020
Engines & Acces. Total	--	--	--	--	--	--	--	--
HARDWARE SUPPORT								
* Itemize:								
Attach Separate List								
HARDWARE SUPPORT TOTAL	$--	$ 4	$138	$ 9	$ 39	$ --	$ 7	$ 197
GROUND SUPPORT EQUIPMENT								
Test and Checkout								
Transp. and Handling								
Other								
GSE TOTAL	$--	$--	$ --	$--	$ --	$ --	$--	$ --
FACILITIES								
Test								
Manufacturing								
KSC								
FACILITIES TOTAL	$--	$--	$ --	$--	$ --	$ --	$--	$ --

Note: *Itemize such items as - Transportation, Communications, Computer Services, Propellant & Pressurants, Range & Base Services, Etc.

Costs should include burden and fee.

TABLE 5.6.2-VI

DELTA OPERATIONAL COSTS
S-IC/S-IVB/IU LAUNCH VEHICLES
S-IC STAGE
(DOLLARS IN THOUSANDS)

5 YEAR PROGRAM
4 INT-20'S/YEAR
1 THROUGH 20

COST BREAKDOWN	ENGRG.	TOOLING	MANUFACTURING	TEST	Q&RA	MATL	MISC.	TOTAL
STAGE HARDWARE								
Structure	$--	$--	$ 12	$ --	$ 4	$ 160	$ 1	$ 177
Sub-Sys. Instal.	--	40	998	95	296	--	68	1,497
Electrical	--	--	12	--	4	100	1	117
Environmental Control	--	--	--	--	--	--	--	--
Flight Control	--	--	--	--	--	--	--	--
Guidance & Navigation	--	--	--	--	--	--	--	--
Instrumentation	--	--	23	--	7	100	2	132
Ordnance Subsystem	--	--	--	--	--	--	--	--
Pressurization Sys.	--	--	12	--	4	200	1	217
Propulsion Sys. (Less Engine)	--	--	117	11	35	9,520	8	9,691
Stage GSE	--	--	--	--	--	--	--	--
STAGE TOTAL	$--	$40	$1,174	$106	$350	$10,080	$81	$11,831
Engines & Acces. Total	--	--	--	--	--	--	--	--
HARDWARE SUPPORT								
* Itemize:								
Attach Separate List								
HARDWARE SUPPORT TOTAL	$--	$ 9	$ 256	$ 19	$ 71	$ --	$15	$ 370
GROUND SUPPORT EQUIPMENT								
Test and Checkout								
Transp. and Handling								
Other								
GSE TOTAL	$--	$--	$ --	$ --	$ --	$ --	$--	$ --
FACILITIES								
Test								
Manufacturing								
KSC								
FACILITIES TOTAL	$--	$--	$ --	$ --	$ --	$ --	$--	$ --

Note: *Itemize such items as - Transportation, Communications, Computer Services, Propellant & Pressurants, Range & Base Services, Etc.

Costs should include burden and fee.

5.6.3 S-IVB Stage Cost Plan

The attached cost estimates have been prepared in accordance with the appropriate INT-20 study ground rules to reflect changes in the Saturn V/S-IVB stage which would be required to implement the INT-20 (S-IC/S-IVB/IU) launch vehicle configuration. Development and operational costs have been estimated for both a baseline and an alternate S-IVB stage configuration in accordance with the definitions of these two configurations as described in Sections 4.2.4 and 4.3.3. The development costs represent the total non-recurring effort required to design, test, tool for and plan the stage hardware changes. The operational costs in all cases represent an incremental reduction in recurring costs due to the deletion of various stage hardware components and installations. In addition, the operational costs are further itemized to reflect the changes due to five alternative five year program plans.

The cost estimates have been based on detail estimates supplied by the appropriate engineering, testing and manufacturing personnel who are closely associated with S-IVB development and production activities. These detail estimates have been factored as necessary to conform to standard bid factors being used in S-IVB contract pricing. In addition, learning curve factors and production rate factors were applied to the operational labor cost estimates. A 90% learning curve was used for quantity extensions and rate factors of 1.40 and 0.84 were applied to the 2/year and 6/year cases respectively.

5.6.3.1 Baseline Configuration Costs

Costs for the INT-20/S-IVB stage are presented in Tables 5.6.3-I through 5.6.3-VI for the baseline configuration. Recurring costs are presented in terms of deltas from the Saturn V/S-IVB stage. Table 5.6.3-I presents INT-20/S-IVB stage non-recurring costs for the design, tooling, production planning, qualification testing of critical components and revising of check-out procedures as discussed in previous sections. The largest single non-recurring cost item is the potential requalification of selected critical components due to the expected higher acoustic levels environment anticipated for the S-IVB stage on the INT-20 vehicle compared to the Saturn V vehicle. Although the requirement for this requalification cannot be specifically verified at this time (see Section 5.2.3) it is included in the cost because of its significant impact. Other non-recurring costs are rather minor, including engineering revision to existing production drawings and test procedures, new tooling for the new mating ring for the S-IVB/S-IC interface and planning for in-line deletions of S-IVB components for the INT-20 configuration. There will be no static firing costs for the INT-20/S-IVB stage and therefore no delta costs are indicated for hardware support or GSE.

Facilities and GSE cost impacts are expected to be zero or negligible. An exception to this is the costs identified for new and modified KSC facilities/GSE in the INT-20 Facilities and Operations study (NAS10-6163) as documented in Boeing Report D5-16785. Since those costs are well documented in the referenced report, are primarily vehicle rather than stage oriented, and vary over many options, they are not repeated here.

Table 5.6.3-I

BASELINE INT-20/S-IVB STAGE DEVELOPMENT COSTS

Costs Include Burden and Fee

Cost Breakdown	(Dollars in Thousands)							
	Engr	Tool	Manuf	Test	Q&RA	Matl	Misc	Total
Stage Hardware	---	---	---	2734*	---	---	---	2734
Structure	17	32	2	---	---	---	---	51
Subsystems installation	39	2	3	---	---	---	---	44
Electrical								
Environmental control	7	---	---	---	---	---	---	7
Flight control								
Guidance and navigation								
Instrumentation	12	---	---	14	---	---	---	26
Ordnance subsystem								
Propulsion/pressurization system (less engine)	64	---	11	6	---	---	---	81
Stage GSE								
STAGE TOTAL	139	34	16	2754	---	---	---	2943
ENGINE AND ACCES. TOTAL								0
Hardware Support								
Itemized on separate list as required								
HARDWARE SUPPORT TOTAL								0
Ground Support Equipment								
Test and checkout								
Transportation and handling								
Other								
GSE TOTAL								0
Facilities								
Test								
Manufacturing								
KSC								
FACILITIES TOTAL								0

*Qual test of selected critical components.

Table 5.6.3-II

BASELINE INT-20/S-IVB STAGE DELTA OPERATIONAL COSTS

Costs Include Burden and Fee **( ) Decrease in Standard Sat. V Stage Costs**

Cost Breakdown	(Dollars in Thousands)							
	Engr	Tool	Manuf	Test	Q&RA	Matl	Misc	Total
Stage Hardware								
Structure	---	---	(126)	---	(52)	---	---	(178)
Subsystems installation	---	---	(46)	---	(19)	---	---	(65)
Electrical								
Environmental control								
Flight control								
Guidance and navigation								
Instrumentation	---	---	---	---	---	(347)	---	(347)
Ordnance subsystem								
Propulsion/pressurization system (less engine)	---	---	(197)	---	(82)	(1179)	---	(1458)
Stage GSE								
STAGE TOTAL	---	---	(369)	---	(153)	(1526)	---	(2048)
ENGINE AND ACCES. TOTAL								0
Hardware Support								
Itemized on separate list as required								
HARDWARE SUPPORT TOTAL								0
Ground Support Equipment								
Test and checkout								
Transportation and handling								
Other								
GSE TOTAL								0
Facilities								
Test								
Manufacturing								
KSC								
FACILITIES TOTAL								0

5 Year Program: 2 Sat. V/yr (518-527); 2 INT-20/yr (1-10).

Table 5.6.3-III

BASELINE INT-20/S-IVB STAGE DELTA OPERATIONAL COSTS

Costs Include Burden and Fee — ( ) Decrease in Standard Sat. V Stage Costs

Cost Breakdown	(Dollars in Thousands) Engr	Tool	Manuf	Test	Q&RA	Matl	Misc	Total
Stage Hardware								
Structure	---	---	(207)	---	(86)	---	---	(293)
Subsystems installation	---	---	(76)	---	(31)	---	---	(107)
Electrical								
Environmental control								
Flight control								
Guidance and navigation								
Instrumentation	---	---	---	---	---	(694)	---	(694)
Ordnance subsystem								
Propulsion/pressurization system (less engine)	---	---	(324)	---	(135)	(2357)	---	(2816)
Stage GSE								
STAGE TOTAL	---	---	(607)	---	(252)	(3051)	---	(3910)
ENGINE AND ACCES. TOTAL								
Hardware Support								
Itemized on separate list as required								
HARDWARE SUPPORT TOTAL								0
Ground Support Equipment								
Test and checkout								
Transportation and handling								
Other								
GSE TOTAL								0
Facilities								
Test								
Manufacturing								
KSC								
FACILITIES TOTAL								0

5 Year Program: 2 Sat. V/yr (518-527); 4 INT-20/yr (1-20)

Table 5.6.3-IV

BASELINE INT-20/S-IVB STAGE DELTA OPERATIONAL COSTS

Costs Include Burden and Fee — ( ) Decrease in Standard Sat. V Stage Costs

Cost Breakdown	(Dollars in Thousands) Engr	Tool	Manuf	Test	Q&RA	Matl	Misc	Total
Stage Hardware								
Structure	---	---	(154)	---	(64)	---	---	(218)
Subsystems installation	---	---	(56)	---	(23)	---	---	(79)
Electrical								
Environmental control								
Flight control								
Guidance and navigation								
Instrumentation	---	---	---	---	---	(520)	---	(520)
Ordnance subsystem								
Propulsion/pressurization system (less engine)	---	---	(242)	---	(101)	(1768)	---	(2111)
Stage GSE								
STAGE TOTAL	---	---	(452)	---	(188)	(2288)	---	(2928)
ENGINE AND ACCES. TOTAL								0
Hardware Support								
Itemized on separate list as required								
HARDWARE SUPPORT TOTAL								0
Ground Support Equipment								
Test and checkout								
Transportation and handling								
Other								
GSE TOTAL								0
Facilities								
Test								
Manufacturing								
KSC								
FACILITIES TOTAL								0

5 Year Program: 3 Sat. V/yr (519-533); 3 INT-20/yr (1-15)

Table 5.6.3-V

BASELINE INT-20/S-IVB STAGE DELTA OPERATIONAL COSTS

Costs Include Burden and Fee — ( ) Decrease in Standard Sat. V Stage Costs

Cost Breakdown	(Dollars in Thousands)							
	Engr	Tool	Manuf	Test	Q&RA	Matl	Misc	Total
Stage Hardware								
Structure	---	---	(180)	---	(75)	---	---	(255)
Subsystems installation	---	---	(66)	---	(27)	---	---	(93)
Electrical								
Environmental control								
Flight control								
Guidance and navigation								
Instrumentation	---	---	---	---	---	(347)	---	(347)
Ordnance subsystem								
Propulsion/pressurization system (less engine)	---	---	(282)	---	(117)	(1179)	---	(1578)
Stage GSE								
STAGE TOTAL	---	---	(528)	---	(219)	(1526)	---	(2273)
ENGINE AND ACCES. TOTAL								0
Hardware Support								
Itemized on separate list as required								
HARDWARE SUPPORT TOTAL								0
Ground Support Equipment								
Test and checkout								
Transportation and handling								
Other								
GSE TOTAL								0
Facilities								
Test								
Manufacturing								
KSC								0
FACILITIES TOTAL								0

5 Year Program: 2 INT-20/yr (1-10)

Table 5.6.3-VI

BASELINE INT-20/S-IVB STAGE DELTA OPERATIONAL COSTS

Costs Include Burden and Fee — ( ) Decrease in Standard Sat. V Stage Costs

Cost Breakdown	(Dollars in Thousands)							
	Engr	Tool	Manuf	Test	Q&RA	Matl	Misc	Total
Stage Hardware								
Structure	---	---	(250)	---	(104)	---	---	(354)
Subsystems installation	---	---	(91)	---	(38)	---	---	(129)
Electrical								
Environmental control								
Flight control								
Guidance and navigation								
Instrumentation	---	---	---	---	---	(694)	---	(694)
Ordnance subsystem								
Propulsion/pressurization system (less engine)								
Stage GSE			(392)		(163)	(2357)		(2912)
STAGE TOTAL	---	---	(733)	---	(305)	(3051)	---	(4089)
ENGINE AND ACCES. TOTAL								0
Hardware Support								
Itemized on separate list as required								
HARDWARE SUPPORT TOTAL								0
Ground Support Equipment								
Test and checkout								
Transportation and handling								
Other								
GSE TOTAL								0
Facilities								
Test								
Manufacturing								
KSC								
FACILITIES TOTAL								0

5 Year Program: 4 INT-20/yr (1-20)

Recurring costs for the INT-20/S-IVB baseline stage are presented in Tables 5.6.3-II through 5.6.3-VI in terms of deltas from the operational costs for the standard Saturn V/S-IVB stage. The costs of the standard S-IVB stages and operations must be added to these INT-20/S-IVB costs to obtain total costs. These costs would have to be re-examined for the low-cost S-IVB stage (i.e., in that case they are not additional). INT-20 baseline S-IVB cost impact results in a total decrease of from \$2 to \$4 million (depending on the program) over the five year operational program life compared to a Saturn V/S-IVB stage. This decrease is all due to deletions in fabrication, installations, and purchased parts. No net change is expected in support, GSE, or Facilities recurring costs.

5.6.3.2 Interface Option Cost Trade

Two basic options were considered for effecting S-IC/S-IVB mating, the first involving direct interface between the S-IVB aft interstage and the S-IC forward skirt (with a new bolt hole pattern), and the second a scheme whereby an adapter ring would be used to connect the two structures utilizing their existing bolt hole patterns. (See detailed discussion in Section discussion in Section 4.2.3.) To assist in evaluating the two options, a cost trade investigation was made. The results of the trade are shown on Figure 5.6.3-1, and indicate that with concurrent Saturn V production, the total program costs of the adapter ring concept were less than that for the direct interface concept up to a total of 22 vehicles. The corresponding quantity trade point for the case of no concurrent Saturn V production was 17 units. The difference was due to the intermittent set-up time requirements for using a common tool. Although a higher recurring cost per vehicle results for the adapter ring, the much higher development costs involved with generating a new master gage and transfer gages for the direct interface concept result in higher production costs.

5.6.3.3 Interstage Option Cost Trade

Two interstage options are available as a result of the S-IVB retrorocket requirements being deleted, one involving making no structural changes to the interstage (i.e., leaving the retrorocket provisions - fittings, intercostals and fairings - as is) and the second involving deletion of all such provisions. (See discussion in Section 4.2.4.2.) The results of the trade are indicated on Figure 5.6.3-2, which shows that a net cost decrease occurs for the interstage (structure) due to deletion of the rocket provisions. The cost savings attributed to the retrorockets themselves are not shown since they apply in either case, and are reflected in the propulsion material cost decrease. Thus, since the non-recurring costs involved with either option were minimal, it appeared advantageous to delete the retrorocket provisions in INT-20 interstage production.

5.6.3.4 Alternate Configuration Costs

Costs for the alternate INT-20/S-IVB configuration (maximum deletions) are presented in Tables 5.6.3-VII through 5.6.3-XII. The results are similar to those for the baseline INT-20/S-IVB configuration. Potential qualification testing of selected critical components dominates non-recurring

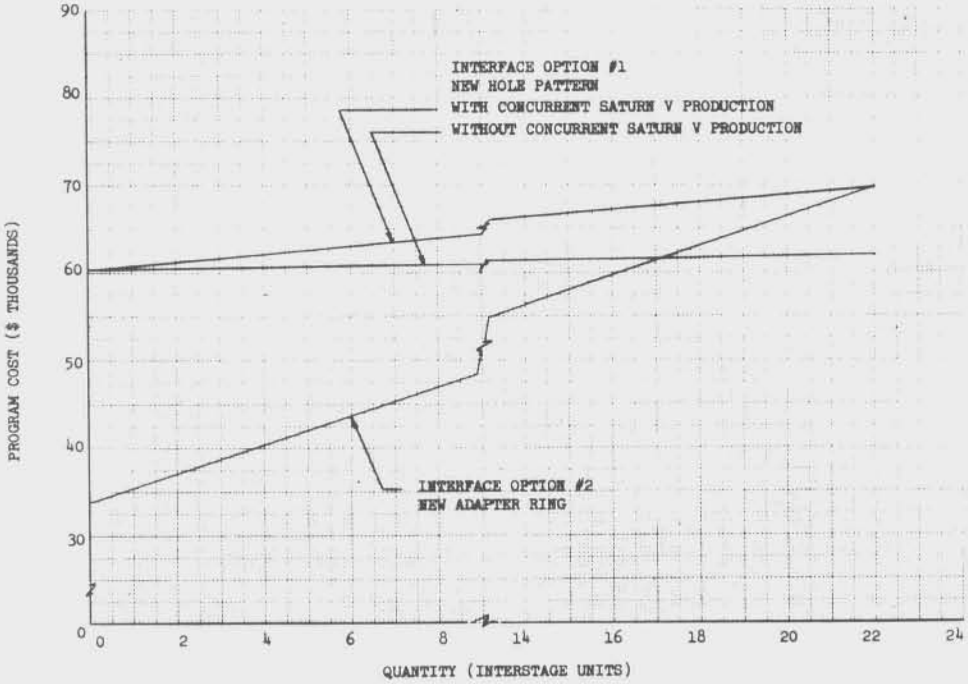


Figure 5.6.3-1. INT-20 Interstage Interface Cost Trade

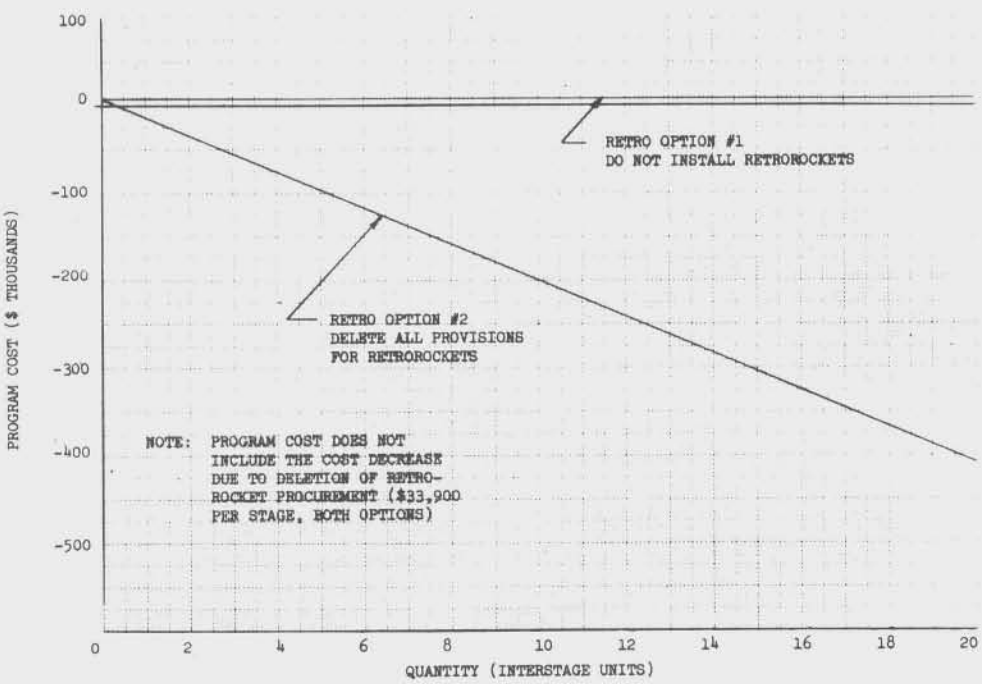


Figure 5.6.3-2. INT-20 Interstage Retro Deletion Cost Trade

Table 5.6.3-VII

ALTERNATE INT-20/S-IVB STAGE DEVELOPMENT COSTS

Costs Include Burden and Fee

Cost Breakdown	(Dollars in Thousands) Engr	Tool	Manuf	Test	Q&RA	Matl	Misc	Total
Stage Hardware	---	---	---	2734**	---	---	---	2,734
Structure	22	32	2	---	---	---	---	56
Subsystems installation	61	2	5	---	---	---	---	68
Electrical								
Environmental control	7	---	---	---	---	---	---	7
Flight control								
Guidance and navigation								
Instrumentation	21	---	---	14	---	---	---	35
Ordnance subsystem								
Propulsion/pressurization system (less engine)	83	---	14	6	---	---	---	103
Stage GSE								
STAGE TOTAL	194	34	21	2754	---	---	---	3,003
ENGINE AND ACCES. TOTAL								0
Hardware Support								
Itemized on separate list as required								
HARDWARE SUPPORT TOTAL								0
Ground Support Equipment								
Test and checkout								
Transportation and handling								
Other								
GSE TOTAL								0
Facilities								
Test								
Manufacturing								
KSC								
FACILITIES TOTAL								0

**Qual test of selected critical components.

Table 5.6.3-VIII

ALTERNATE INT-20/S-IVB STAGE DELTA OPERATIONAL COSTS

Costs Include Burden and Fee ( ) Decrease in Standard Sat. V Stage Costs

Cost Breakdown	(Dollars in Thousands) Engr	Tool	Manuf	Test	Q&RA	Matl	Misc	Total
Stage Hardware								
Structure	---	---	(126)	---	(52)	---	---	(178)
Subsystems installation	---	---	(87)	---	(36)	---	---	(128)
Electrical								
Environmental control								
Flight control								
Guidance and navigation								
Instrumentation	---	---	---	---	---	(347)	---	(347)
Ordnance subsystem								
Propulsion/pressurization system (less engine)	---	---	(249)	---	(104)	(2582)	---	(2935)
Stage GSE								
STAGE TOTAL	---	---	(462)	---	(192)	(2929)	---	(3583)
ENGINE AND ACCES. TOTAL								0
Hardware Support								
Itemized on separate list as required								
HARDWARE SUPPORT TOTAL								0
Ground Support Equipment								
Test and checkout								
Transportation and handling								
Other								
GSE TOTAL								0
Facilities								
Test								
Manufacturing								
KSC								
FACILITIES TOTAL								0

5 Year Program: 2 Sat. V/yr (518-527); 2 INT-20/yr (1-10)

Table 5.6.3-IX

ALTERNATE INT-20/S-IVB STAGE DELTA OPERATIONAL COSTS

Costs Include Burden and Fee ( ) Decrease in Standard Sat. V Stage Costs

Cost Breakdown	(Dollars in Thousands) Engr	Tool	Manuf	Test	Q&RA	Matl	Misc	Total
Stage Hardware								
Structure	---	---	(207)	---	(86)	---	---	(293)
Subsystems installation	---	---	(142)	---	(59)	---	---	(201)
Electrical								
Environmental control								
Flight control								
Guidance and navigation								
Instrumentation	---	---	---	---	---	(694)	---	(694)
Ordnance subsystem								
Propulsion/pressurization system (less engine)	---	---	(409)	---	(170)	(5164)	---	(5743)
Stage GSE								
STAGE TOTAL	---	---	(758)	---	(315)	(5858)	---	(6931)
ENGINE AND ACCES. TOTAL								0
Hardware Support								
Itemized on separate list as required								
HARDWARE SUPPORT TOTAL								0
Ground Support Equipment								
Test and checkout								
Transportation and handling								
Other								
GSE TOTAL								0
Facilities								
Test								
Manufacturing								
KSC								
FACILITIES TOTAL								0

5 Year Program: 2 Sat. V/yr (518-527); 4 INT-20/yr (1-20)

Table 5.6.3-X

ALTERNATE INT-20/S-IVB STAGE DELTA OPERATIONAL COSTS

Costs Include Burden and Fee ( ) Decrease in Standard Sat. V Stage Costs

Cost Breakdown	(Dollars in Thousands) Engr	Tool	Manuf	Test	Q&RA	Matl	Misc	Total
Stage Hardware								
Structure	---	---	(154)	---	(64)	---	---	(218)
Subsystems installation	---	---	(106)	---	(44)	---	---	(150)
Electrical								
Environmental control								
Flight control								
Guidance and navigation								
Instrumentation	---	---	---	---	---	(520)	---	(520)
Ordnance subsystem								
Propulsion/pressurization system (less engine)	---	---	(305)	---	(127)	(3873)	---	(4305)
Stage GSE								
STAGE TOTAL	---	---	(565)	---	(235)	(4393)	---	(5193)
ENGINE AND ACCES. TOTAL								0
Hardware Support								
Itemized on separate list as required								
HARDWARE SUPPORT TOTAL								0
Ground Support Equipment								
Test and checkout								
Transportation and handling								
Other								
GSE TOTAL								0
Facilities								
Test								
Manufacturing								
KSC								
FACILITIES TOTAL								0

5 Year Program: 3 Sat. V/yr (519-533); 3 INT-20/yr (1-15)

Table 5.6.3-XI

ALTERNATE INT-20/S-IVB STAGE DELTA OPERATIONAL COSTS

Costs Include Burden and Fee — ( ) Decrease in Standard Sat. V Stage Costs

Cost Breakdown	Engr	Tool	Manuf	Test	Q&RA	Matl	Misc	Total
	(Dollars in Thousands)							
Stage Hardware								
Structure	---	---	(180)	---	(75)	---	---	(255)
Subsystems installation	---	---	(124)	---	(51)	---	---	(175)
Electrical								
Environmental control								
Flight control								
Guidance and navigation								
Instrumentation	---	---	---	---	---	(347)	---	(347)
Ordnance subsystem								
Propulsion/pressurization system (less engine)	---	---	(356)	---	(148)	(2582)	---	(3086)
Stage GSE								
STAGE TOTAL	---	---	(660)	---	(274)	(2929)	---	(3863)
ENGINE AND ACCES. TOTAL								0
Hardware Support								
Itemized on separate list as required								
HARDWARE SUPPORT TOTAL								0
Ground Support Equipment								
Test and checkout								
Transportation and handling								
Other								
GSE TOTAL								0
Facilities								
Test								
Manufacturing								
KSC								
FACILITIES TOTAL								0

5 Year Program: 2 INT-20/yr (1-10)

Table 5.6.3-XII

ALTERNATE INT-20/S-IVB STAGE DELTA OPERATIONAL COSTS

Costs Include Burden and Fee — ( ) Decrease in Standard Sat. V Stage Costs

Cost Breakdown	Engr	Tool	Manuf	Test	Q&RA	Matl	Misc	Total
	(Dollars in Thousands)							
Stage Hardware								
Structure	---	---	(250)	---	(104)	---	---	(354)
Subsystems installation	---	---	(172)	---	(71)	---	---	(243)
Electrical								
Environmental control								
Flight control								
Guidance and navigation								
Instrumentation	---	---	---	---	---	(694)	---	(694)
Ordnance subsystem								
Propulsion/pressurization system (less engine)	---	---	(495)	---	(206)	(5164)	---	(5865)
Stage GSE								
STAGE TOTAL	---	---	(917)	---	(381)	(5858)	---	(7156)
ENGINE AND ACCES. TOTAL								0
Hardware Support								
Itemized on separate list as required								
HARDWARE SUPPORT TOTAL								0
Ground Support Equipment								
Test and checkout								
Transportation and handling								
Other								
GSE TOTAL								0
Facilities								
Test								
Manufacturing								
KSC								
FACILITIES TOTAL								0

5 Year Program: 4 INT-20/yr (1-20)

costs, so that even though the engineering design effort is greater, total costs are nearly the same. The greater degree of deletions, however, is reflected in the greater decrease in recurring costs ($3.5 to $7.1 million from a standard Saturn V/S-IVB) compared to the baseline case.

5.6.3.5 Launch Operations Costs

MDAC launch operations and support for the INT-20/S-IVB stage were examined in the context of the subsystem deletions, and a review of the Saturn V/S-IVB and Saturn IB/S-IVB stages launch operations and support. Based on direct and analogous evaluations, it is estimated that the MDAC launch operations and support costs would decrease by $600,000 annually for the INT-20/S-IVB stage compared to an equivalent Saturn V/S-IVB stage program. This fixed increment decrease would hold for all five of the operational programs under consideration, regardless of rate of launch mix.

5.6.4 IU Cost Plan

The technical evaluation of the Phase I part of this study confirmed the choice of Saturn V IU for conversion to INT-20 IU. Through that choice, the major impact for the baseline centered on elimination of those functions associated with the Saturn V mission S-II stage. Across the board, it was found that if the S-II networks, were simply open ended, the interface with the lower stages would be unchanged. Within the IU, minor changes to switching circuits made conversion from INT-20 to Saturn V possible. The Flight Programs differ because of uniquely different sequencing requirements, channelization, and time bases. Therefore, the essential differences can be localized to software. Within the flight control computer, the unused S-II stability filter banks are simply open ended and retained in the design.

The resulting Saturn V/INT-20 IU program permits treatment of a Saturn V or an INT-20 IU as being identical from a recurring cost point of view in the context that:

INT-20 and Saturn V IU's can be intermixed in Manufacturing.

Engineering release of the INT-20 is the same as an in-scope release of a normally modified Saturn V IU where mission-to-mission or vehicle-to-vehicle changes have been assigned an effectivity.

5.6.4.1 Groundrules and Assumptions

Cost estimates include all overhead, G and A, General Research and IRAD, and seven percent fee.

Cost estimates are in 1968 dollars without inflationary factors applied.

Cost analyses are separated into two parts: (1) Non-Recurring or Development Costs including design, development, test and evaluation activities, (2) Recurring or production costs. Recurring costs (and schedules) have been prepared assuming a rate of INT-20 production of two, four, six without the Saturn V, and two and four INT-20 with two Saturn V production.

The Saturn V INT-20 Program will not interfere with the existing Apollo delivery schedule.

The operational program will be at the rate of two, four and six deliveries per year and costs is calculated for the first five years of operation (total of 10, 20 and 30 operational vehicles).

A program definition phase (PDP) of at least six months will be required prior to stage development.

5.6.4.1 (Continued)

Costs and schedules are based on a one shift, five day week for Engineering and a two shift, five day week for Manufacturing.

Maximum use will be made of existing facilities and tooling.

Costing is based on definitions and cost data presented in the CCSD National Space Booster Study; which data has not been adjusted to 1969 rates.

A learning curve factor is not applied.

Delta costs for the operational program costs do not consider or include costs of launch support effort at KSC.

INT-20 IU production costs do not differ from the current Saturn V IU program.

INT-20 and Saturn V IU mixes are equivalent to Saturn V IU.

5.6.4.2 Elements of Cost

Definitions of the elements of cost, appearing in the cost format of Table 5.6.4.2-I are provided below:

Engineering - Includes design, analysis and data generation including test support liaison, engineering technicians and subcontract services for any of these functions. Other engineering includes system integration and mission support engineering including the development, writing and implementation of computer programs and subcontract services for any of these functions.

Manufacturing - Includes effort for such functions as fabrication, disassembly and assembly operations, the generation of manufacturing routings, transportation and handling, shipping and receiving.

Tooling - Includes effort to design, fabricate and maintain tooling required for fabrication and assembly of a product.

Quality and Reliability - Includes effort required to assure that products meet or exceed all specifications or design requirements and generally includes, but is not limited to, the preparation of quality control operating procedures and sampling plans, part and material analysis, the determination and reporting of the qualification status of parts and materials, vendor surveillance and reliability analysis for failure modes, failure effects and criticality.

(DOLLARS IN THOUSANDS)

COST BREAKDOWN	Engineering	Mfg. and Tooling	Logistics	Quality & Reliab.	Material	Prog. Supt. & Mgmt.	Test	Misc.	Total
INSTRUMENT UNIT									
Structures									
Environmental Cont.									
Electrical									
Control	5.0				4.5	2.4		0.6	12.5
Guidance									
Measuring									
Telemetry									
Radio Frequency									
Other									
Instr. Unit Total									
GROUND SUPPORT EQUIP.									
Test and Checkout	304.238				367.547			25.252	697.037
Transp. and Hndlg.									
Other (6/yr rate only)									
GSE Total									
FACILITIES									
Test									
Manufacturing	0				0	0		0	0
Other									
Facilities Total									
IU SYSTEM TOTAL	309.238				372.047	2.4		25.852	709.537

5.6.4.2 (Continued)

Material - Includes the costs for manufacture or production of raw material and subcontract purchases (excluding engineering services).

Program Support and Management - Includes effort required to adequately manage and support a program including such functions as contract administration and reporting, program and production planning, analysis and change control, the generation and maintenance of engineering documentation (drawings, specifications, standards, procedures, records and release), configuration and data management.

Test - Includes effort required to perform all test operations including support of test operations, the preparation of test plans, reduction of test data and test report preparation.

IU GSE Engineering - Includes engineering in support of all ground support equipment for factory operations including the systems checkout test complexes.

Miscellaneous - Included in miscellaneous are computer time and travel.

5.6.4.3 Presentation of Cost Data

Table 5.6.4-I, IU Development Cost Summary presents the breakdown of nonrecurring developmental costs summarized as follows:

Item	Reference Schedule	Cost in Thousands
FCC Design Modification	Figure 5.1.4.2-2	$ 12.5
System GSE Development (For INT-20 and Saturn V production rate of six per year.)	Figure 5.1.4.2-1	$697.0

It should be reemphasized that these flight equipment costs are essentially negligible because, as was shown in Figure 5.5.4-1, the Instrumentation, Program and Component List, Mission Definition Document, and the Final Filter Design Document incur a contract recurring cost which in effect covers the development cost of each mission impacted Saturn V IU where unique requirements are in general imposed.

Recurring operational costs are negligible and may be considered as zero delta from the Saturn V costs. A possible minor saving is possible in the Flight Control Computer if the S-II filter circuits were removed. It would be

5.6.4.3 (Continued)

necessary to replace the electronic card assemblies with dummy weighted cards to preserve the vibratory characteristics of the motherboards which would incur redesign. The advantage of interchangeable with the Saturn V FCC is considered to outweigh the 20 percent reduction in FCC unit cost potentially possible in the redesign.

Facility modification costs are straight forward and basically independent of the INT-20 requirement; having to do with sensitivity to rate of production.

Five year summary costs for IU production based on manufacturing rates are reported elsewhere. In summary, the five year costs, considered as deltas to the Saturn V program costs reported are negligible except for a rate of six per year in which case the total five year cost delta is simply the conversion cost of the facility GSE.

SECTION 6
THE INT-20 RETROFIT PLAN

6.0 GENERAL

An INT-20 vehicle or vehicles can be assembled from Saturn V stages retrieved from storage.

A retrofit kit for the S-IC stage would be designed and manufactured to cap lines and cover openings remaining after removal of the center F-1 engine. The five-inch high adapter ring to mate the S-IC and S-IVB stages would be designed and manufactured. The adapter ring would conform to the S-IC bolt hole pattern on the bottom and the S-IVB bolt hole pattern on the top. An electrical connector cable will be designed and manufactured to connect the S-IC cables to the S-IVB cables which are located nearly 90^{o} apart.

Testing for the retrofit INT-20 is the same as for an in-line INT-20. The F-1 engine long duration firing qualification is needed. The recommendation for the first flight to be unmanned, but with a useful payload remains the same.

The retrofit schedule (Figure 6.0-1) shows the time to retrieve Saturn V stages and an I. U. from storage and modify for use as an INT-20 vehicle. As in the schedule for the in-line INT-20, the S-IC stage is the pacing item with a time from ATP (Authority to Proceed) to first item delivery to KSC of 18 months. The retrofit schedule is half as long as the in-line INT-20 schedule, which is 36 months.

The cost to obtain the first retrofit INT-20 is $4 million. This cost is primarily for development functions. If a second retrofit INT-20 were needed there would be no development cost. The recurring cost of the second INT-20 would be about $200,000. The S-IVB stage exhibits an overall savings because retrorockets would not be purchased and this exceeds the cost for hardware work by $23,700. The cost saving for not installing the center F-1 engine and engine related hardware in the S-IC stage would be about $3 million. KSC launch facility modification would add $3.2 million to the first INT-20 vehicle. Conversion of the Mobile Service Structure would add $200,000 to the second retrofit INT-20. Retrofit costs are shown on Table 6.0-I.

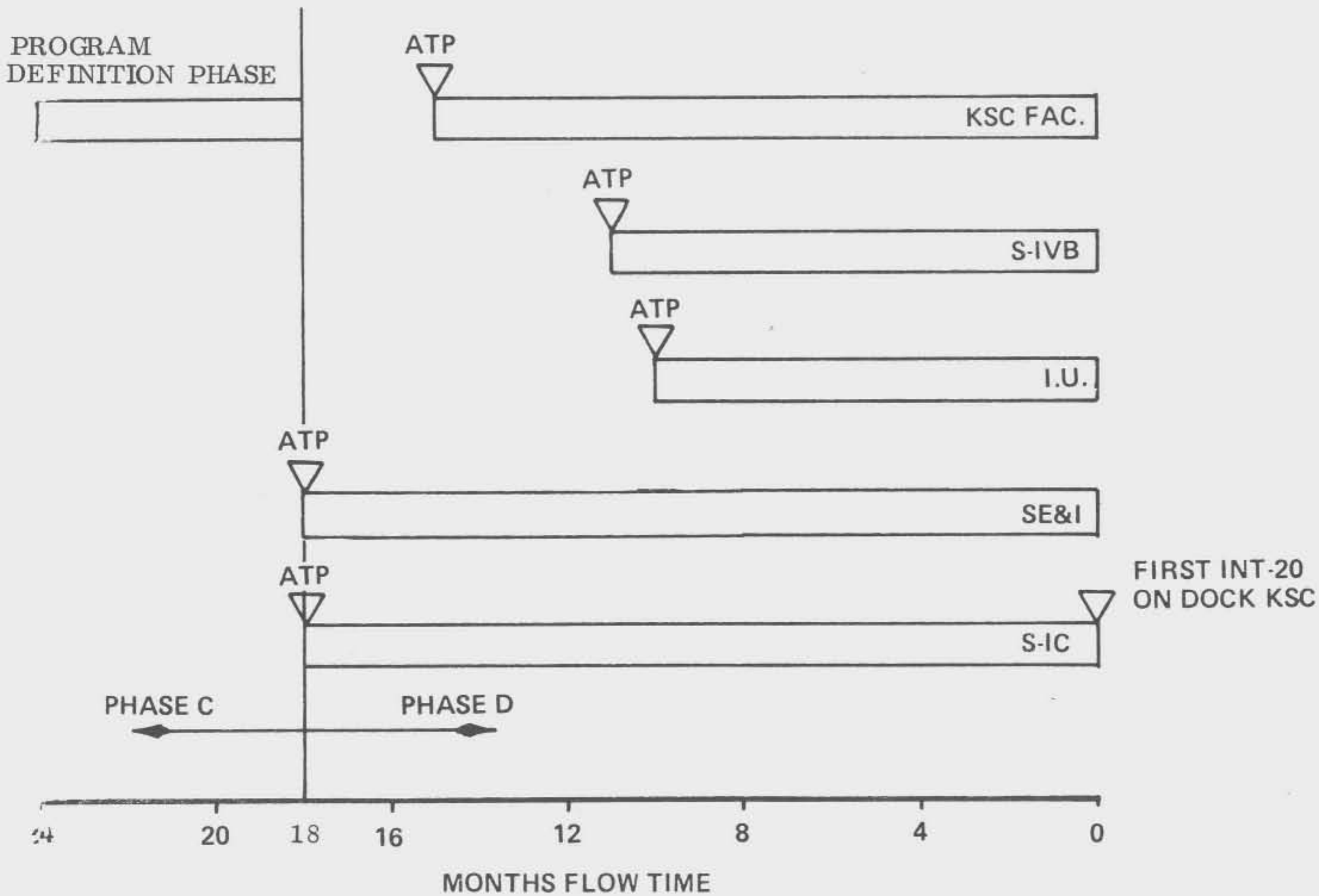


FIGURE 6.0-1 INT-20 RETROFIT SCHEDULE

TABLE 6.0-I

RETROFIT INT-20 COSTS

	1st Vehicle	2nd Vehicle	
S-IC	$1,111,000*	$200,000	
S-IVB (including attaching ring)	$2,927,500	$-23,700	
I. U.	$ 12,000	$ 12,000	
F-1 Engine	$ 225,000	None	
J-2 Engine	None	None	
SE&I	$ 110,000	None	
Total	$4,385,500	$188,300	
KSC Facilities	$3,200,000	$200,000	(Conversion from Saturn V to INT-20 and back to Saturn V)
TOTAL	$7,585,500	$388,300	

* F-1 engine and other deleted S-IC hardware cost saving ($3M) is not included.

** Cost Saving by not purchasing retrorockets.

6.1 S-IC STAGE

The retrofit S-IC stage configuration for the INT-20 vehicle is the same as the baseline 4 engine S-IC stage (Section 4.2.2.1) except as defined in this section. This definition is based on the premise that the retrofit configuration will be one of the first two INT-20 flight stages.

6.1.1 STRUCTURES SUBSYSTEMS

6.1.1.1 Oxidizer Tank

The flat plate cover with a floating flange used for the baseline S-IC will also be used for the retrofit S-IC (refer to Section 4.2.2.1.a.2). The LOX standpipe, which is deleted for the baseline S-IC, will be retained for retrofit. The support ring added for the baseline S-IC will not be required.

a. Oxidizer tank backup data

Retention of the standpipe for the retrofit configuration is based on ease of retrofit considerations and the possible consequence of tank damage during its removal. Although the LOX standpipe could be removed from an assembled stage, it would require disassembly of the standpipe and removal and reinstallation of the GOX distributor. An assessment of the loads induced on the bulkhead due to the LOX trapped in the standpipe when the tank level is below the standpipe openings indicated no problem. A similar situation was experienced on S-IC-4 during flight with the programmed early shutdown of the center engine.

6.1.1.2 Fuel Tank

The eight lower fuel bulkhead base gore segments which are increased in thickness for the baseline S-IC (Section 4.2.2.1.a.4.c) will not be revised for the retrofit S-IC. The required 1.4 factor of safety can be maintained for hoop compression on the retrofit S-IC by revising the trajectory so that the acceleration is reduced to an acceptable level. This is accomplished for the second iteration trajectory (FIGURE A-23 of APPENDIX A) by cutting off the first two engines at 126 seconds, thus reducing the critical acceleration from 4.68g to 3.68g (nominal). A 1.25 factor of safety can be maintained by cutting off the first two engines at 133 seconds thus limiting the acceleration to 4.05 g (nominal).

a. Fuel tank backup data

An alternate method of maintaining the 1.4 hoop compression factor of safety for retrofit was studied. It consists of adding three support rings to the inner lower fuel bulkhead surface to provide the increased hoop compression capability required for the baseline INT-20 trajectory (FIGURE 6.1-1 & -2). A preliminary assessment of this method established feasibility from both a structural and

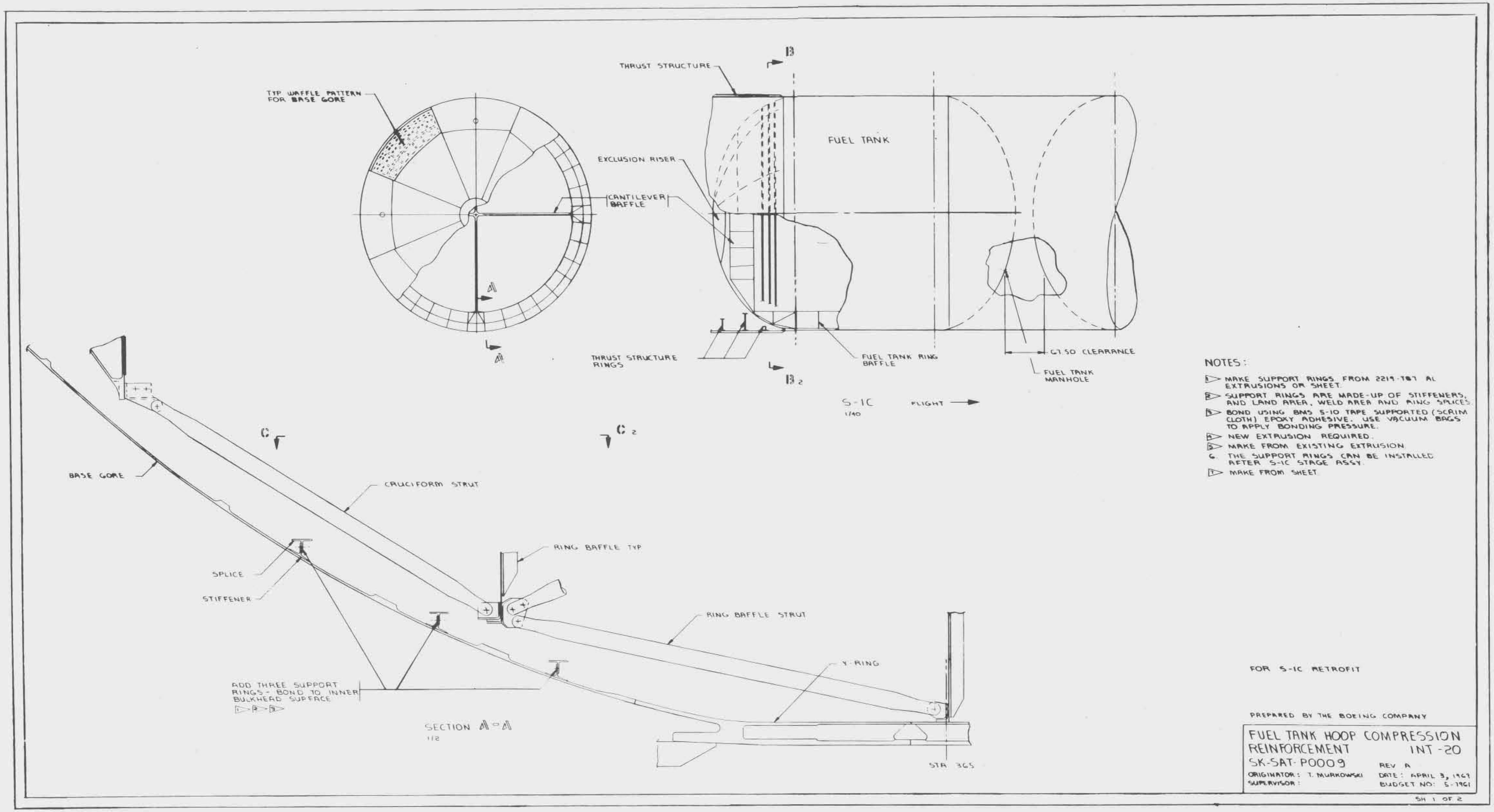

TYP WAFFLE PATTERN
FOR BASE GORE
THRUST STRUCTURE
B
FUEL TANK
EXCLUSION RISER
CANTILEVER
BAFFLE
A
A
THRUST STRUCTURE
RINGS
FUEL TANK RING
BAFFLE
B
67.50 CLEARANCE
FUEL TANK
MANHOLE
S-IC
1/40
FLIGHT
NOTES:
1 MAKE SUPPORT RINGS FROM 2219-T87 AL
EXTRUSIONS OR SHEET.
2 SUPPORT RINGS ARE MADE-UP OF STIFFENERS,
AND LAND AREA, WELD AREA AND RING SPLICES.
3 BOND USING BMS 5-10 TAPE SUPPORTED (SCRIM
CLOTH) EPOXY ADHESIVE. USE VACUUM BAGS
TO APPLY BONDING PRESSURE.
4 NEW EXTRUSION REQUIRED.
5 MAKE FROM EXISTING EXTRUSION.
6. THE SUPPORT RINGS CAN BE INSTALLED
AFTER S-IC STAGE ASSY.
7 MAKE FROM SHEET.
C
C
BASE GORE
CRUCIFORM STRUT
RING BAFFLE TYP
SPLICE
STIFFENER
RING BAFFLE STRUT
Y-RING
ADD THREE SUPPORT
RINGS - BOND TO INNER
BULKHEAD SURFACE
1 2 3
SECTION A-A
1/2
STA 365
FOR S-IC RETROFIT
PREPARED BY THE BOEING COMPANY
FUEL TANK HOOP COMPRESSION
REINFORCEMENT INT-20
SK-SAT-P0009
REV A
ORIGINATOR: T. MURKOWSKI
SUPERVISOR:
DATE: APRIL 3, 1967
BUDGET NO: 5-1961
SH 1 OF 2

FIGURE 6.1-1

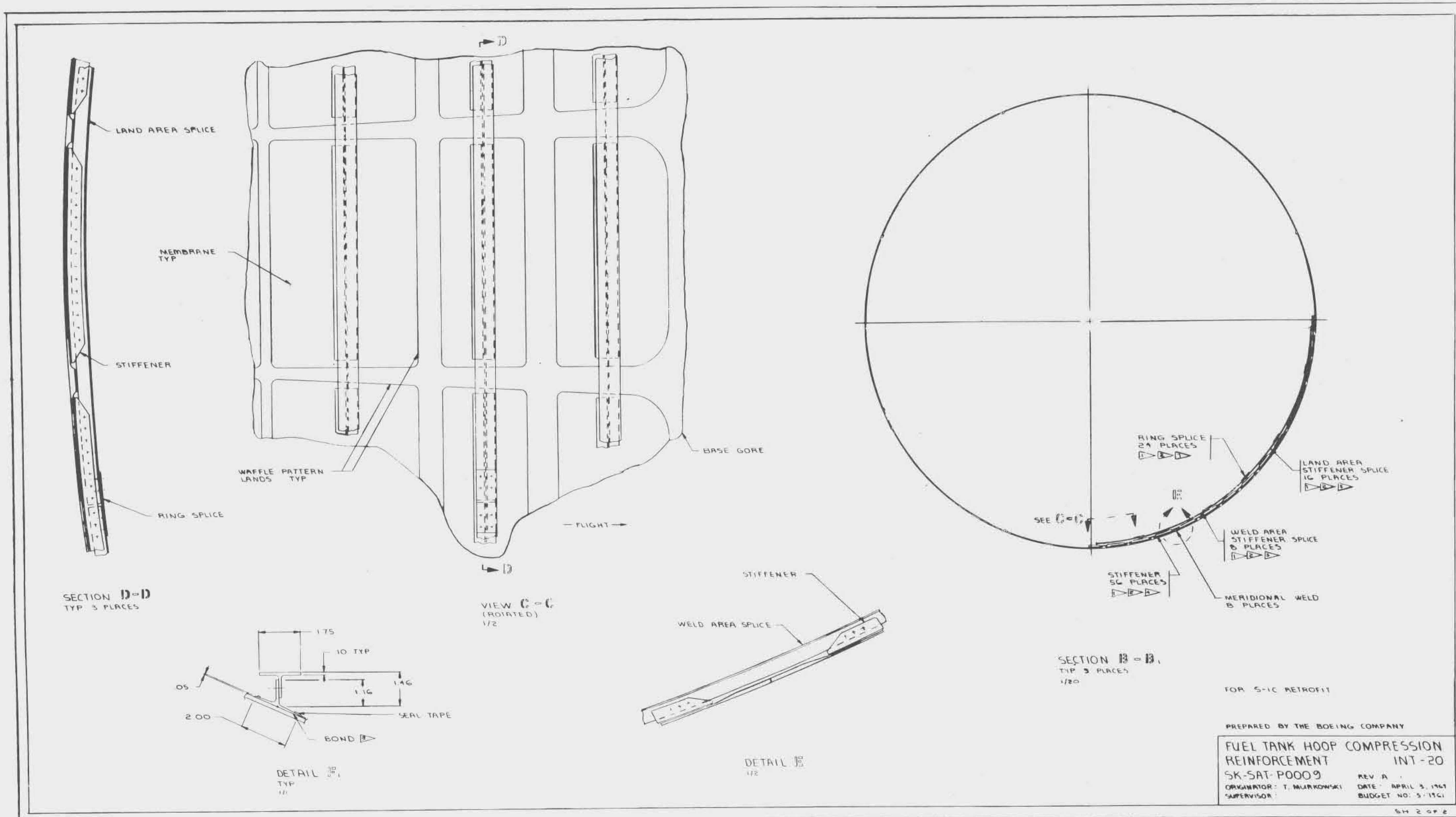

LAND AREA SPLICE
STIFFENER
RING SPLICE
SECTION D-D
TYP 3 PLACES
MEMBRANE
TYP
WAFFLE PATTERN
LANDS TYP
BASE GORE
FLIGHT
VIEW C-C
(ROTATED)
1/2
RING SPLICE
24 PLACES
LAND AREA
STIFFENER SPLICE
16 PLACES
SEE C-C
WELD AREA
STIFFENER SPLICE
8 PLACES
STIFFENER
56 PLACES
MERIDIONAL WELD
8 PLACES
SECTION B-B
TYP 3 PLACES
1/20
1.75
.10 TYP
1.46
1.16
.05
2.00
SEAL TAPE
BOND
DETAIL F
TYP
1/1
STIFFENER
WELD AREA SPLICE
DETAIL E
1/2
FOR S-IC RETROFIT
PREPARED BY THE BOEING COMPANY
FUEL TANK HOOP COMPRESSION
REINFORCEMENT INT-20
SK-SAT-P0009
REV A
ORIGINATOR: T. MURKOWSKI
SUPERVISOR:
SH 2 OF 2

FIGURE 6.1-2

TABLE 6.1-I

SYSTEMS RETROFIT

SYSTEM	ITEMS DELETED	WEIGHT POUNDS	ITEMS ADDED	WEIGHT POUNDS	REMARKS
L OX Interconnect (60B41014)	1	28	1	7	Spool - Figure 4-16 2 Connections
LOX Bubbling (60B41221)	6	1	2	Nil	1 Cap & 1 Plug
LOX Pressurization (60B51400)	3	36	3	10	Flange
Fuel Pressurization (60B49600)	8	24	7	10	Flanges
Control Pressure (60B52500)	19	6	2	Nil	Caps
Turbopump Oxidizer Seal (60B37601)	5	5	1	Nil	Plug
Radiation Calorimeter Purge	0	0	3	1	Line, Union, Elbow
LOX Dome & Gas Gen. Purge (60B37600)	7	11	1	Nil	Plug
Engine Cocoon Thermal Purge (60B37602)	7	8	1	Nil	Plug
Thrust O.K. Checkout (60B37600)	11	1	1	Nil	Cap
Thrust Chamber Prefill (60B37500)	5	5	1	Nil	Plug
POGO Supression (60B41340)	12	5	1	Nil	Plug
Fluid Power (60B82000)	43	51.6	1	2	Flange
TOTAL CONNECTIONS	127		25		

6.1.1.2 (Continued)

a. manufacturing standpoint. However, extensive development of the bond application and quality control techniques would be required. The installation of the support rings on a completed stage is possible but undesirable. This method, therefore, is recommended only if revision to the trajectory and the attendant loss in payload is unacceptable.

6.1.2 Propulsion and Mechanical Subsystems

6.1.2.1 Engine Support Purge Systems

The retrofit configuration will utilize the radiation calorimeter purge systems defined in the baseline configuration (Section 4.2.2.1.b.3.c.2).

6.1.2.2 Fuel Loading Probe

The fuel loading probe, which is lengthened for the baseline configuration, will not be changed for retrofit. The revised retrofit trajectory reduces the acceleration during the period of critical fuel tank pressure and hence affects the total fuel maximum bottom pressure. In addition, propellant ballast will be added for retrofit to compensate for the reduced payload capability. The increased fuel level and additional liquid head capability due to reduced acceleration for retrofit assure that the fuel load level can be high enough to preclude the requirement to increase the fuel loading probe length while meeting tank pressure requirements.

6.1.3 Electrical/Electronic Subsystems

The INT-20 retrofit configuration will be the same as the baseline configuration. The 292 measurements defined for the first two flight stages apply to the retrofit configuration.

6.1.4 S-1C Retrofit Manufacturing Plan

6.1.4.1 Background

A retrofit INT-20 is a vehicle made to the INT-20 retrofit configuration definition Section 6.1.1 through 6.1.3 from an existing S-1C stage. It is essentially the same as the baseline INT-20 except that for economics and manufacturing convenience the stage interface hole pattern, the thickness of the lower fuel tank base gores and the center engine LOX standpipe remain as presently installed on S-1C-11.

The assumption that the lower fuel base gores need not be altered is based on the assumption that it will be acceptable to modify the retrofit INT-20 flight trajectory as defined in the retrofit description to avoid reduction of the hoop comprehension safety factor. In the event that this is undesirable, the alternate method of reinforcing the lower fuel base gores as described in the retrofit description is discussed.

6.1.4.2 Forward Skirt (60B14009)

The forward skirt of the first stage INT-20 retrofit vehicle remains the same as the S-1C. An adapter ring is added to the retrofit S-IV-B by McDonnell-Douglas to pick up existing S-1C stage interface holes.

6.1.4.3 Oxidizer Tank (60B03101)

The only change from the five engine S-1C to the INT-20 retrofit configuration oxidizer tank is the addition of the center engine suction fitting closure plate and floating flange shown in method 2 of FIGURE 4.2.2.1-3 of the baseline INT-20 Engineering Documentation. This closure is discussed in the LOX Duct removal sequence below.

6.1.4.4 Intertank (60B29800)

There are no changes to the Intertank.

6.1.4.5 Fuel Tank (60B25001)

The two inboard fuel suction cover plates and the inboard LOX tunnel closure cover shown in FIGURE 4.2.2.1-4 of the Baseline Design Description will be fabricated and installed. No changes are made to the lower fuel base gores. No new tooling is required.

a. Fuel tank backup data

Paragraph 6.1.1.2.a. and FIGURES 6.1-1 & -2 of the Engineering Retrofit Definition deal with a method of adding to the hoop compression capability of the lower fuel base gores. This method is only a backup and is not the one proposed by Engineering nor is it included in the Operations cost or schedule plans.

6.1.4.5 (Continued)

a. It is considered appropriate however, to include a discussion of the Manufacturing sequence and problems which might be encountered if the alternate proposal was undertaken.

b. Two extrusion dies would be ordered to extrude the straight lengths of tee stiffeners assuming that the 90 degree tees would be made from standard dies. Three stretch press block and jaw sets would be fabricated to stretch form the extrusions. The extrusions would be trimmed to drawing sizes for installation into the tank. A development program of undetermined magnitude would be conducted to obtain proven structural bond capability on hand stripped and cleaned base gores with tools designed to clamp the short segments to the gore membranes using vacuum and incorporating heater coils with variable temperatures to about 200°F. Gores with segments bonded in this manner would be required to pass tensile and pressurization tests. A method of inspecting the bonds in the fuel tank would have to be selected or developed.

   If the vehicle selected were one which had been static fired, the fuel vapor and residue in the tank could make cleaning for bonding a big problem. Experience gained on previous tank structural bonds, notably the LOX tunnel stiffener rings indicate that problems are encountered unless the level of cleanliness is very closely controlled. In the opinion of Manufacturing Engineering these problems can be overcome and the tank successfully reinforced but a Manufacturing development program with Engineering participation would be required.

6.1.4.6 Thrust Structure (60B18054)

The center engine adapter fitting, support strut and attach hardware will be unbolted and removed from the thrust structure as illustrated in FIGURE 4.2.2.1-7 of the Baseline Design Description. Eight inboard fuel suction duct support links will be unbolted and removed as shown in FIGURE 4.2.2.1-8.

6.1.4.7 Heat Shield (60B20800)

The retrofit vehicle heat shield is the same as the baseline INT-20 heat shield installation. The work to be performed is the same plus the removal of all the existing flight panels and the support beams identified as deletions in Section 3.1.6 of Appendix A, installation of the new support beams in the center area and installation of the static firing panels to INT-20 configuration on the vehicle. After static firing all the static firing panels will be replaced with INT-20 configuration flight panels.

6.1.4.8 F-1 Engine Removal

All connections to the center F-1 engine will be broken and the engine removed. This is an established procedure used previously on the S-1C.

6.1.4.9 Fuel and LOX Prevalve and PVC Duct Removal

These components have been successfully removed for rework on the S-1C. They will be removed from INT-20 Retrofit vehicles by mechanical disconnection and lifting out with a mobile crane and existing slings. See Figures 4.2.2.1-14 & 24 of the baseline Design Description. The LOX interconnect Spool 60B41021-1 will then be temporarily removed to allow clearance for the LOX Suction Duct to be removed as described below and the inboard fuel suction fittings will be capped per the fuel tank manufacturing plan.

6.1.4.10 Removal of Inboard LOX Suction Duct Background

a. Background

Tool HT2-B370-12000 (existing) is a 63 foot long trough with supporting structure and a winch. This tool has been successfully used to change out helium bottles in S-1C LOX Tanks and to remove and outboard LOX Suction Duct from an S-1C.

b. Description of sequence

Fabricate three (3) welded steel support structures as additions to tool HT2-B370-12000. These additions will be approximately 4000 lbs. of rough steel structure adapting existing HT2-B370-12000 to transportation trailer TNTR-B370-18050. See FIGURE A-68 Structure added is denoted by 2.

Fabricate a hositing cover plate similar to the plate in FIGURE 4.2.2.1-3 of the Baseline Design Description except make from .75 thick 6061-T6 Aluminum and provide a hole for lifting eye in the center.

Unbolt the center engine propellant lines support assembly. Support the assembly with cables and swing it outboard for duct removal clearance.

Mount HT2-B370-12000 on TNTR-B370-18050 and roll into position as shown on FIGURE A-68. Level tool so that trough centerline coincides with the inboard LOX suction duct centerline.

Personnel platform PP-B370-8014 may be in position as shown in FIGURE A-68 but is not essential to this operation.

Attach the hoisting cover plate to the aft end of the inboard LOX duct and attach the cable from the winch at the far end of HT2-B370-12000 to the lifting eye of the center of the plate.

6.1.4.10 (Continued)

b. Disconnect the inboard LOX suction duct from the inboard LOX suction fitting in the intertank. Take up on winch on HT2-B370-12000 and roll the inboard LOX suction duct out of vehicle and into the trough of the tool. Cap off the inboard LOX suction fitting and seal per FIGURE **4.2.2.1-3 of the Baseline Design Description.**

Roll TNTR-B370-18050 with HT2-B370-12000 and the suction duct aboard away from vehicle to area unde the overhead cranes. Remove the inboard suction duct from the tool with the cranes and HT-B470-8016 suction duct hoisting tool. Place duct in its storage container.

c. Reinstall the propellant lines support assembly.

6.1.4.11 Center Engine Systems Deletions

a. Deletion and plugging of the various systems to and from the center engine are outlined in the First Stage INT-20 Manufacturing plan. The only significant difference with respect to these systems on the Retrofit plan is that they are already installed and must be removed whereas installation was simply omitted on the baseline manufacturing plan. These systems consist of plumbing items, tubes, flanges, plugs, etc. To present the magnitude of the hardware removal task without introducing repetition, Table 6-1 below lists the quantities and weights of items deleted and added, excluding the larger ducts, pre-valves and PVC Ducts discussed in more detail above. The number of items deleted is equal to the number of plumbing items which must be disconnected. The number of items added is equal to the number of plumbing items which must be made. Seals are excluded from totals since they are within a connection. Weights are to nearest pound and the delta cost of the items added for retrofit which includes the radiation calorimeter purge system is equivalent to that calculated for these systems for the first Manufactured INT-20.

6.1.4.12 Electrical/Electronic Equipment

The twenty-nine (29) disconnected connectors will be protected with dust caps, the "hot" wires will be identified and the excess cable assemblies will be coiled and stowed. Eleven (11) existing cable assemblies requiring minor modifications will be removed from the vehicle and reworked in the cable fabrication area utilizing existing facilities. Thirteen (13) new cables will be fabricated and routed with existing wire bundles on the vehicle using available clamping devices. Retrofit kits will be made available when necessary.

Six (6) electrical distributors requiring rework will be removed from the vehicle and modified in the electrical fabrication area by adding and/or deleting wires.

A rework and installation sequence will be developed.

6.1.4.13 Stage Instrumentation

Approximately seven instruments, servo-accelerometers, resistance thermometers and pressure transducers and the respective amplifiers will be deleted as the center engine is removed. Approximately eleven instruments, temperature and vibration and twenty-one (21) amplifiers will be added to the present S-1C configuration.

Additional instrumentation as defined by a Specification Control Drawing will be procured from approved commercial sources. Minor assemblies and testing will be accomplished utilizing existing facilities.

Retrofit kits will be prepared and made available for installation while the vehicle is in the final assembly position.

The measuring racks will be modified on the vehicle by adding and/or removing the required amplifiers. The Heat Shield panels will be reworked on the bench in the electrical fabrication area to add the thermocouple assemblies. The installation of the resistance thermometer and servo-accelermeters will be accomplished on the vehicle in the final assembly position. A rework and installation sequence will be developed.

6.1.5 S-IC Retrofit Implementation Plan

6.1.5.1 Configuration Management

The configuration management approach for incorporation of the changes defined in Paragraph 6.1.1 through 6.1.3 would be the same as defined for the baseline INT-20 production incorporation (Paragraph 5.1.2) except for the following:

a. The change would be authorized and implemented under a change of the contract applicable to the S-IC-14 stage.

b. The configuration change would be documented as shown in Figure 6.1.5-1 in compliance with S-IC retrofit kit procedures.

c. The change would be planned and incorporated in accordance with S-IC retrofit kit processing requirements.

Figure 6.1.5-2 shows the definition phase milestones for converting the S-IC-14 to an INT-20 configuration.

6.1.6 S-IC Retrofit Schedule

Two schedules have been generated for incorporation of the INT-20 retrofit kit on S-IC-14 subsequent to storage of the stage. Figure 6.1.5-2 shows the minimum flow period from contract go-ahead to delivery in September 1972, based on the following.

a. The kit incorporation will be made after a storage period.

b. The stage is static fired before storage.

c. Static firing of the stage after modification to INT-20 is not required.

Figure 6.1.5-3 shows flow bars and milestones for an INT-20 retrofit program, in relation to the existing scheduled events for S-IC-14, to consider the possibility of static firing of the stage after retrofit incorporation. This schedule shows that the existing scheduled static firing commitment occurs at the same time as a go-ahead for an INT-20 delivery in September 1972. However, the scheduled events for S-IC-15* would allow time for negotiating change commitments for static firing the Sat V S-IC stage before storage to requiring the first INT-20 (retrofitted) stage to be static fired after retrofit.

* Note that the present contract requires that all present stages be static fired before storage.

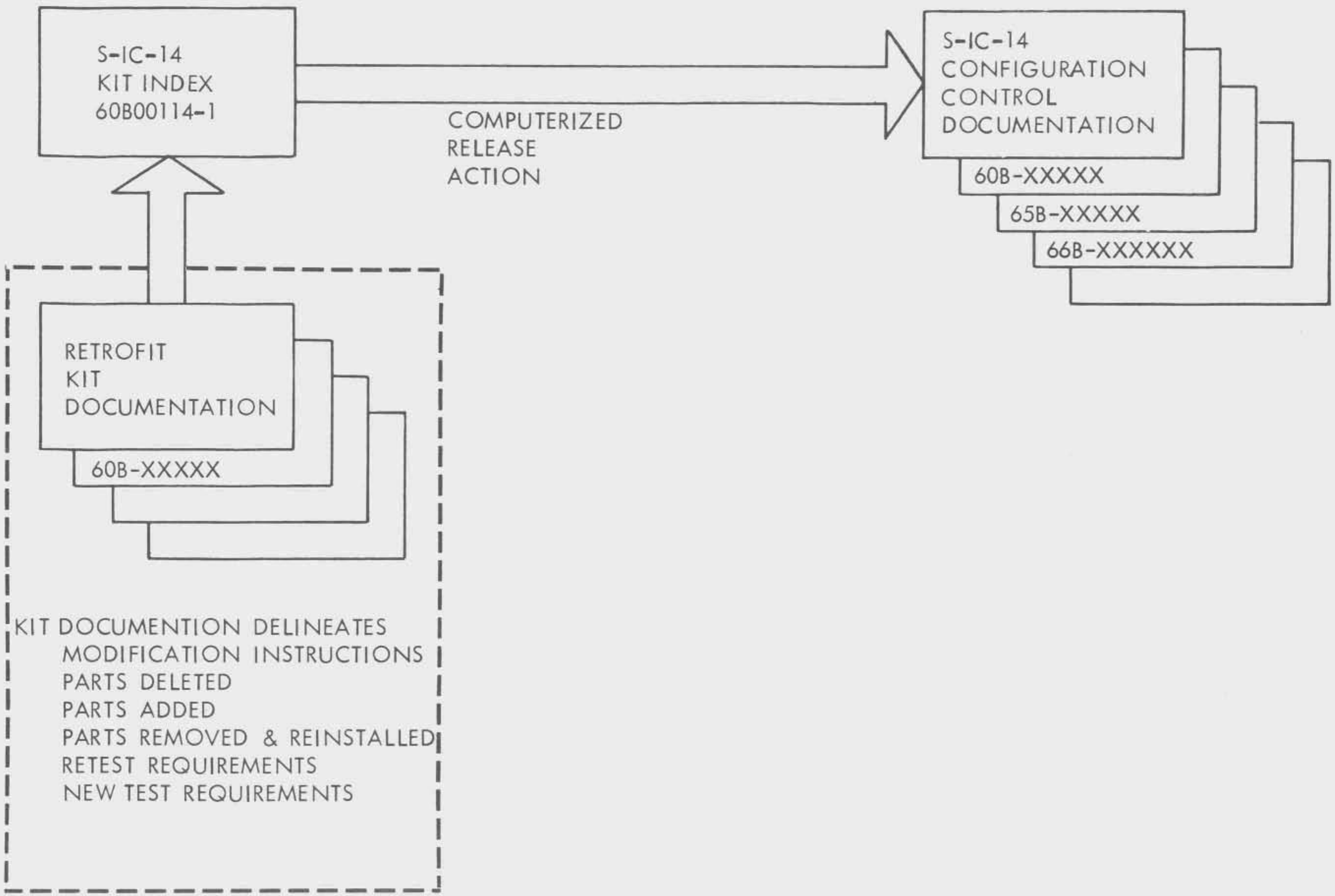


FIGURE 6.1.5-1 INT-20/S-IC RETROFIT DOCUMENTATION PLAN (FOR CONVERTING EXISTING STORED STAGE)

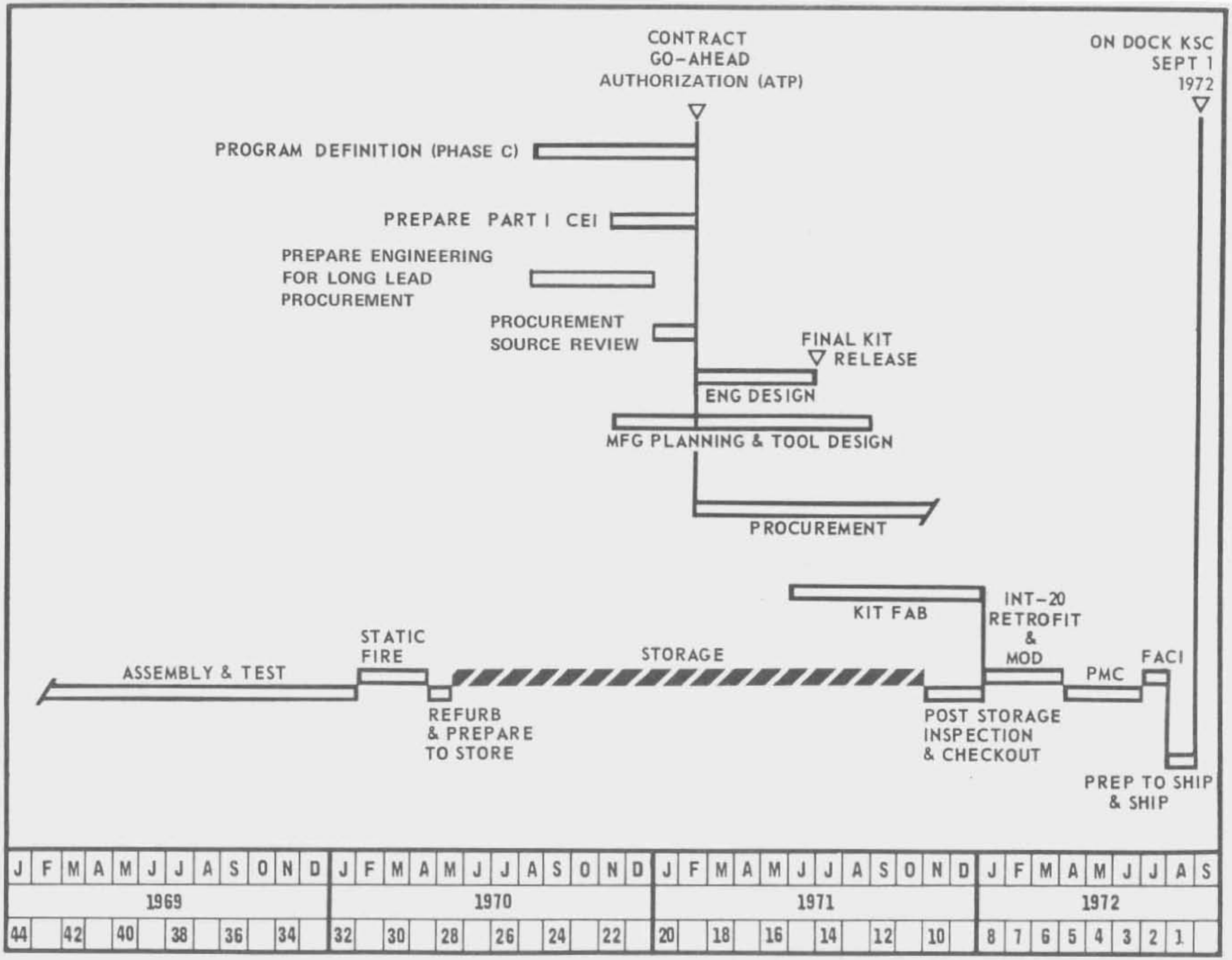


FIGURE 6.1.5-2 RETROFIT OF S-IC-14 TO INT-20

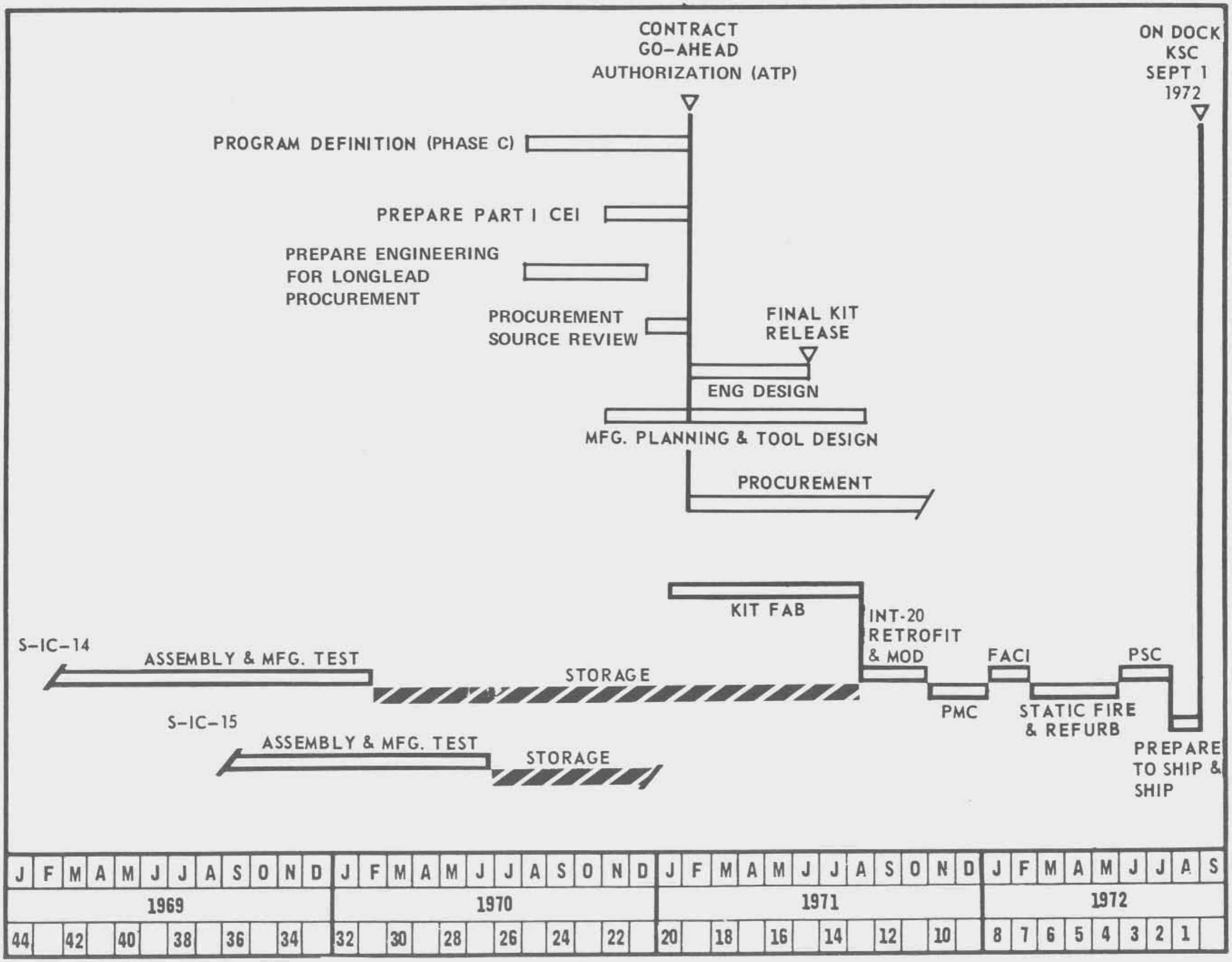


FIGURE 6.1.5-3 . RETROFIT OF S-IC-14 TO INT-20

6.1.7 S-IC Retrofit Delta Price Estimates

These planning estimates are based upon sections 6.1.5.1 and 6.1.6 of this document and the following pricing ground rules:

a. 1968 dollars and rates were used.

b. No disruption costs to the contract for Stages S-IC-11 through S-IC-15 were considered.

6.1.7.1 Delta Price Estimate to Convert a S-IC/Sat V to S-IC/INT-20 after Storage

This planning estimate, see Table 6.1.7-I , is based upon the retrofit of S-IC-14 after this stage had been static fired, refurbished and placed into storage. The post manufacturing, FACI and preparation for shipping costs after the conversion to an INT-20 configuration are offset by the post static firing and preparation for shipping costs which are included in the S-IC-14 costs.

However, if the S-IC/INT-20 stage requires static firing, the estimated price would increase, see Figure 6-6.

6.1.7.2 Delta Price Estimate to Convert a S-IC/Sat V to S-IC/INT-20 After Manufacturing Complete

This planning estimate, see Table 6.1.7-I is based upon the retrofit of S-IC-14 after this stage had been completed in the factory. The costs to accomplish PMC, FACI, static firing, refurbishment, PSC and preparation for shipping of the S-IC/INT-20 are approximately the same costs as for the S-IC/Sat V. Therefore, the estimated delta cost for the S-IC/INT-20 would be for the retrofit only.

TABLE 6.1.7-I S-IC RETROFIT DELTA PRICE ESTIMATE SUMMARY

S-IC-14 DELTA PRICE
(000 OMITTED)

		DOLLARS
ENGINEERING		$ 359
OPERATIONS		169
Q & RA		113
SYSTEMS TEST		326
OTHER		37
	TOTAL LABOR	$ 1,004
MATERIALS		107
	TOTAL PRICE	$ 1,111

NOTE: The estimated planning estimate to static fire the S-IC-14 the second time (per paragraph 6.1.7.1) will be $3.3 million.

6.2 S-IVB STAGE RETROFIT PLAN

This plan considers the retrofit of one Saturn V/S-IVB stage and interstage into the INT-20/S-IVB baseline configuration. This retrofit operation will not significantly impact the design or test plans. The main impact is in the areas of manufacturing and schedules.

6.2.1 Manufacturing Plan

The retrofit INT-20/S-IVB stage manufacturing flow plan depicted in Figure 6.2-1 reflects the manufacturing operations and positions where the deletions, "cap-off" and "stowing" functions will be performed. New planning paper will be issued to institute these operations.

Most modifications will be performed on the stage while erected in one of the available tower complexes at the MDAC Huntington Beach S-IVB Manufacturing Facility. It will not be necessary to remove any of the major subassemblies to effect the deletions. The stage will then be relocated to a horizontal position for final deletions and checkout preparations. Deletion of the ullage engines from the APS modules is a separate operation.

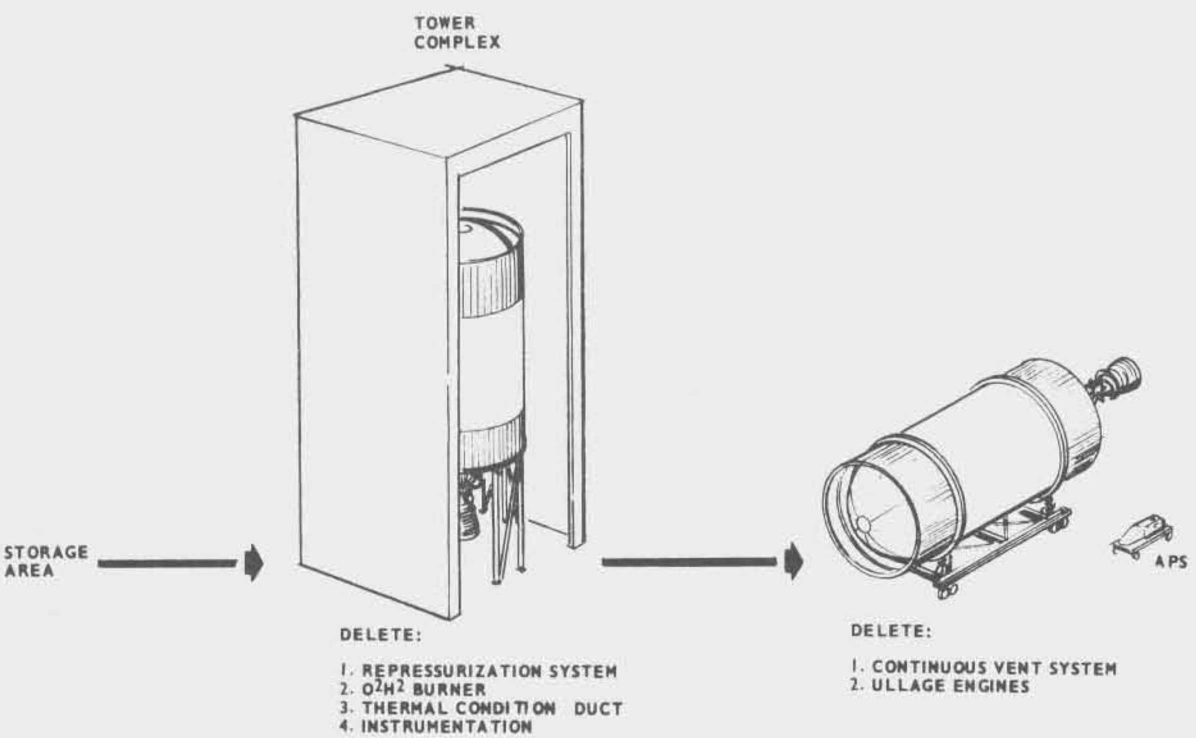


Figure 6.2-1. Manufacturing Plan, Retrofit INT-20/S-IVB Stage

The stage will then be placed in the checkout tower for post-manufacturing acceptance checkout of the affected systems. Following that, it will be prepared for shipment directly to KSC (no static firing repeat).

For the INT-20/S-IVB interstage, proper interface with the S-IC stage will be accomplished by use of the new adapter ring provided by MDAC. The manufacture of the adapter ring was discussed previously in Section 5.3.3. Figure 6.2-2 illustrates the installation of the assembled ring to an existing interstage prior to shipment to KSC. Note that in this case, the retrorocket provisions cannot be deleted, and existing wiring is coiled and stowed. The retrorockets and their attendant ordnance will not be subsequently installed.

Tooling requirements are the same as those discussed under the in-line Manufacturing Plan, Section 5.3.3, with possible exception of some new minor work stands to facilitate access for removal of components.

6.2.2 Schedule Plan

The schedule for retrofit (manufacturing) is presented in Figure 6.2-3 in terms of months from ATP (initial drawing release will be at three months). Tooling, procurement, and new fabrication will be paced by the new attach ring. Retrofit is paced by planning following drawing release.

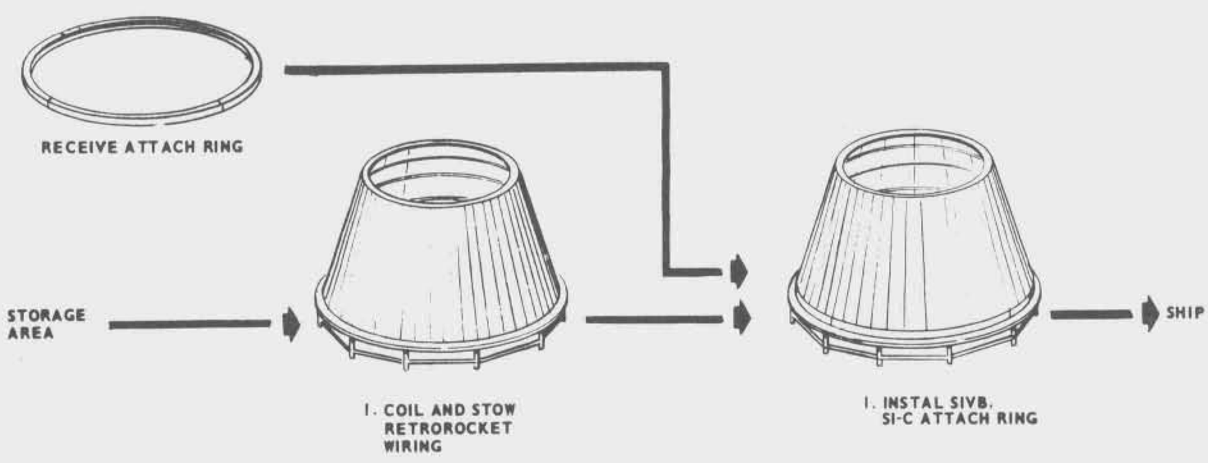


Figure 6.2-2. Manufacturing Plan, Retrofit INT-20/S-IVB Interstage

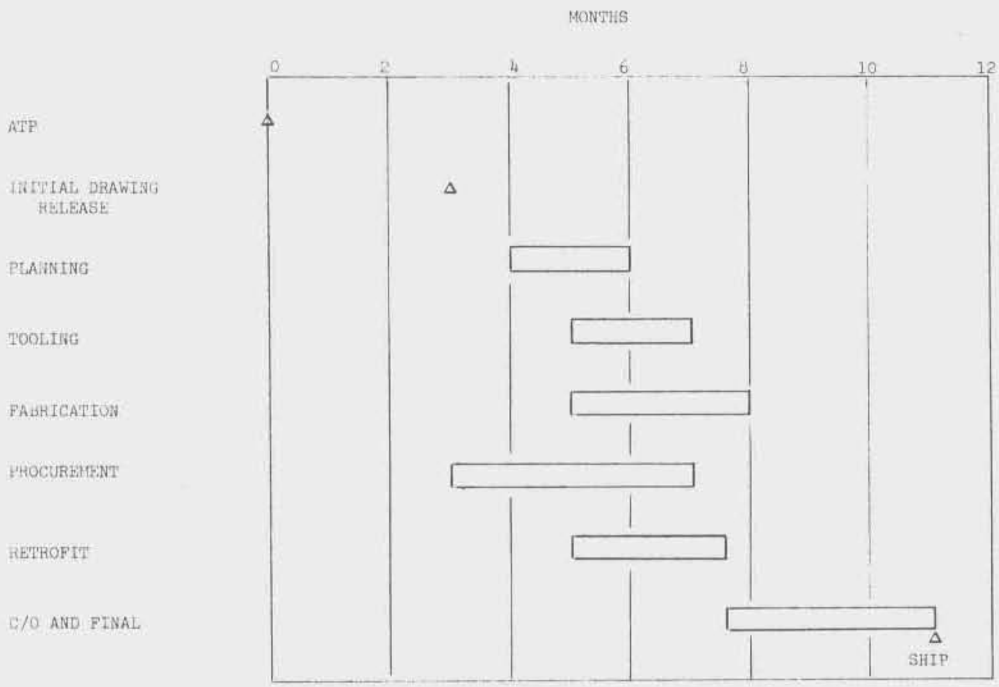


Figure 6.2-3. INT-20/S-IVB Retrofit Schedule

The post-retrofit checkout and final closeout for shipment leads to a stage ready for shipment to KSC at about eleven months after ATP. This is approximately one year earlier than for the in-line case presented in Figure 5. 5. 3-1.

6. 2. 3 Cost Plan

The increment in non-recurring and recurring costs for the retrofit INT-20/S-IVB case is presented in Table 6. 2-I. The non-recurring costs are those over and above the previously incorporated in-line non-recurring costs ($2, 943, 000). The recurring costs for one stage include charges for component removal and the new attach ring, and a credit for not purchasing 4 retrorockets, resulting in a new credit from the basic S-IVB stage cost.

The added cost for component removal in the retrofit case results in salvaging of some $150, 000 in parts that may be useful as spares for a concurrent Saturn V program. As Table 6. 2-I indicates, the deletion of the retrorockets dominates the recurring costs and in themselves result in a net decrease in total retrofit costs.

Table 6.2-I

COST FOR RETROFIT OF ONE SATURN V/S-IVB STAGE TO INT-20/S-IVB CONFIGURATION

Non-Recurring Costs (incremental from in-line costs)	
Planning	$ 5,300
Manufacturing Set-Up	2,900
Total Non-Recurring	$ 8,200
Recurring Costs (total for 1 stage)	
Component Removal	$ 8,400
Interstage Attach Ring	1,800
Retrorockets (deletion)	(33,900)
Total Recurring	$ (23,700)

6.3 IU RETROFIT PLAN

In evaluating the impact on the INT-20 IU program of requiring a retrofit of Saturn V IU's, it was possible to draw heavily on the Retrofit Analysis, J-2S Improvement Study, the Boeing Company Document D5-15772-8. Several key points made in that study remain valid. J-2S engine imposed changes were confined to the Flight Control Computer and the Control Distributor which is identically the case for the INT-20 IU although the nature of the changes are different. It was ground ruled that retrofit would not occur until after the facility had gone through PDP and delivery on at least one J-2S IU. With the same ground rule on the INT-20 program, the first engineering release of drawings to INT-20 (1) would be available and the following events would then be admissible as in the case of the J-2S retrofit program.

The total effort required to remove an IU from storage and the validation of flight readiness is basic; irrespective of modifications directed to a particular IU being removed from storage. IU refurbish requirements are detailed in Table III of IBM Drawing No. 7915953, Long Term Procedure for S-IB/V Instrument Unit Storage. These requirements are listed by system (measuring, telemetry, radio frequency, guidance and control, environmental control, electrical and structural) and further detailed by component. This table also takes into consideration the effect duration of storage (6, 12, 18 and 24-30-36 months) has upon the refurbish requirements.

In addition to a detailed description of the affected component (P/N, panel location, storage life, age critical parts, etc.); specific action required to refurbish is listed and detailed inspection requirements are indicated.

The following represents a brief summary by system of basic effort required to be performed on IU components upon being removed from storage.

I&C - Remove and check all accelerometers, pressure transducers, flowmeters, DC converters, channel selectors, all TM components. TM components will have a solder joint inspection performed. All RF components will be removed for visual inspection. An Acceptance Test Procedure will be performed on all removed components.

G&C - Remove all G&C components. The following are stored separately from the IU structure and do not require removal: LVDA, LVDC, ST-124, FCC, EDS Rate Gyro, CSP and Control Accelerometer. The FCC and CSP are returned to the vendor for complete breakdown and inspection. AZTEC compatability is performed on the LVDA and LVDC; an FCP on the ST-124; an Acceptance Test Procedure on the EDS Rate Gyro and an Acceptance Test Procedure is performed by the vendors on the FCC and CSP.

6.3 (Continued)

Electrical - Remove all components except J-box and network cables. Remove covers from distributors, 56 Volt Power Supply and Switch Selector. Perform component test on all components except J-Box.

Environmental Control - Remove the Water Methanol Accumulator, Coolant Check Valve, Gas Bearing Solenoid Valve, Pneumatic Filter, Water Accumulator, Quick Disconnects, Electronic Controller, Temperature Sensor, Gas Bearing Regulator, First Stage Regulator. Component Disassembly will be performed on the Water Methanol Accumulator, Heat Exchanger and Water Accumulator. An Acceptance Test Procedure will be performed on all removed components.

The IU retrofit manufacturing effort can be described in terms of three distinct phases: Disassembly, Reassembly and Preparation for Shipment. Nominal time periods for the completion of each phase are as follows:

Disassembly - 24 weeks. This phase involves removal of component hardware for retest and refurbishment. These removals must begin 46 weeks prior to the scheduled shipping date of the IU Assembly. This lead time is based on the refurbish requirements for IU component hardware as contained in Table II of IBM Long-Term Storage Procedure Document No. 7915953. This lead time further considers the refurbish requirements after a storage period of 24 months. The lead time for hardware removal is a requirement independent of any subsequent hardware modification requirements. It should be noted however, that this lead time is adequate to accomplish those component modifications necessary due to INT-20 or mission application.

Reassembly - 12 weeks. This phase involves IU structures assembly alignment and reinstallation of cables, thermal conditioning panels and all other component hardware. Required component acceptance testing will begin during the Disassembly phase and will be completed concurrent with completion of the Reassembly phase of manufacturing operations. All systems are installed in readiness for IU systems checkout at the end of the Reassembly phase.

Preparation for Shipment - 2 weeks. This phase is accomplished subsequent to completion of IU systems checkout which requires eight weeks. It involves the removal and packaging of selected flight hardware components and assemblies for separate shipment and otherwise securing the IU stage for shipment.

6.3 (Continued)

The manufacturing retrofit effort, generally described above will be controlled by manufacturing routings which outline, step-by-step, the procedure to accomplish all the discrete operations required, including the essential inspections. The manufacturing routings are machine prepared and afford the flexibility of being responsive to changes in manufacturing instructions as brought about by engineering releases of new or changed requirements. There are no new tooling or fixture requirements for retrofit to the INT-20 configuration.

Note that as stated, the FCC is stored separately and returned to the vendor for complete breakdown and assembly. It is at that time that the minor changes required in the baseline INT-20 IU design would be made. It was pointed out in the inline baseline INT-20 discussions that all other software and hardware modifications are within the normal mission-to-mission or vehicle-to-vehicle change activity. It is therefore procedurally the same to take an IU from storage for redelivery as a Saturn V IU or an INT-20 IU. Additionally, it should be pointed out that in general, block changes accumulated since the Saturn V IU was fabricated and placed in storage must be released through the engineering release system. INT-20 IU changes are simply integrated within the same process.

The schedule for retrofit is unchanged from the Saturn V redelivery schedule. Costs incurred for removal from storage or for final systems test will not be charged as a delta to the INT-20 program. Only those delta costs are identified where a component is designated for retrofit as well as refurbishment. An example of this would be the Flight Control Computer which would be returned to the vendor for refurbishment upon removal from storage but will also require retrofit for a particular INT-20 mission.

SECTION 7
COST REDUCTION

7.0 GENERAL

Efforts have intensified to identify areas for cost reduction in the follow-on Saturn V program. Cost reduction should be achieved by decreasing the cost of material, labor and overhead by improved management techniques, procedures and processes without sacrificing quality and reliability. Great cost reduction potential lies in reducing program support which reduces the labor cost and the associated overhead cost. A prime factor in cost reduction is a change from an R&D (Development) philosophy to a "production" philosophy for stage production. This change to "production" philosophy would be directed toward producing a standard "no engineering change" vehicle.

This approach removes all of the design and development engineering, most of the software and some of the testing, checkout and quality activity. The only changes left in the program would be those caused by mission peculiar requirements. A standard "no engineering change" vehicle would maintain presently attained quality and reliability with significantly reduced test effort and fewer program management controls. Static firings are not required for stages whose maturity has been certified by a substantial number of successful static firings and flights during the R&D period. These reductions associated with the "production" philosophy approach, reduce labor cost.

These approaches have been applied to Saturn V cost reduction. The results are that Saturn V launch vehicle configuration and program costs can be significantly reduced while providing program support without reducing overall reliability and quality of the flight hardware.

Two other areas that significantly reduce costs are (1) maintain continuous production to avoid start-up and shutdown costs; and (2) maintain production and launch at a rate that is efficient with respect to existing facilities and man-loading.

7.1 S-IC COST REDUCTION

The Boeing Company has just completed cost reduction analyses for the S-IC stage. S-IC configuration simplification includes using steel fasteners instead of Titanium fasteners, and elimination of upper cantilever fabbles, four retromotors, some telemetry, prevalves and titanium engine shrouds. Manufacturing cost is reduced by producibility changes. Redundant testing is reduced or eliminated.

Boeing cost reduction data has been presented to NASA. The NASA target of 50 percent cost reduction appears to be achievable for the S-IC stage with present safety and reliability retained.

7.1 (Continued)

The Rocketdyne Division of North American Rockwell has made cost reduction evaluations. The Rocketdyne cost reduction results have been provided to NASA. Preliminary data showing cost reduction potential is shown below for F-1 and J-2 engines at a rate of 2 Saturn Vs per year.

COST (Dollars in Millions)	Present Cost	Reduced Cost	% Reduction
F-1 Hardware (5 flight engines)	10.46	8.50	19%
F-1 Rocketdyne Support	9.48	3.82	60%
Total F-1	19.94	11.32	44%
J-2 Hardware (6 flight engines)	10.50	7.80	26%
J-2 Rocketdyne Support	10.91	4.13	62%
Total J-2	21.41	11.93	44%

All cost reduction techniques applicable to the Saturn V are equally applicable to the INT-20.

7.2 S-IVB COST REDUCTION

Extensive cost saving potential is available for the S-IVB Stage. These savings originate from three specific areas: (1) a reduction in stage requirements for the logistics resupply mission; (2) the incorporation of the J-2S engine; and (3) programmatic simplifications. Each of these areas has been under continual study in the past several months at MDAC.

Assuming that the mission requirement for the INT-20 launch vehicle is restricted to injecting the payload into low earth orbit, the S-IVB stage merely acts as a velocity increment stage with no requirements for orbital attitude control, zero g propellant management, or extended coast. This results in a reduced stage life time of approximately 10 minutes. An in-house study is currently being completed to define the stage simplifications for this particular mission. This Low Cost S-IVB Study has identified the following stage subsystem modifications resulting from these reduced mission requirements.

- Gas bleed roll control system replaces both APS units
- Non-propulsive venting system deletion
- Extensive reduction in RF and telemetry systems
- Deletion of pneumatic control system
- Replacement of the propellant utilization system by point level sensors
- Extensive reduction in the electrical control and power distribution systems
- Simplification of the wiring installation
- Replacement of the cold plate system with a single ambient panel
- Simplification of the hydraulics, ordnance and environmental control systems

Replacing the J-2 engine with a J-2S engine results in several other cost saving S-IVB Stage system modifications. The use of the J-2S engine permits deletion of the chilldown system and the ullage control rockets. In addition, the LOX low-level sensors can be deleted and the electrical power requirements are considerably reduced.

With respect to programmatic simplifications, considerable cost reduction can be realized as described in the follow-on procurement studies currently being performed by all Saturn V contractors. Applicable portions of this study are being incorporated into the definition of the Low Cost S-IVB Stage.

Preliminary costing information indicates that use of the Low Cost S-IVB configuration on the S-IC for the space station logistics resupply vehicle results in a stage delivery cost reduction of between 40% and 50% of current S-IVB costs.

7.3 IU COST REDUCTION CONSIDERATIONS

The groundrules for the study of the defined Baseline INT-20 vehicle admit to use of the Saturn V IU as configured for AS-511 effectivity. Secondly, it has been assumed that the Saturn V/INT-20 IU program would be identical to the contractual and procedural methods used on the AS-501 through AS-515 Saturn V program.

General cost reduction potential ideas have been promulgated and are briefly summarized in the following paragraphs.

The primary function of the Instrument Unit is to provide Guidance, Navigation, Control and Sequencing for the vehicle. Power, thermal control, telemetry and structural and/or packaging considerations are ancillary. The primary functional requirements are combined into the Inertial Platform Launch Vehicle Digital Computer and Data Adapter, the Flight Control Computer and the Switch Selector. (Note that the Switch Selector in the IU is identical to one in each of the lower stages.) The platform and LVDC/LVDA together represent approximately 70 percent of the hardware costs of the IU including the structure. The technology would support immediate replacement of these equipments at the expense of incurred development costs. Lesser return can be expected from other equipment substitutions. IBM has made in-house studies which indicate that the performance of the Saturn V IU can be retained with a unit cost reduction of 50 percent at the expense of approximately 20 million dollars non-recurring development cost. If the conversion were made to the Saturn V IU, the IU would be directly interchangeable with the INT-20 IU with all mission imposed changes residing in the flight software.

The foregoing proposal assumed Saturn V, LOR capability. If the INT-20 mission were in fact restricted to 100 n.m. insertion as baselined in this study, the following drastic reductions in capability may be considered:

Simplified computer with smaller memory.

Less accurate inertial platform.

No thermal control.

Reduced power.

Simplified telemetry.

Removal of the command links.

Removal of tracking system.

7.3 (Continued)

This reduction could be made as a dedicated design or be a pared down subset of the newly configured Astrionic System.

By retaining the Saturn V IU (Reference AS-511) for LOR, Synchronous and LEO missions a stripped down version for 100 n.m. insertion can be configured which allows limited lifetime reconfiguration.

Studies are currently in progress which propose to reduce costs by attacking the programmatic methodology and scrubbing down specifications based on known flight experience. In this regard, whether an IU is newly designed or entering a mature operational phase as in the case of the Saturn IU, at the point of departure from the R&D or development phase, the procurement can be based on a "frozen design", with explicit definition and separate contracting of mission-to-mission and/or vehicle-to-vehicle changes. Expected savings on a unit cost basis are to approach 50 percent but in the case of the IU are limited by Guidance and Control costs as previously pointed out.

SECTION 8

SYNCHRONOUS/POLAR MISSION REQUIREMENTS

8.0 IU REQUIREMENTS

8.1 IU SUBSYSTEM EFFECTS - SYNCHRONOUS MISSION

The INT-20 effects on the IU described in paragraphs 4.2.3 and 4.2.5.3 a and b are also required for the Synchronous Orbit Mission. The additional requirements are brought about by the mission only, not the INT-20 baseline. The added requirements are defined in detail in the assessment of Astrionic System and the IU Impact for J-2S Engine Implementation on Saturn V Vehicles, Reference 8.1-1, and are summarized as follows:

8.1.1 Lifetime Extension

To extend the lifetime of the IU electrical power to 15 hours, it is recommended that the three 350 ampere-hour batteries of the baseline IU be replaced with four redesigned 470 ampere-hour batteries. This approach will require development and qualification of the new batteries.

To extend the lifetime of the GN_2 for the ST-124M platform, an additional Gas Bearing Supply (GBS) panel will be located adjacent to the present one. This will provide a total of four cu ft of GN_2 which is sufficient for this mission.

The lifetime of the GN_2 required for TCS pressurization will be extended by redesigning the orifice regulator.

8.1.2 Communication Requirement

To meet the communications requirement imposed during the Hohmann transfer, a configuration employing six of the present CCS directional antennas is recommended. The selection of a single antenna for "pointing" purposes will be controlled by the LVDA/LVDC and Switch Selector. To maintain satisfactory circuit margins, two modified Power Amplifiers are used. A power divider and coax switches are added to permit antenna selection.

8.1.3 Yaw Requirement

There will be no hardware modifications to meet the large yaw requirement of certain synchronous orbit missions. This requirement will be met by using a yaw bias technique. This technique involves intentional misalignment of the ST-124M platform to take advantage of the available 90 degree range of yaw.

8.1.4 Structural Requirements

The addition of the Gas Bearing Supply (GBS) panel requires structural modifications to delete the thermal conditioning panel brackets in location 23 and to add GBS panel brackets. The addition of the GBS panel brackets requires core modification in the basic structure to prevent core crushing.

8.1.4 (Continued)

The requirement of positioning the six CCS Directional Antennas is that the antennas be positioned equally spaced around the IU. These antennas cannot be located in the umbilical or access door location because of interface with GSE. Because of this and because of the antenna spacing requirements, the only feasible locations for the antennas are locations 1-2, 5-6, 9-10, 13-14, 17-18 and 21-22.

It should be understood that the IU is divided into 24 locations which are numbered 1 through 24. A designated location such as 1-2 indicates the position midway between locations 1 and 2. It should be noted that positions midway between locations are required for antenna installation because the coaxial load must be accessible so that major disassembly of the IU is not required when attaching or removing the coaxial leads from antennas.

The baseline IU configuration has a CCS directional antenna in location 2-3. This antenna must be relocated. Five additional CCS Directional Antennas are required.

The CCS antenna mounting provisions will be cutouts in the structure for the coaxial leads and inserts in the structure for mechanical fasteners to assemble the antenna to the structure.

The baseline IU configuration has a VHF TM antenna in location 9-10 and in location 21-22. Since it is not possible to locate both a VHF and CCS antenna in each of these locations, the VHF TM antennas must be relocated. The only possible areas for relocation are 10-11 and 22-23, 14-15 and 2-3 or 15-16 and 3-4 because (1) the two VHF antennas must be located approximately 180 degrees apart and (2) it is not desirable to locate antennas on the IU splice joints. Locations 10-11 and 22-23 are proposed because they represent the least impact on electrical interconnections.

It should be noted that the bulk of the effort is brought about by the communication requirement and its effect on the structure. The INT-20 has very limited payload capability to Synchronous altitudes and therefore it is doubtful it will be manned. This may relieve the continuous communication requirement and thus eliminate the bulk of the effort. However, it should also be considered that the nature of the payload may require booster communication, therefore the magnitude of effort is somewhat open and will have to be determined when a firm definition of the payload becomes available.

8.1.5 Software Requirements

The software impact is the sum of baseline INT-20 impact plus the software modifications detailed in Reference 8.1-1. The Synchronous Orbit associated modifications are summarized as:

8.1.5 (Continued)

CCS antenna switching.

Added time bases for S-IVB restarts.

Transformations to allow yaw biasing.

8.2 IU SUBSYSTEM EFFECTS - POLAR ORBIT MISSION

The INT-20 effects on the IU described in paragraph 4.2.3 and 4.2.5.3 a and b are also required for the Polar Orbit Mission. The additional requirements are brought about by the mission only, not the INT-20 baseline. The added requirements are defined in detail in the Assessment of Astrionic System and IU Impact for J-2S Engine Implementation on Saturn V Vehicles, Reference 8.1-1, and are as follows:

8.2.1 Yaw Requirement

There will be no hardware modification to meet the large yaw requirement of the Polar mission. This requirement will be met by using a yaw bias technique. This technique involves intentional offset of the ST-124 platform to take advantage of the available 90 degree range of yaw.

8.2.2 Software Requirements

The software impact is the sum of the baseline INT-20 impact plus the software modifications detailed in Reference 8.1-1. The Polar Orbit associated modifications caused by deletion of the S-IVB stage in Reference 8.1-1 should be ignored, however the transformation to allow yaw biasing is valid.

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FOREWORD

The ten-month "Saturn V Derivative (S-IC/S-IVB) Launch Vehicle System" study program was performed under National Aeronautics and Space Administration Contract NAS8-30506. The study effort was supervised and administered by the Marshall Space Flight Center.

The purposes of this study were to provide a preliminary design and analysis of an S-IC/S-IVB/IU (or INT-20) intermediate launch vehicle, and to estimate the resources required to design, develop, and produce the INT-20. The Boeing Company, Southeast Division, performed overall vehicle and S-IC stage studies and was the study integrator. Subcontractual assistance was provided by the Federal Systems Division of the International Business Machines Company (IBM) on astrionics systems and the McDonnell Douglas Astronautics Company (MDAC) on the S-IVB stage.

The North American Rockwell Corporation's Rocketdyne Division provided F-1 engine data for the study; its Space Division supplied Service Module data. The Convair Division of General Dynamics Corporation contributed data on the Centaur stage.

Two methods of vehicle implementation were studied. These were the incorporation of INT-20 stages and requirements in the Sat V assembly line and launch facilities (in-line) and the conversion of stored Saturn V stages (retrofit) to INT-20 use

The ability of the INT-20 to handle the Big Gemini logistics payload was also investigated.

A companion study, "KSC Facilities and Operations for Saturn MS-IC/MS-IVB (Intermediate 20) Launch Vehicle", was performed by Boeing for NASA/KSC under Contract NAS10-6163. This Kennedy Space Center study examined the technical and economic requirements necessary to adapt Launch Complex 39 to prepare and launch the INT-20 vehicle. A synopsis of results of the KSC study is included in this report.

The results of this INT-20 study are contained in three documents:

D5-17009-1	Executive Summary
D5-17009-2, Vol. I	Final Technical Report
D5-17009-2, Vol. II	Appendices

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SECTION 1

INTRODUCTION

1.0 GENERAL

The National Aeronautics and Space Administration has studied several launch vehicles with payload capabilities in the "intermediate" range between those of Saturn IB and Saturn V. The S-IC/S-IVB/IU launch vehicle is one of these. A conceptual feasibility study of an S-IC/S-IVB/IU, or INT-20, launch vehicle was performed in 1966 (Reference 1.0-1). The 1966 study showed that the INT-20 could be used to satisfy potential "intermediate payload" requirements. However, further definition of the INT-20 was necessary to provide the data and information needed to thoroughly evaluate the vehicle for use in potential future manned and unmanned mission applications. This study was intended to provide this detailed data and information.

1.1 PROGRAM DESCRIPTION

The INT-20 launch vehicle consists of a combination of the Saturn V's S-IC and S-IVB stages and Instrument Unit. Several variants of this vehicle are possible. These are obtained by varying the number of F-1 engines on the S-IC and/or increasing the peak vehicle axial acceleration from 4.68 g's (Saturn V design value) to 6.0 g's (limit based on overall vehicle structural integrity). The resulting INT-20 configurations encompass a very wide range of payload performance capabilities.

The eight basic INT-20 configurations (2, 3, 4, or 5 F-1's, 4.68 or 6.0 max g's each) were evaluated in terms of technical feasibility, development cost, and performance. A baseline vehicle was selected; a preliminary design made of it; and the resources required for its development and implementation estimated.

The study was performed in a time span of 10 months -- eight months of technical activity and two months of final documentation. The Boeing Company, as prime contractor, performed the S-IC stage and vehicle analysis/design tasks, the S-IC stage and vehicle resources tasks, and integrated the study efforts. The McDonnell Douglas Astronautics Company (MDAC), under subcontract to Boeing, performed the S-IVB stage design/analysis and resources tasks and was responsible for defining the S-IC/S-IVB interface. The Federal Systems Division of International Business Machines Company (IBM), also under subcontract, performed the Instrument Unit and stage astrionics systems design/analysis and resources tasks.

1.2 STUDY OBJECTIVES

The objectives of this study were:

a. To delineate the preliminary design of an S-IC/S-IVB/IU launch vehicle system most responsive to a wide variety of manned and unmanned mission requirements.

b. To provide a Design, Development, Test and Evaluation (DDT&E) Plan, for development and implementation of the INT-20.

1.3 STUDY APPROACH

The study effort was divided into four phases to meet the objectives of the study:

a. Phase I was a configuration evaluation culminating in the selection of a baseline INT-20;

b. Phase II was a technical analysis and preliminary design of the selected baseline;

c. Phase III was a resources analysis to define the Design, Development, Test, and Evaluation (DDT&E) Plan for the baseline; and

d. Phase IV was the study documentation effort (oral presentations, status reports, final report).

Several supplementary tasks were performed during Phase II. These were not essential for meeting the stated study objectives, but provided data useful for INT-20 evaluation. The studies were a payload/wind sensitivity study; a cursory investigation of the use of the Big G logistics spacecraft on INT-20; an evaluation of the complete removal of restart from the S-IVB stage; a study of an improved (digital) flight control system; and an evaluation of the performance gains made by the use of J-2S on the INT-20 S-IVB stage.

1.3.1 Phase I Trades

The delineation of an INT-20 preliminary design required a specific vehicle configuration definition. Such a definition resulted from cursory performance, technical feasibility, and cost analyses of the eight basic configurations. Use of either Saturn V or Saturn IB hardware was also traded. A baseline launch vehicle configuration meeting certain selection criteria was identified and defined for NASA/MSFC approval.

1.3.2 Phase II Analysis/Design

The baseline configuration was studied in detail during Phase II. Following configuration approval by NASA/MSFC, a set of design criteria were generated. These included aerodynamics, weights, a baseline trajectory, loads, heating and acoustic environments, and preliminary flexible body controls data. The detailed stage and I.U. design and design analysis efforts were conducted with these design criteria as a basis.

Design of the INT-20 hardware considered minimum structural change and maximum use of existing components. Vehicle/stage design considered both in-line and retrofit implementation. Where requirements differed for the same hardware, the differences were noted.

The payload/wind sensitivity analysis was performed by Boeing to show the influence of winds (expected peak wind speed in month of launch) on the permissible overall payload length, or payload density, allowed on the INT-20. Loads were prepared for various payload weights/lengths and the critical vehicle stations were identified.

The INT-20/Big G analysis was conducted by Boeing to show the logistics support capabilities of the INT-20 and to compare critical vehicle characteristics with those of the baseline.

Since the baseline vehicle would be designed for a 100 N.M. circular orbit payload (and trajectory), an analysis was conducted by MDAC to ascertain if the effects of removing restart capability from the S-IVB stage would be beneficial. Two methods were investigated: removal of only enough equipment to enable subsequent addition of restart if desired; and the complete removal of S-IVB restart capability, which would delete additional weight but not allow reverting to a restartable stage configuration.

The use of a digital control system that would absorb many functions of the Flight Control Computer into the Launch Vehicle Digital Computer/Data Adapter was compared to the existing I.U. attitude control system in terms of function, versatility, and cost.

1.3.3 DDT&E Plan Development

The development and implementation requirements (DDT&E Plan) for the baseline were prepared using in-line implementation as the basis for resources estimates. The DDT&E Plan was comprised of design, test, manufacturing, facility, schedule, and cost plans. Ground rules were prepared based on using the National Space Booster Study cost data as a base (Reference 1.3.3-1), and using projected Saturn V schedules to derive INT-20 schedules. The plans were developed by assessing the stages and I.U. and showing implementation requirements. In the case of hardware, each

1.3.3 (Continued)

contractor determined the delta cost between Saturn V hardware and the corresponding INT-20 hardware. Basic Saturn V design and implementation philosophy was used. A separate plan was prepared for retrofit stages

1.4 CONSTRAINTS AND GUIDELINES

The NASA/MSFC-approved constraints and guidelines under which the study was conducted are presented below.

a. General

1. The baseline Saturn V launch vehicle from which the INT-20 will be derived is AS-511, using a J-2 engine on S-IVB.

2. Apollo-Saturn V design criteria will be used, except where otherwise specified or approved by NASA/MSFC.

b. Analysis/Design

1. The vehicle will be designed for a 100 NM Earth-orbit mission with the maximum payload envelope determined within the structural limits of the manned and unmanned requirements. The structural factor of safety for manned applications is 1.4, and for unmanned applications, 1.25.

2. Basic investigations will use an MSFC double-angle (MLV) nose cone above the final stage/IU combination, with a 260-inch diameter cylindrical section between the cone and the IU as required.

3. Distribution of mass within the external payload envelope is assumed to be uniform.

4. Nominal wind assumptions will be furnished by MSFC and are consistent with Apollo wind restrictions:

   (a) 99.9% probability on pad;

   (b) 99% probability during lift-off (twenty seconds);

   (c) 95% probability during powered ascent with 99% wind shear; and,

   (d) Gust conditions as specified by MSFC.

1.4 (Continued)

5. Atmospheric model and geopotential function will be provided by NASA/ MSFC.

6. A flight performance reserve of 3/4% of the total vehicle characteristic velocity will be provded for in the last stage. These reserves will be considered as part of the usable mainstage propellants.

7. Initial thrust-to-weight ratio at launch is to be held at 1.25, if possible; exceptions are to be noted.

8. Trajectory and propellant distribution procedures will be compatible with methods in use at MSFC. Detailed assumptions during ascent trajectory will be approved by MSFC.

9. A nominal launch azimuth of 90° will be used.

10. Modes of rigid body control will be a minimum control frequency of 0.15 Hertz and a damping 75% of critical. The gains will be chosen such that a most desirable compromise will exist with respect to lateral drift, gimbal angle requirements, and maximum dynamic pressure (q).

c. Resources

1. The program outlined to qualify the vehicle for operational flights shall include all facility modifications, hardware, and test operations for all necessary ground testing (all-systems tests, dynamic test vehicle, injection stage test, etc.).

2. Man-rating is required.

3. Funds will be assumed available as required.

4. The Saturn V INT-20 Program will not interfere with the existing Apollo delivery schedule.

5. A program definition phase (PDP) of at least six months will be required prior to stage development

6. Stage development time will be consistent with completion of a test program.

7. Scheduling will not be calendar-oriented but will be based upon an assumed first flight (mid-1974) and appropriate time-phasing to launch.

1.4 (Continued)

8. Current stage acceptance test firing cost will be identified separately.

9. Maximum use will be made of existing facilities and tooling.

10. Cost analyses will be separated into two parts, (1) Non-Recurring or Development Costs including design, development, test and evaluation activities, (2) Recurring or production costs.

11. Costs and schedules will be based on a one-shift, a five-day week for engineering and a two shift, five day week for manufacturing.

12. All stage, Instrument Unit and engine costs will be based on learning curve percentages, which will be coordinated with NASA.

13. Cost estimates will be in 1968 dollars without inflationary factors applied.

14. Spare parts costs will not be used

15. Logistics planning is included in stage costs.

16. Costs will be total costs to the government, including all overhead and fee. All government manpower and transportation costs will be excluded.

17. Costs of stage static test will be identified as a separate entity.

18. The study cost numbers will be based on those presented by the National Space Booster Study.(Reference 1.3.3-1)

SECTION 2

SUMMARY, CONCLUSIONS AND RECOMMENDATIONS

2.0 CONCLUSIONS AND RECOMMENDATIONS

The 4 F-1 S-IC/S-IVB/I. U. vehicle was designated the baseline vehicle at end of the Phase I trade study. Further analysis and design during Phase II provided a refined, final INT-20 configuration that met the study objectives of best vehicle performance with least overall vehicle impact and cost.

2.0.1 Recommended INT-20 Configuration

The INT-20 launch vehicle with 4 F-1 engines on S-IC, 500-series (Saturn V) S-IVB stage and Instrument Unit, a 5-inch adapter ring between the interfaces of the S-IC and S-IVB stages, and a controlled-acceleration trajectory emerged as the recommended final design. This configuration, with the baseline MLV payload shape, is depicted in Figure 2.0.1-1. The Apollo and Big Gemini (Big G) manned logistics payloads which the vehicle can accommodate are also shown.

A summary of the recommended system design is shown in Figure 2.0.1-2. The recommended system configuration is designed to be capable of delivering a 126,000 lb. manned payload to low Earth orbit, or 132,000 lb. unmanned. The I. U. retains its S-II-associated circuitry to simplify changeover. The S-IVB will have restart removed for the nominal Earth-orbit mission, but retains the capability for simple addition of this feature when needed. The small adapter ring used between the S-IC stage and the S-IVB aft interstage eliminates the need to change the Saturn V-configured bolt-hole patterns on each interface and simplifies retrofitting existing hardware into an INT-20 vehicle. The center engine is removed from the S-IC first stage. The basic vehicle height is about 200 ft. (61 m). The payload length will vary depending on manned/unmanned mission restraints and the inflight loads caused by inflight design winds. The manned MLV payload limit is 43.2 ft (13.2 m), based on a standard March 95 percentile inflight design wind profile.

The INT-20 vehicle is trimmed to match the payload mission requirements and vehicle structural capabilities by flying controlled-acceleration trajectories, as shown in Figure 2.0.1-3. For manned missions (structural factor of safety of 1.4), an axial acceleration peak of 3.68 g's is reached at cutoff of the first pair of F-1 engines (limited by S-IC RP-1 bulkhead structural capability). A second peak of 4.68 g's occurs at S-IC shutdown. For unmanned missions, the two peaks are 4.05 g's and about 5.3 g's (depending on the payload weight), respectively. Note that the S-IC burntime for INT-20 is 230 to 240 seconds, compared to 160 seconds for Sat V.

The unmanned mission capabilities of the two-stage INT-20 can be enhanced through the use of Centaur and Service Module injection stages.

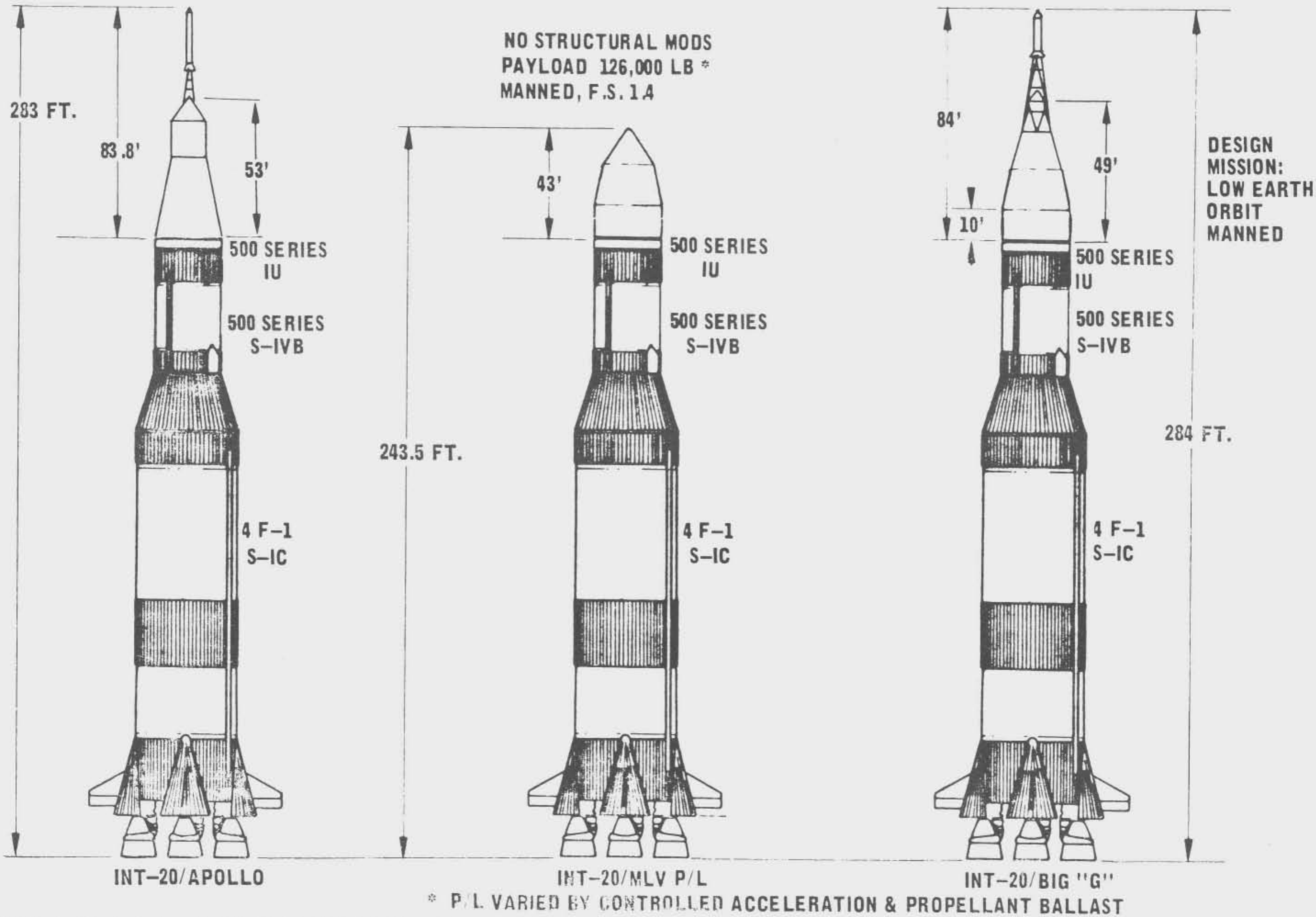
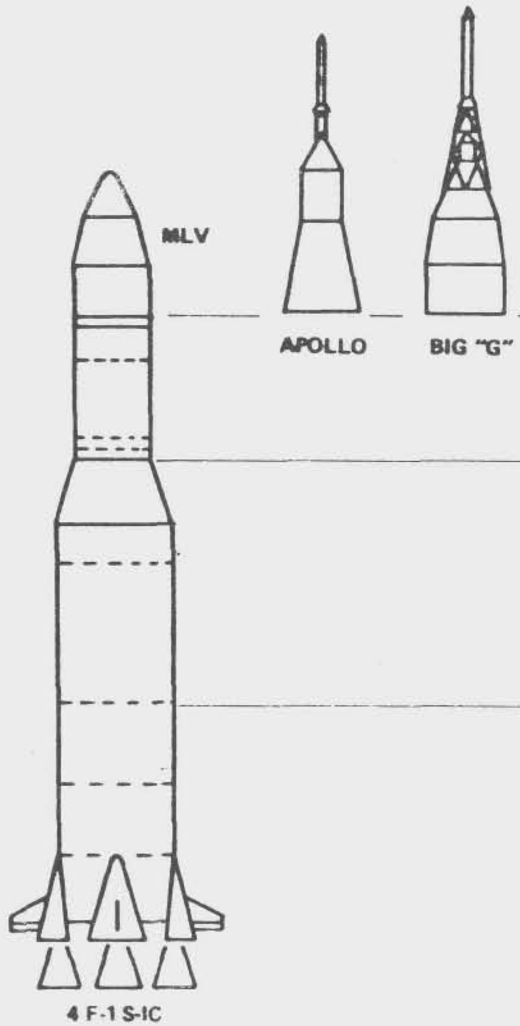


FIGURE 2.0.1-1 CHOSEN CONFIGURATIONS



	MANNED		UNMANNED
PAYLOAD	126,000 LB	MAX. PAYLOAD	132,000 LB
	1.4 F. S.	MARCH WINDS	1.25 F. S.
THIRD STAGE			DESIGN SUITABLE FOR SM & CENTAUR TO EXPAND MISSION VERSATILITY
I. U.		500 SERIES RETAIN S-II CIRCUITS	
S-IVB		500 SERIES RETAIN ABILITY TO INCLUDE RESTART & MATE WITH S-II	
S-IC		(4) F-1 ENGINE	
	3.68 g's	FIRST ACCEL. PEAK	4.05 g's
	4.68 g's	SECOND PEAK	~ 5.3 g's
LC-39		USE EXISTING FACILITIES	

FIGURE 2. 0. 1-2 RECOMMENDED SYSTEM CONFIGURATION SUMMARY

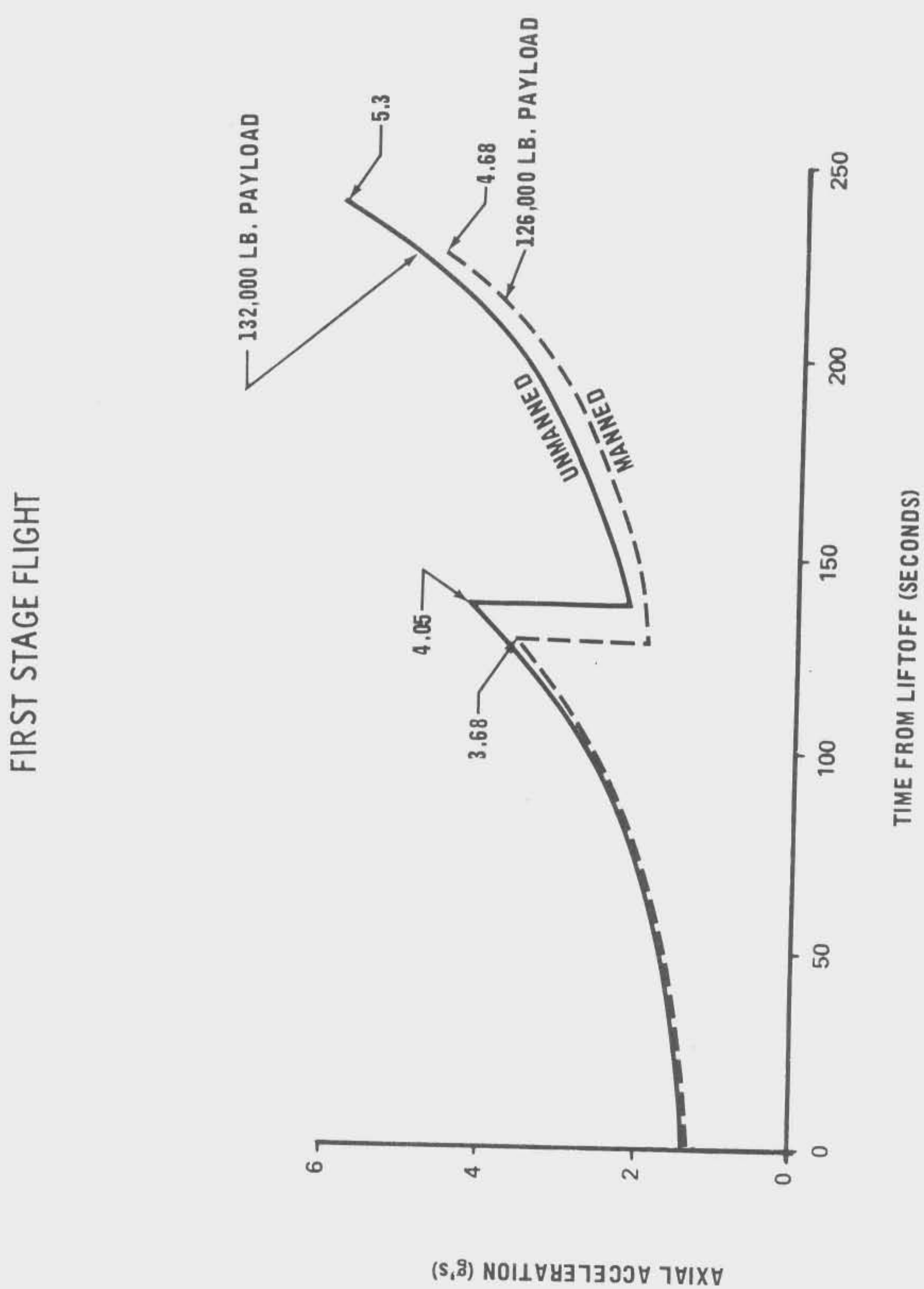


FIGURE 2.0.1-3 INT-20 CONTROLLED ACCELERATION

2.0.1 (Continued)

The Kennedy Space Center Launch Complex 39 facilities, with some modification, will be used to process and launch the INT-20. The KSC adaptations will allow launching a mix of INT-20's and Saturn V's. The KSC equipment that needs to be modified to enable LC-39 to accommodate both INT-20 and Saturn V include one Mobile Launcher, one VAB high bay, one Launch Control Center firing room, and the Mobile Service Structure (MSS). The single MSS must be adapted for convertible use by both Saturn V and INT-20.

2.0.2 Conclusions

The study program resulted in the following conclusions:

a. The recommended INT-20 vehicle design is capable of a wide variety of manned and unmanned missions using only current hardware and launch facilities.

b. Implementation of INT-20 will result in a more efficient use of the Saturn V product base and launch facilities.

c. Development costs for the INT-20 vehicle and launch facilities are extremely low at $7.5M.

d. Recurring costs will benefit from the current strong Saturn V cost reduction effort since all INT-20 components and services are directly derived from the Saturn V.

e. Adapting LC-39 will lead to a maximum capability of up to twelve INT-20 launches per year or a mix of ten to eleven Sat V and INT-20's per year, depending on payload used.

f. INT-20 is readily available in that only 18 months are required to retrofit stored stages or Sat V production can be expanded to include INT-20 vehicles in 36 months.

g. INT-20 mission capability and versatility can be expanded by using either Centaur or the Service Module as a third stage. When J-2S engines become available they too will increase INT-20 capability.

2.0.3 Recommended Test Program

The contracted effort was comprehensive enough so that recommended future activity is limited to accomplishing the (government furnished) laboratory test program. These tests include:

2.0.3 (Continued)

a. Wind tunnel force and pressure model tests for various payload shapes.

b. A flexure model test which with analysis is accomplished in lieu of full scale vehicle dynamic testing.

c. The F-1 engine extended burntime test at the MSFC single engine test stand.

All other work requires a specific mission definition and assignment and can therefore wait for the Program Definition Phase.

2.1 BASELINE VEHICLE SELECTION

The Phase I trades resulted in the selection of the 4.68 g, 4 F-1 INT-20 as the baseline vehicle (center in Figure 2.0.1-1). Since development requirements and costs are roughly the same for all configurations studied, the selection was made on the basis of vehicle performance and compatibility with existing spacecraft. The 4 F-1 vehicle had sufficient payload capability to support manned orbital and unmanned lunar logistics missions. The 5 F-1 configurations also had these capabilities, but were not recommended because they experience relatively high dynamic pressures.

2.2 DESIGN CRITERIA

Design criteria were prepared for the selected baseline in support of the Phase II design activity. As noted in Table 2.2-I, the loads and controls parameters are similar for Sat V and INT-20. The lower face of the S-IC base heat shield experiences a lower peak temperature. Aerodynamic heating on the forward skirt is slightly higher for INT-20 as a result of the change in section immediately adjacent to the S-IC forward skirt. Structural strength at this slightly elevated temperature is entirely adequate.

Table 2.2-II shows an example of how propellant ballast is used to trim the vehicle to vary payload. In the two missions only the S-IC propellant loaded varies (directly with payload) and the quantity of unused S-IC propellant varies between missions.

TABLE 2.2-II CONTROLLING PAYLOAD BY PROPELLANT BALLAST
UNMANNED (F.S. = 1.25) MISSION

	POUNDS	
Payload	79,000	132,000
Launch Thrust	6,088,000	6,088,000
S-IVB Propellant	230,000	230,000
S-IC Propellant Loaded*	4,189,320	4,136,320
S-IC Propellant Burned	4,122,320	4,122,320
Propellant Ballast Discarded with S-IC	67,000	14,000

*Excluding Residuals

TABLE 2.2-I INT-20, SAT V CHARACTERISTICS COMPARISON

PARAMETERS	INT-20	SAT V DESIGN
LIFTOFF T/Wo	1.25	1.25
LOADS		
q MAX	729 LB/FT^2	766 LB/FT^2
g's @ (qα)	1.86	1.95
MAX. g's @ CUTOFF	4.68	4.68
CONTROL		
MAX. F-1 ENGINE DEFLECTION (β)	1.92 DEG.	3.5 DEG
HEATING		
BASE MAX. TEMP	1720^o F	1900^{o}F
S-IC FWD. SKIRT MAX. TEMP.	197^{o}F	167^{o}F

2.2 (Continued)

The overall INT-20 launch acoustic environment is lower than the Saturn V environment. But because the S-IVB and the Instrument Unit are relocated adjacent to the S-IC, see Figure 2.2-1, they experience slightly higher overall or integrated, sound pressure levels. The overall acoustic level for the S-IVB is about one decibel higher, although for specific, low frequency vibrations, the difference approaches about 5 db (Ref. 2×10^{-5} N/M^2). This difference in the low frequency range is sufficient to require requalification of a few S-IVB components. The Instrument Unit is not adversely affected by the increased sound pressure levels because of its structure's inherent damping characteristics.

2.3 DESIGN/ANALYSIS

2.3.1 S-IC Stage

S-IC stage design was governed by the requirement to accommodate an S-IVB second stage, the deletion of the center engine, a 2-2 F-1 engine cutoff sequence, and a propellant loading different from Saturn V. Stage adaptation is summarized on Figure 2.3.1-1. These changes are for a vehicle using a controlled-acceleration trajectory. Analyses showed that first acceleration peak may not exceed 3.68 g's (1.4 F.S. manned) to avoid hoop compression overload in the fuel tank lower bulkhead.

This limit can be raised to about 4.05 g's for unmanned (F.S. = 1.25) flights. A two-4.68 g peak (manned) acceleration history can be accommodated only if the upper part of the fuel tank aft bulkhead is strengthened. This would add about 300 lb. of weight to the bulkhead.

2.3.2 S-IVB Stage

The S-IVB design was based upon requirements for a new S-IC/S-IVB interface, higher structural loads, higher stage surface temperatures, and the elimination of the need for the retromotors in the S-IVB interstage. Also, since the baseline design mission was single-burn, direct ascent to orbit, stage design allowed for the removal of restart capability, with an option to simply add this capability if so desired. The removal of restart-related components reduces stage complexity and weight and reduces recurring costs. Figure 2.3.2-1 summarizes the major adaptations required to implement a baseline INT-20 stage. There will also be changes to the electrical and ordnance systems.

The current Saturn V insulation pattern is satisfactory for the aft interstage used with the INT-20 S-IVB, as shown in Figure 2.3.2-2.

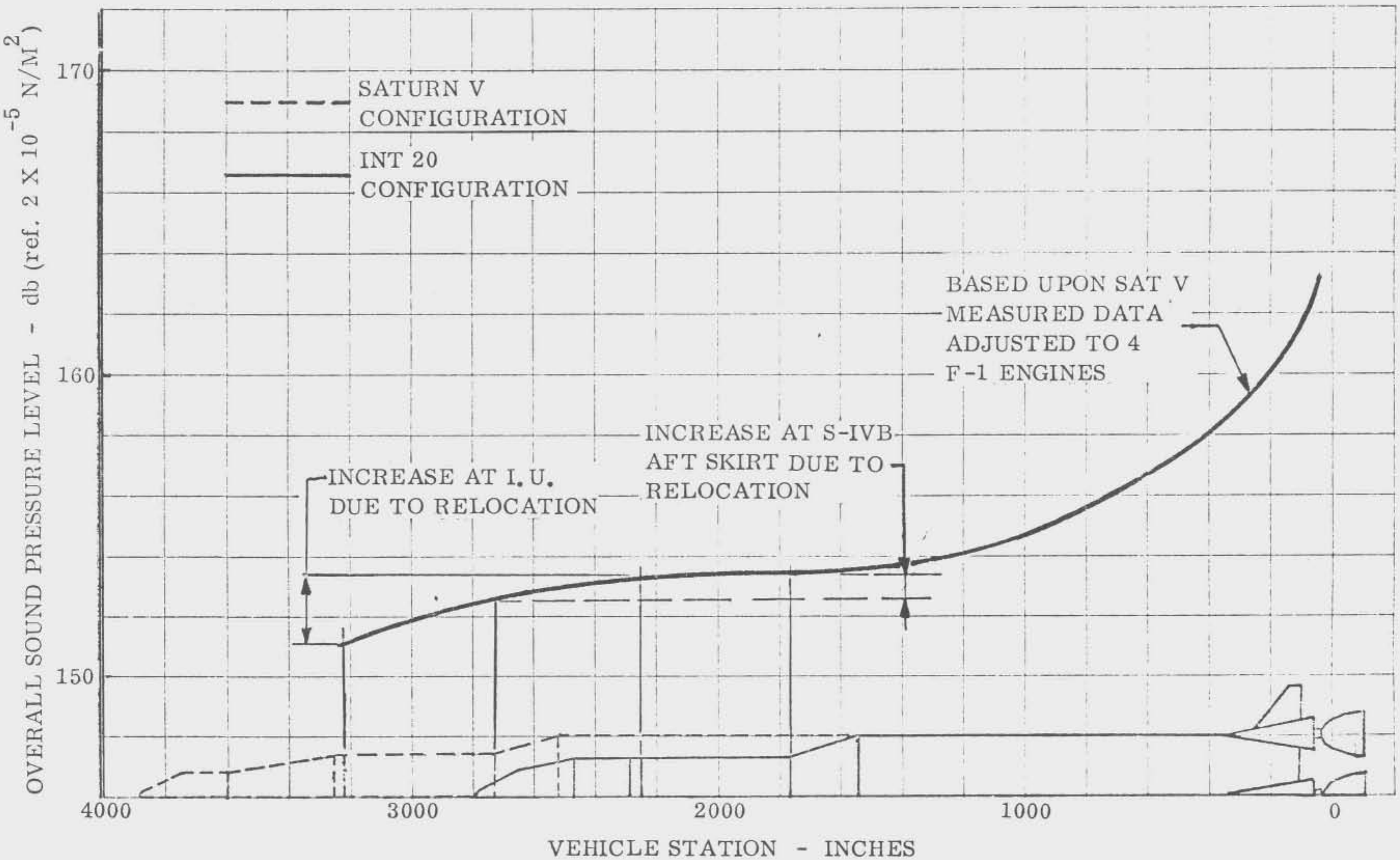


FIGURE 2.4-1 ACOUSTIC ENVIRONMENT - SAT-INT-20 WITH 4 F-1 ENGINES AND MLV NOSE

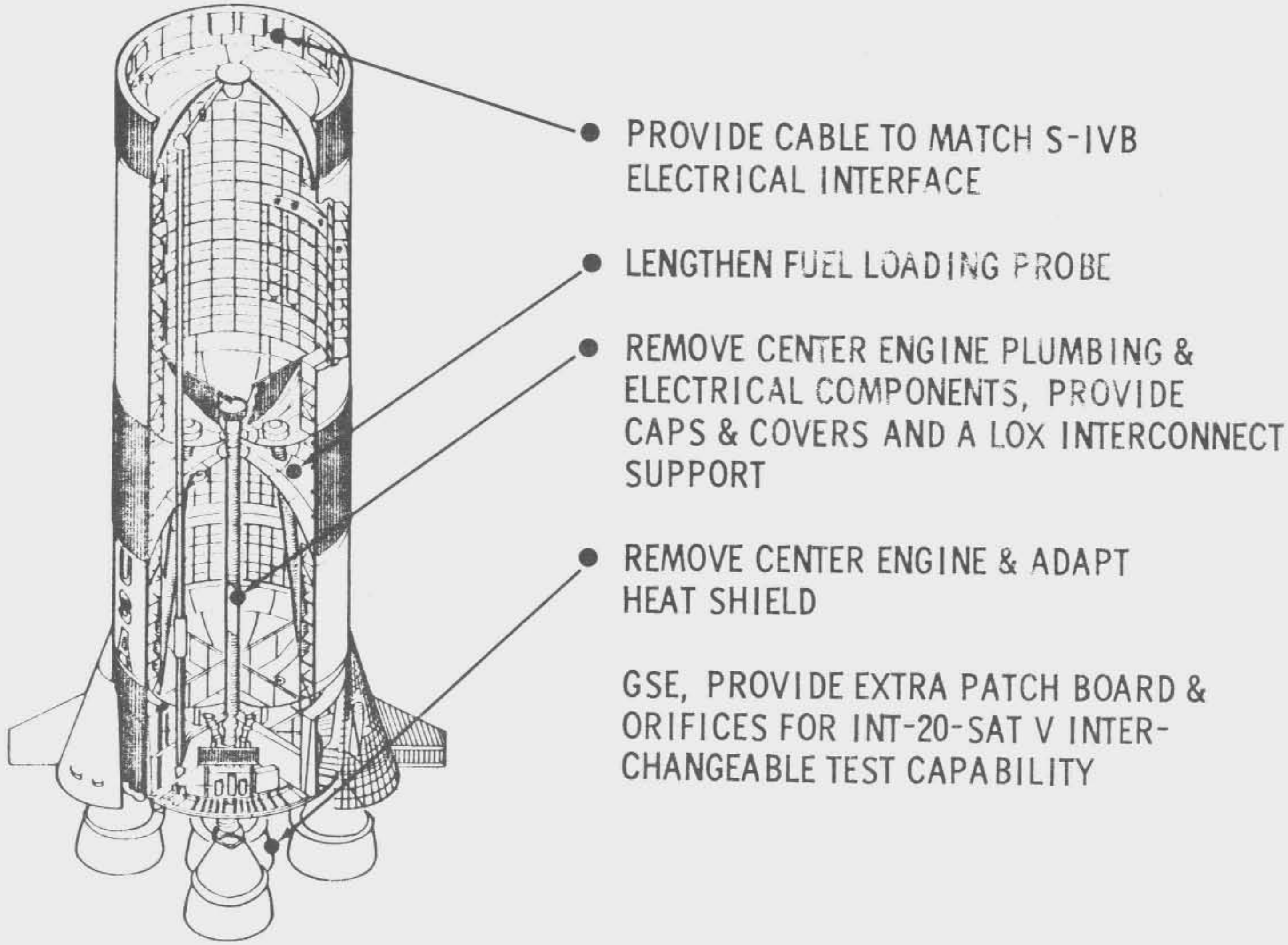
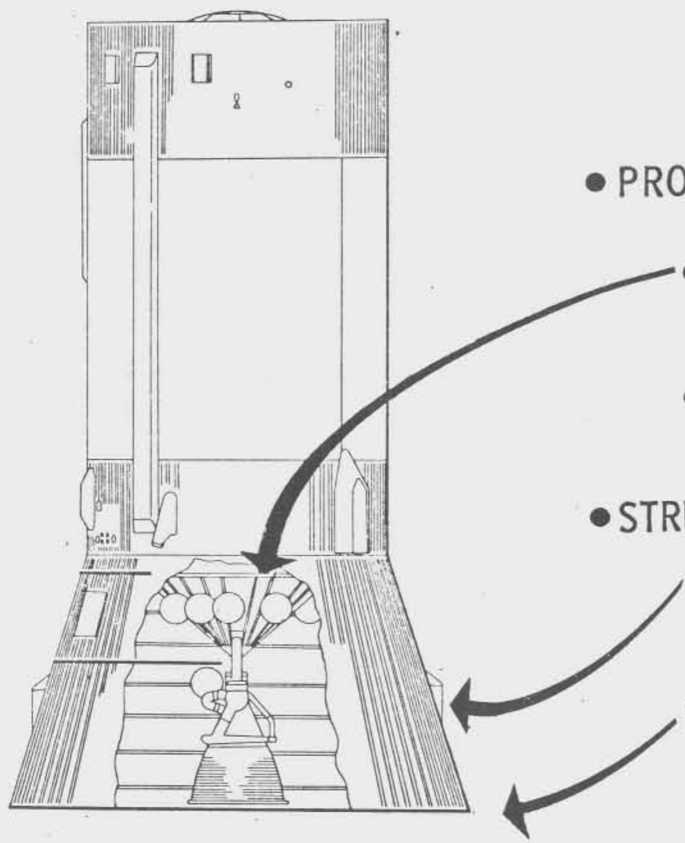


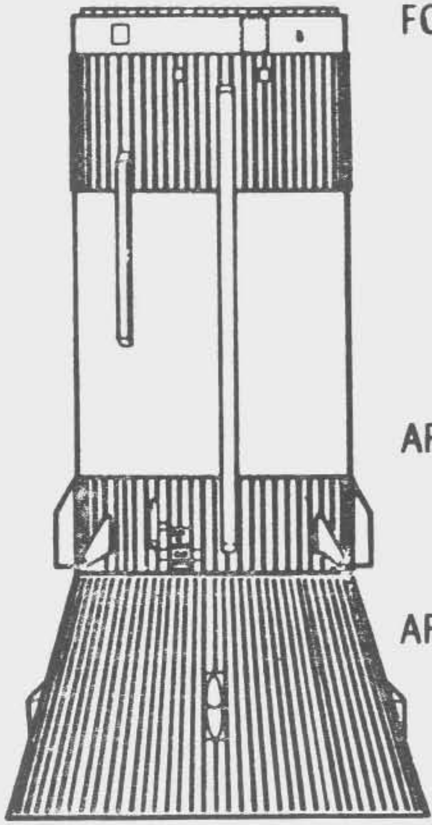
FIGURE 2.3.1-1 S-IC STAGE ADAPTATION



- PROPULSION & ELECTRICAL
  - REMOVE RESTART (*RETAIN ABILITY TO READILY REINSTALL)
  - REQUALIFY 10 COMPONENTS FOR HIGHER ACOUSTIC ENVIRONMENT
- STRUCTURE
  - REMOVE RETRO-ROCKET AND PROVISIONS
  - ADAPT S-IVB/S-II INTERFACE TO S-IC/S-IVB

*REQUIRED FOR SYNCHRONOUS MISSION

FIGURE 2.3.2-1 S-IVB ADAPTATION SUMMARY



	TEMPERATURE (°F) INT-20/S-IVB	SAT V/S-IVB
FORWARD SKIRT		
SKIN	417	389
STRINGERS	329	320
AFT SKIRT		
SKIN	277	258
STRINGERS	249	235
AFT INTERSTAGE		
SKIN*	319	330
STRINGERS*	274	320

*INSULATED WITH 0.01 IN. KOROTHERM SAME AS SAT V

FIGURE 2.3.2-2 INT-20/S-IVB SURFACE TEMPERATURES

2.3.2 (Continued)

The S-IVB aft interstage is currently an assembly of 4 panels having retromotor provisions, one panel having an access door, and three plain panels, as shown in Figure 2.3.2-3. For in-line INT-20 applications (new production), the interstage will be made up of seven plain panels and one access door panel. Retrofit interstages would only have retromotors and associated ordnance removed.

For both retrofit and in-line INT-20's, mating will be accomplished via a 5-inch interface adapter ring, shown in Figure 2.3.2-4. This mating concept was selected over various direct interface mating concepts because of lower program costs and the simplicity of direct mating.

2.3.3 Instrument Unit

The Instrument Unit adaptations, as summarized in Figure 2.3.3-1, result from the elimination of the S-II stage and the use of only 4 F-1 engines on S-IC. These changes have little effect on the I. U. The I. U. is mission-oriented and adaptation to an INT-20 configuration will be handled similar to the normal Saturn V mission-to-mission modifications. The software changes are summarized in Figure 2.3.3-2. Guidance and control and other systems changes are summarized in Figures 2.3.3-3 and 2.3.3-4, respectively. Changes to the guidance and control system include gain changes for the S-IC's F-1 engines and use of the existing Saturn V abort-to-orbit program to effect INT-20 orbital flight.

Since the baseline vehicle uses an MLV payload shape, which does not use the launch escape system, the Q-Ball (angle-of-attack meter) wiring will be left spare.

2.3.4 Performance and Applications

The performance capabilities of the recommended manned and unmanned INT-20 configurations are summarized in Table 2.3.4-I.

The injected lunar payload of the two-stage INT-20, 32,000 lb., could be used with a lunar module derivative to deliver 5,000 lb. to the lunar surface.

The INT-20 can be used to support space station crew rotation, using either Apollo or Big Gemini spacecraft and associated logistics or experiment packages. The mission profile required is an injection of the payload into an elliptic orbit, with 80 to 100 NM (185.3 km) or lower perigee (injection) altitude dictated by the Apollo/Big Gemini abort re-entry angle limitations. Figure 2.3.4-1 shows elliptic orbit payloads achieved by INT-20, and demonstrates the net payload resulting in a 270 NM circular orbit if the Service Module or Propulsion Module were used to circularize the orbit of the Apollo Command Module or Big Gemini, respectively.

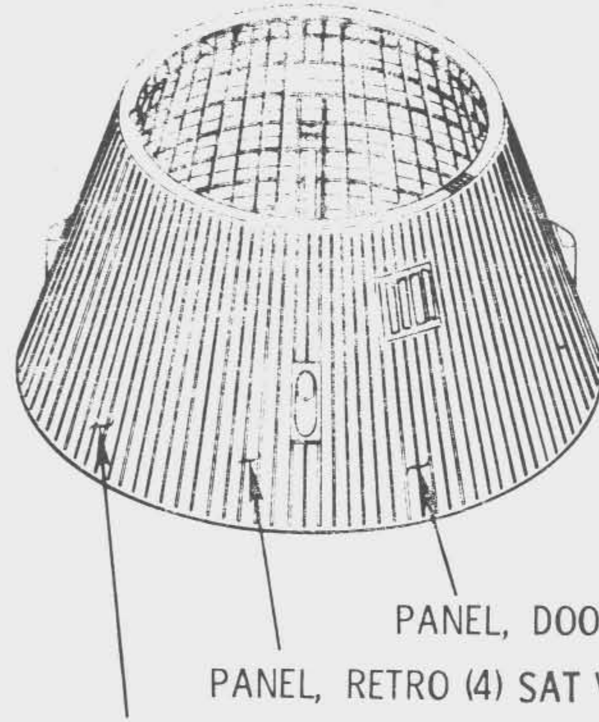


FIGURE 2.3.2-3 INTERSTAGE CONFIGURATION

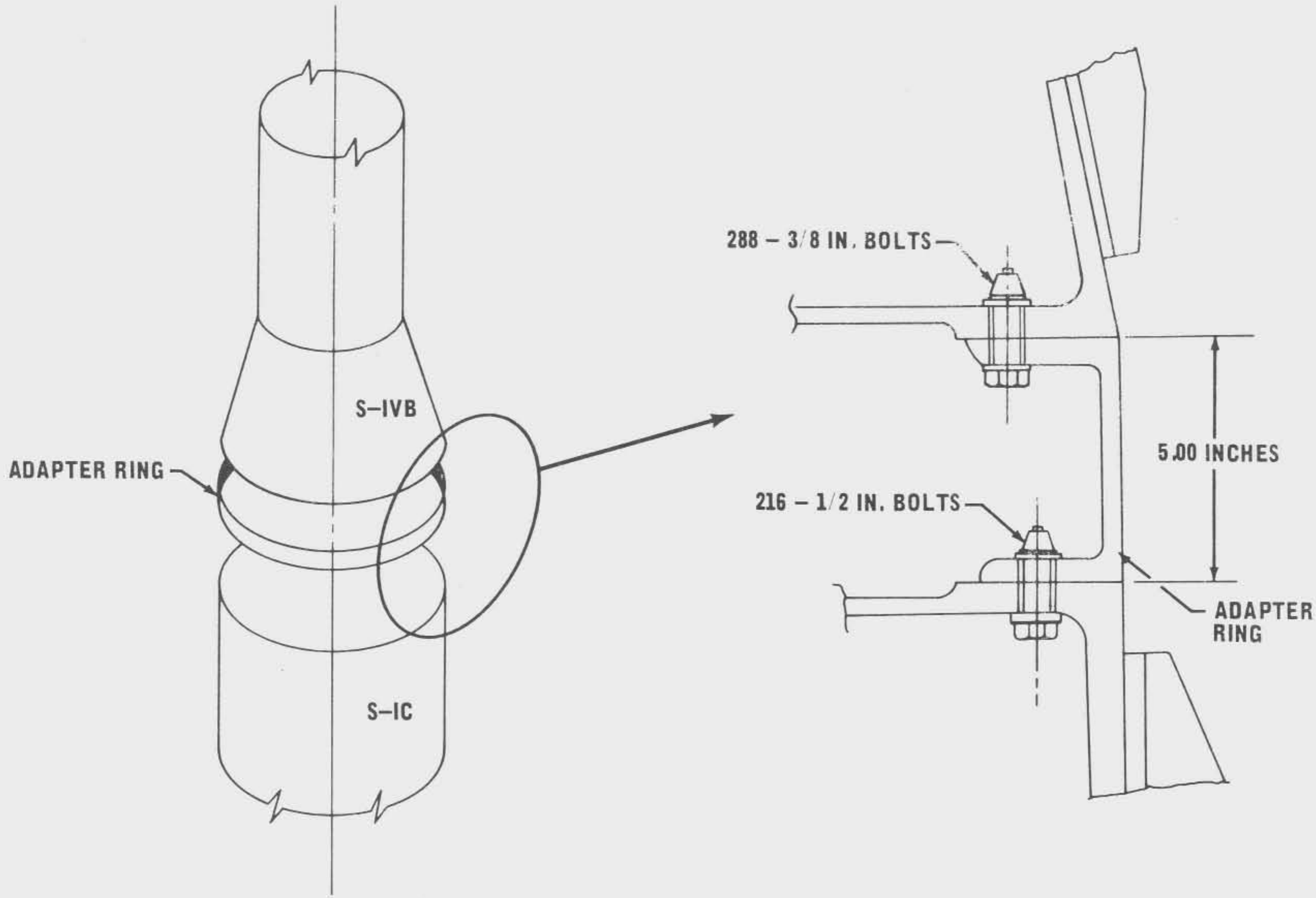
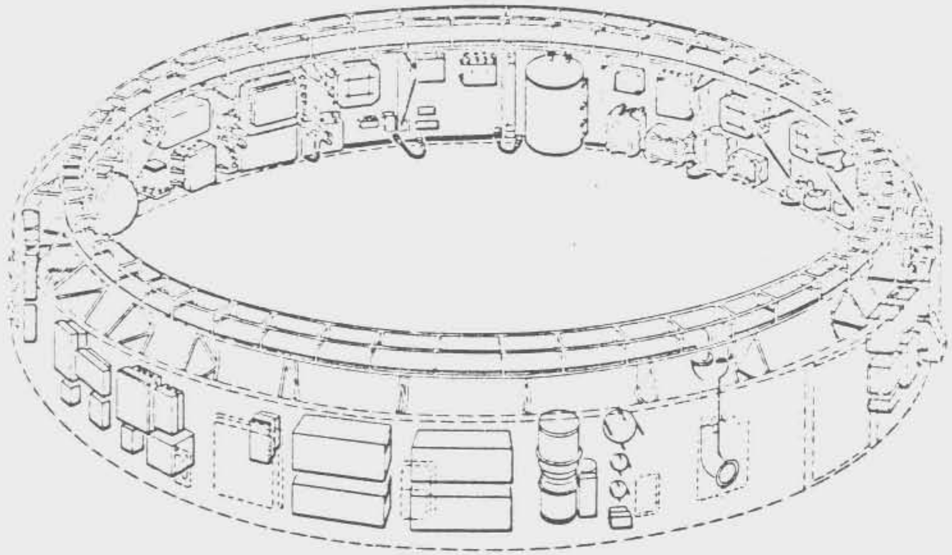


FIGURE 2.3.2-4 S-IC/S-IVB INTERFACE



THE SAT V (500 SERIES) I. U. IS VERSATILE AND, THEREFORE,

- ADAPTABLE TO INT-20 BY MINOR HARDWARE & SOFTWARE CHANGES
- READILY ADAPTABLE TO BIG G FROM CURRENT CSM CAPABILITY
- REVERSIBLE, I. E., INT-20 TO SATURN V*

*S-II CAPABILITY RETAINED

FIGURE 2.3.3-1 INSTRUMENT UNIT (I.U.)

SOFTWARE MODS

ALL PROGRAM FUNCTIONS REQUIRE ONLY DATA CHANGE EXCEPT FOR THE FOLLOWING:

GROUND RETARGETING	DELETE
VARIABLE LAUNCH AZIMUTH	DELETE
S-IVB CUTOFF	NO CHANGE
S-IVB RESTART	DELETE
TIME BASES	MINOR LOGIC CHANGES
DISCRETES	MINOR LOGIC CHANGES

FIGURE 2.3.3-2 ADAPTING THE FLIGHT PROGRAM FOR INT-20

- MINOR MODIFICATION OF THE FLIGHT CONTROL COMPUTER TO ACTIVATE EXISTING SPARE SWITCH POINTS FOR S-IC BURN
- IMPLEMENTATION OF S-IC TWO ENGINE SHUT DOWN AS A FUNCTION OF ACCELERATION
- SELECTIVE S-IC ENGINE SHUT DOWN BASED ON MALFUNCTION LOGIC
- "ABORT TO ORBIT" PROGRAM IMPLEMENTED

FIGURE 2.3.3-3 ADAPTING THE GUIDANCE AND CONTROL SYSTEM

ELECTRICAL

- SLIGHT REDUCTION IN 28V POWER FROM BATTERIES
- S-II WIRING WILL BE LEFT SPARE

INSTRUMENTATION AND COMMUNICATION

- Q-BALL AND S-II MEASUREMENTS NOT REQUIRED
- EXCESS MEASUREMENT HARDWARE LEFT SPARE

ENVIRONMENTAL CONTROL

- NO CHANGE

STRUCTURE

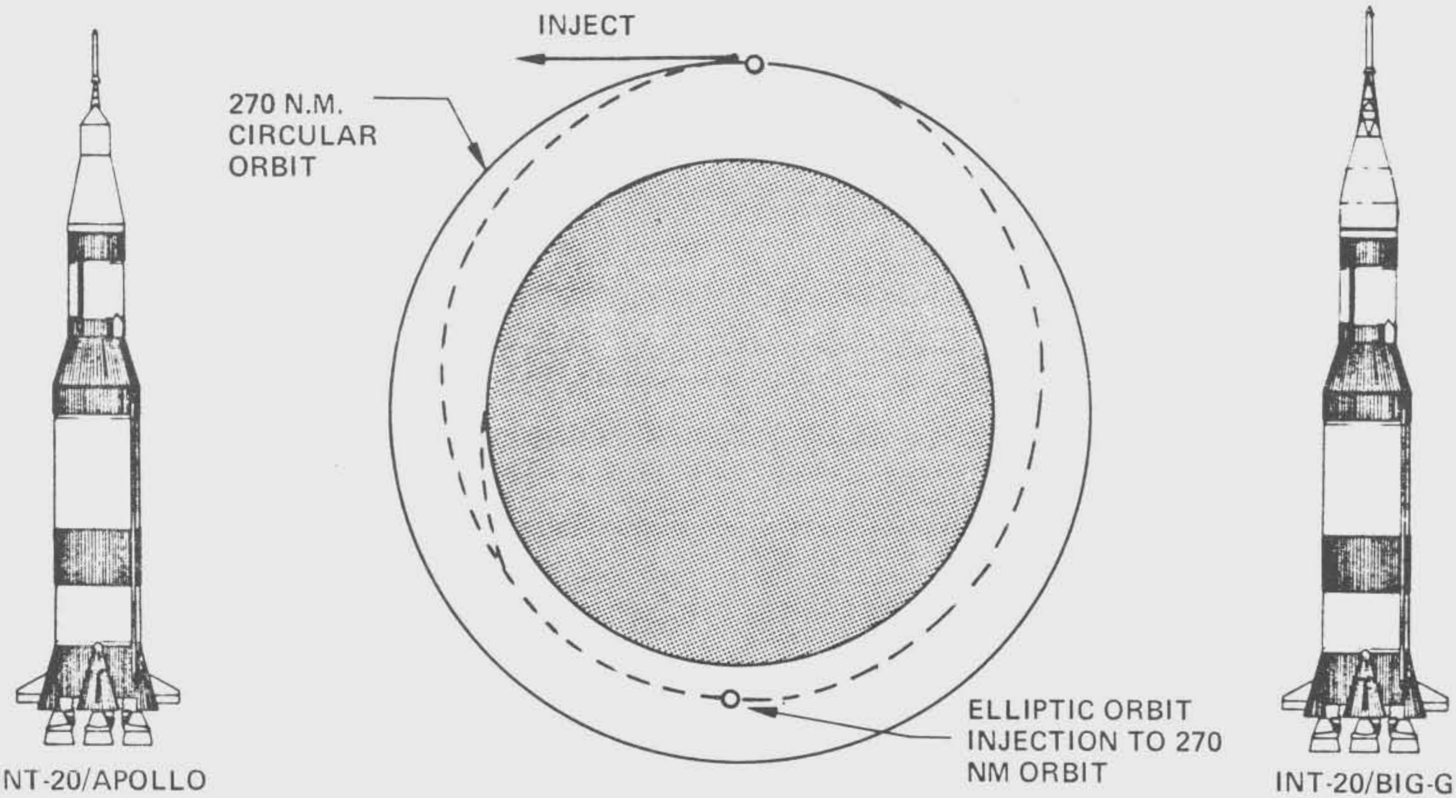
- NO CHANGE

FIGURE 2.3.3-4 ADAPTING OTHER I.U. SYSTEMS

TABLE 2.3.4-I RECOMMENDED SYSTEM PAYLOAD CAPABILITY SUMMARY

MANNED (F.S. = 1.4)	
100 N MI, 28.5° INCL, CIRC ORBIT	126,000 LB
100 x 270 NMI, 50° INCL, ELIP. - ORBIT	112,000 LB
200 N MI POLAR ORBIT (KSC, YAW STEERING)	82,000 LB
UNMANNED (F.S. = 1.25)	
EQUATORIAL SYNCHRONOUS (C_3 = 25)	14,000 LB.*
EQ SYNC WITH CENTAUR (C_3 = 25)	29,000 LB.*
EQ SYNC WITH SM (C_3 = 25)	21,000 LB.*
LUNAR (TLI, 72 HRS)	32,000 LB.
MARS (C_3=20)	18,000 LB.
JUPITER FLY BY WITH CENTAUR (C_3 = 100)	11,000 LB.

*I.U & S–IVB REQ. MISSION DEPENDENT ADAPTATION



INJECTION ALTITUDE (N.M.)	ELLIPTIC ORBIT PAYLOAD - (LBS)	270 N. M CIRCULAR ORBIT PAYLOAD * (LBS)
80	112,000	108,200
100	111,300	107,800

* BASED ON ISP = 300, $\lambda' = .85$ INJECTION PROPULSION

FIGURE 2.3.4-1 INT-20 ELLIPTICAL ORBIT PAYLOAD CAPABILITY

2.3.4 (Continued)

The INT-20 payload capability enhancement available through the use of injection stages is demonstrated in Figure 2.3.4-2.

2.4 DEVELOPMENT PROGRAM PLAN

A Development Program Plan was prepared to show the design, development, test, and evaluation (DDT&E) requirements for implementation of an INT-20 vehicle program.

2.4.1 Design Requirements

Design requirements include adaptations for new interfaces and the two-stage mission and new Systems Engineering and Integration (SE&I) functions for the vehicle. The SE&I tasks include flight prediction and analysis computer program development and vehicle interface documentation preparation.

2.4.2 Testing

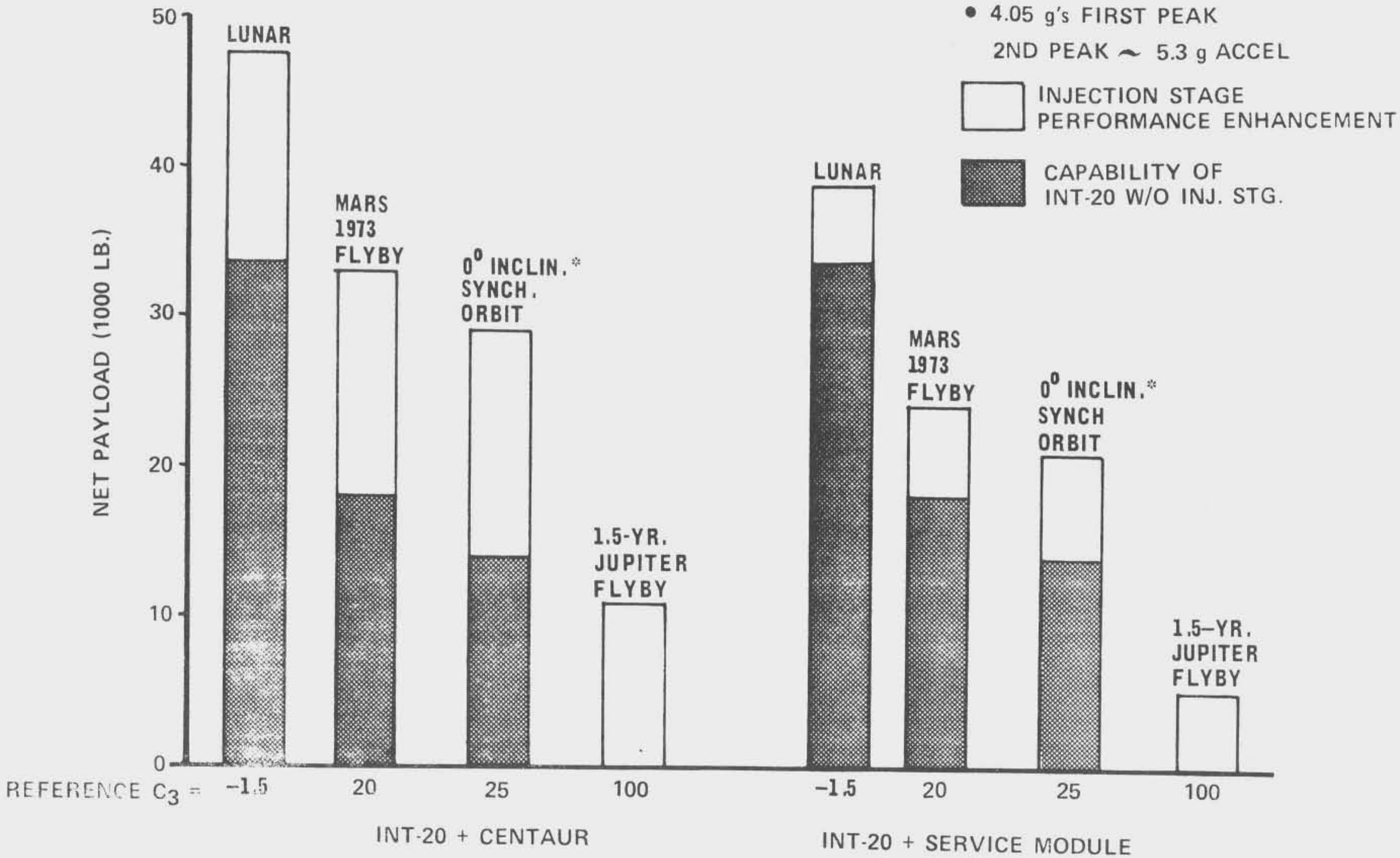
The test program required to implement the INT-20 is small. Tests are required to qualify the F-1 engine for long-duration (up to 240 seconds) operation. Wind tunnel force and pressure model tests are needed to verify analytical aerodynamic data on the INT-20 with various payload shapes.

INT-20 dynamic characteristics will be derived by analysis and correlation of Saturn V flight data, Saturn V dynamic test data, and flexure model tests. New payload configurations that differ from Apollo in dynamic characteristics may need "short stack" dynamic testing. The short stack is a combination of the S-IVB stage, the I. U., and the payload.

The first INT-20 vehicle flown should perform a useful mission since the first INT-20 has a calculated reliability equivalent to the first manned Saturn V. Interface and separation variations between Sat V and INT-20 lead to a Boeing Company recommendation for the first INT-20 to be flown unmanned. This unmanned but useful mission is recommended to increase management confidence prior to manned use of INT-20.

2.4.3 Manufacturing

Manufacturing plans for INT-20 stages are the same as for Saturn V stages. INT-20 stages will be interspersed with Saturn V stages during production. New tooling will be required for the interface adapter ring.



*I.U. & S–IVB REQ. MISSION DEPENDENT ADAPTATION

FIGURE 2.3.4- 2 INT-20 WITH INJECTION STAGES

2.4.4 Schedule

The Development and Delivery Plan shown in Figure 2.4.4-1 is applicable to all in-line procurement programs of this study. The S-IC is the pacing item for the INT-20 with 36 months from ATP (Authority to Proceed) to on-dock KSC. The flow times shown are all the same as current requirements for Sat V components procurement except for KSC and SE&I activities. The KSC bar shows the 15 months needed to activate the convertible MSS. The SE&I bar shows a six-month period for development of pre- and post-flight analysis programs for the INT-20 two-stage mission plus the normal twelve month SE&I cycle.

2.4.5 Development Cost

Development costs of $7.49M as shown in Table 2.4.5-I will buy:

a. A new data base and recoded drawings for the four-engine S-IC stage.

b. Requalifying a few S-IVB components to their new acoustic environment near the S-IC stage.

c. Reprogramming the Instrument Unit for the INT-20.

d. Requalifying F-1 engines for longer firing duration.

e. Reprogramming SE&I flight analysis computers for the INT-20 mission.

f. Modifying KSC launch facilities to accommodate the shorter two-stage INT-20 vehicle.

2.4.6 Retrofit Plan

An INT-20 vehicle can be adapted from Saturn V S-IC and S-IVB stages and the Instrument Unit that have been retrieved from storage. These would be modified to the INT-20 configuration in much the same manner as for an in-line vehicle. The retrofit S-IC would differ from the in-line stage in that the existing fuel tank loading probe would be used and the LOX tank standpipe would be retained. The retrofit S-IVB differs only in that the aft interstage will retain retrorocket provisions, although the retromotors and ordnance would not be installed. The retrofit Instrument Unit would be the same as the in-line unit.

Testing for the retrofit INT-20 is the same as for the in-line INT-20.

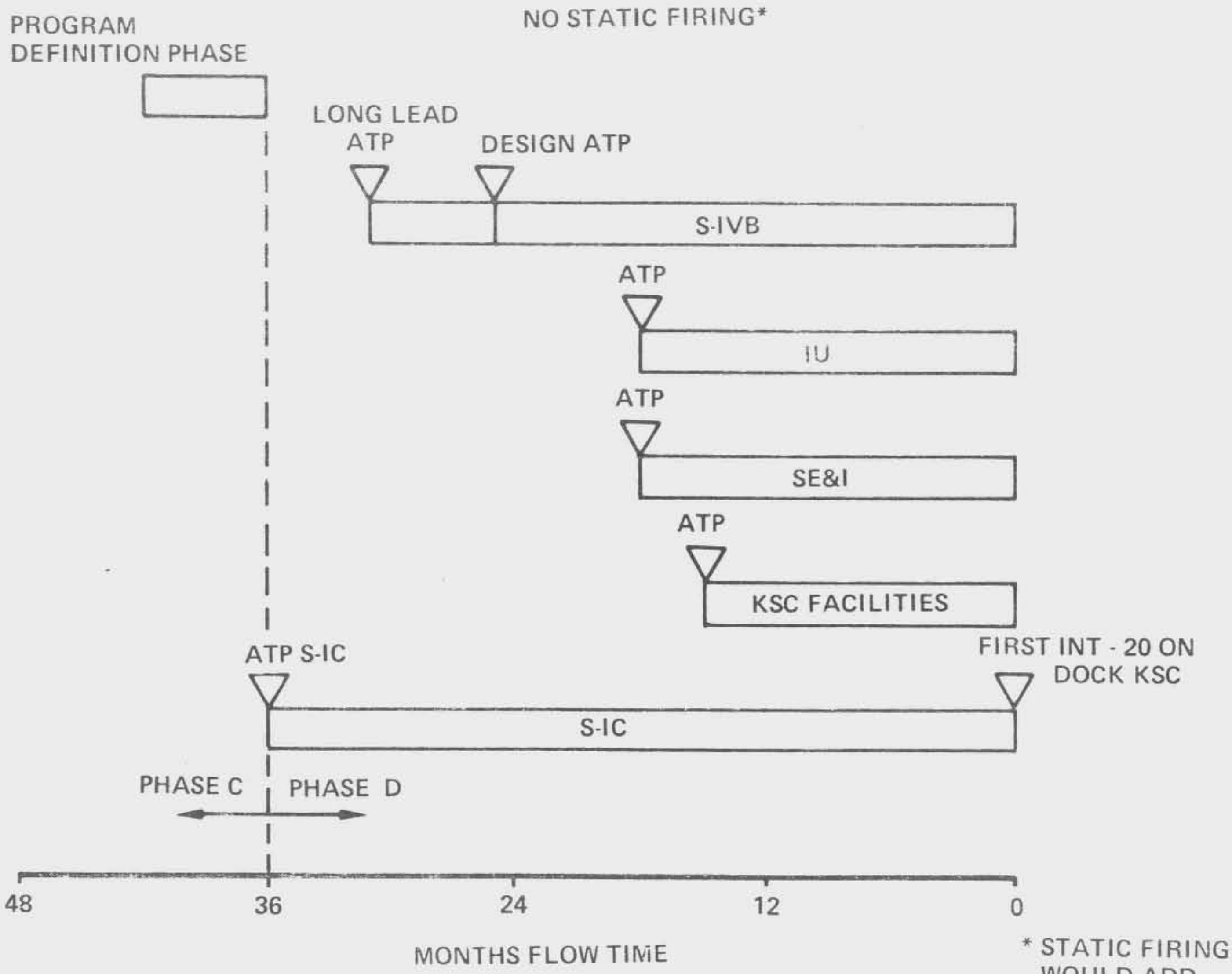


FIGURE 2.4.4-1 INT-20 DEVELOPMENT AND DELIVERY SCHEDULE

TABLE 2.4.5-I INT-20 DEVELOPMENT COST

1968 DOLLARS IN MILLIONS

S-IC	1.00
S-IVB	2.94
I. U.	.01 (ABOVE 5 UNITS/YR ADD 0.7)
F-1 ENGINE	.23
J-2 ENGINE	NONE
SE&I	.11
TOTAL	4.29
KSC FACILITIES	3.20 (10 TO 11 PER YR, BIG G TYPE PAYLOAD ADD 1.6 ROM)
TOTAL	$7.49M

2.4.6 (Continued)

The retrofit schedule (Figure 2.4.6-1) shows 18 months from ATP (Authority to Proceed) to on-dock KSC. The schedule is paced by the need for new two-stage pre- and post-flight evaluation programs (SE&I) and procurement of S-IC heat shield panels.

Investment costs to procure the first and second retrofit INT-20 are shown on Table 2.4.6-I. These costs are over and above the normal Sat V costs incurred to retrieve a vehicle from storage, check it out, and ship to KSC.

TABLE 2.4.6-I RETROFIT COST

1968 DOLLARS IN MILLIONS

	FIRST VEHICLE	SECOND VEHICLE
S-IC	$1.11	$.20
S-IVB	2.93	(.02)
I.U.	.01	.01
F-1 Engine	.23	-
J-2 Engine	-	-
SE&I	.11	-
Sub-Total	4.39	.19
KSC Facilities	3.20	.20
Total	$7.59M	$0.39M

* F-1 Engine and other deleted S-IC hardware cost saving ($3M) not included.

( ) Cost Saving

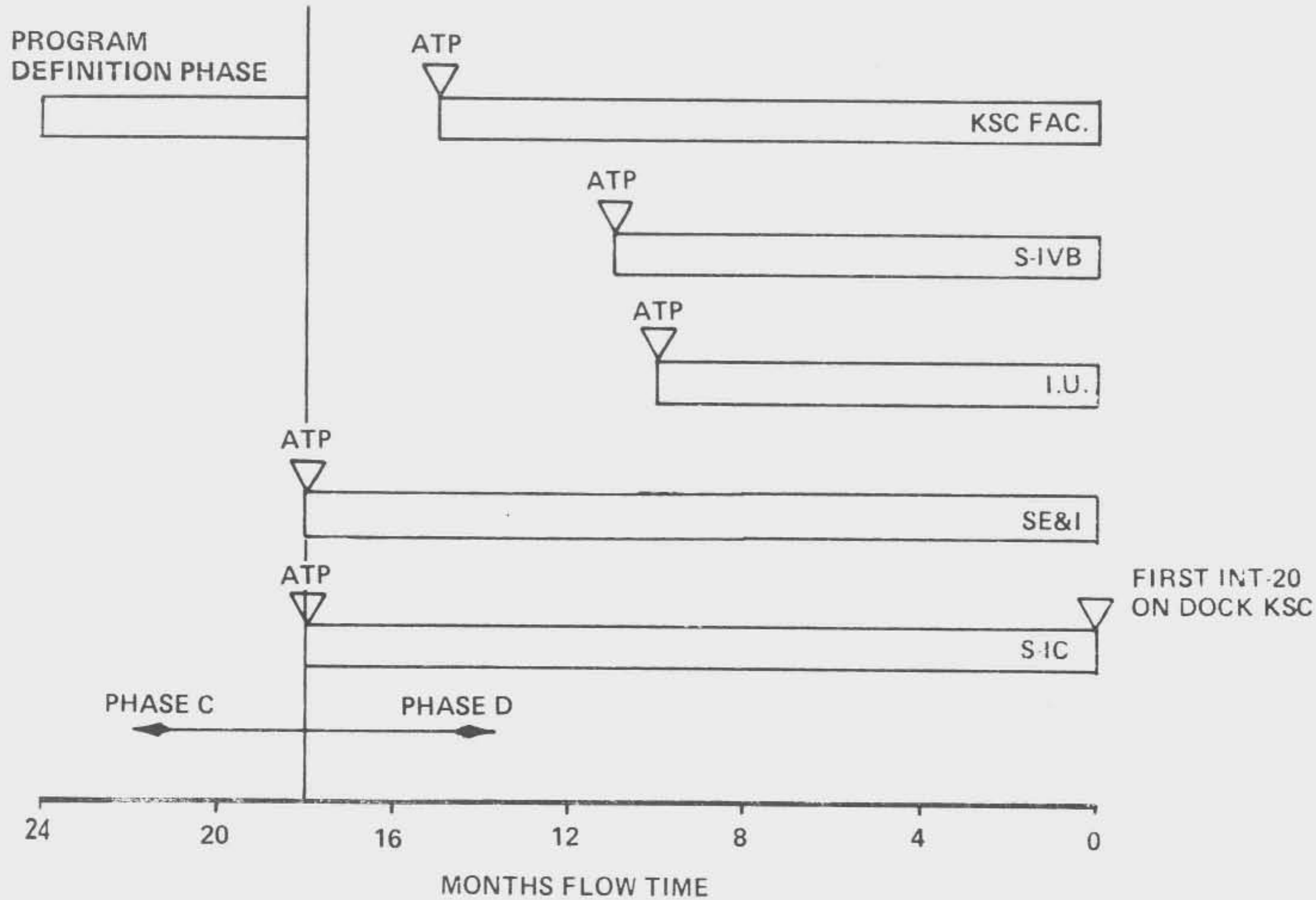


FIGURE 2.4.6-1 RETROFIT SCHEDULE

SECTION 3

PHASE I TRADE STUDIES

3.0 GENERAL

Several INT-20 candidate configurations were compared on the basis of technical feasibility, flight performance, and cost. These comparisons resulted in the selection of the 4 F-1, 4.68-g maximum acceleration vehicle as the study baseline. The alternatives considered and evaluated during the Phase I trade study (see Figure 3.0-1) included:

a. S-IC stage with 2, 3, 4 and 5 F-1 engines;

b. S-IVB stage 200 series (used on the Saturn IB) and 500 series (Saturn V) configurations;

c. Instrument Unit 200 and 500 series configurations;

d. Maximum axial acceleration during first stage operation of from 4.68 to 6.0 g's.

The performance enhancement available from Centaur or Service Module Injection stages was also determined, but was not a consideration in baseline configuration selection.

3.1 STAGE ANALYSIS

Each stage and the Instrument Unit was analyzed to ascertain its capabilities and limitations in INT-20 applications. Development and production cost estimates were derived from these findings.

3.1.1 S-IC Stage

Data were developed to show the effects of, and requirements resulting from, varying both the number of F-1 engines on the stage and the maximum axial acceleration experienced by the stage. The trades analyses revealed no major modification requirements for the S-IC stage.

3.1.1.1 F-1 Engine Deletion

Data developed during the 1966 INT-20 study (Ref. 3.1.1.1-1) described the requirements for deleting engines from the S-IC stage. These deletion requirements were used to develop stage weights and costs for the trades analyses.

TRADE STUDY ALTERNATIVES

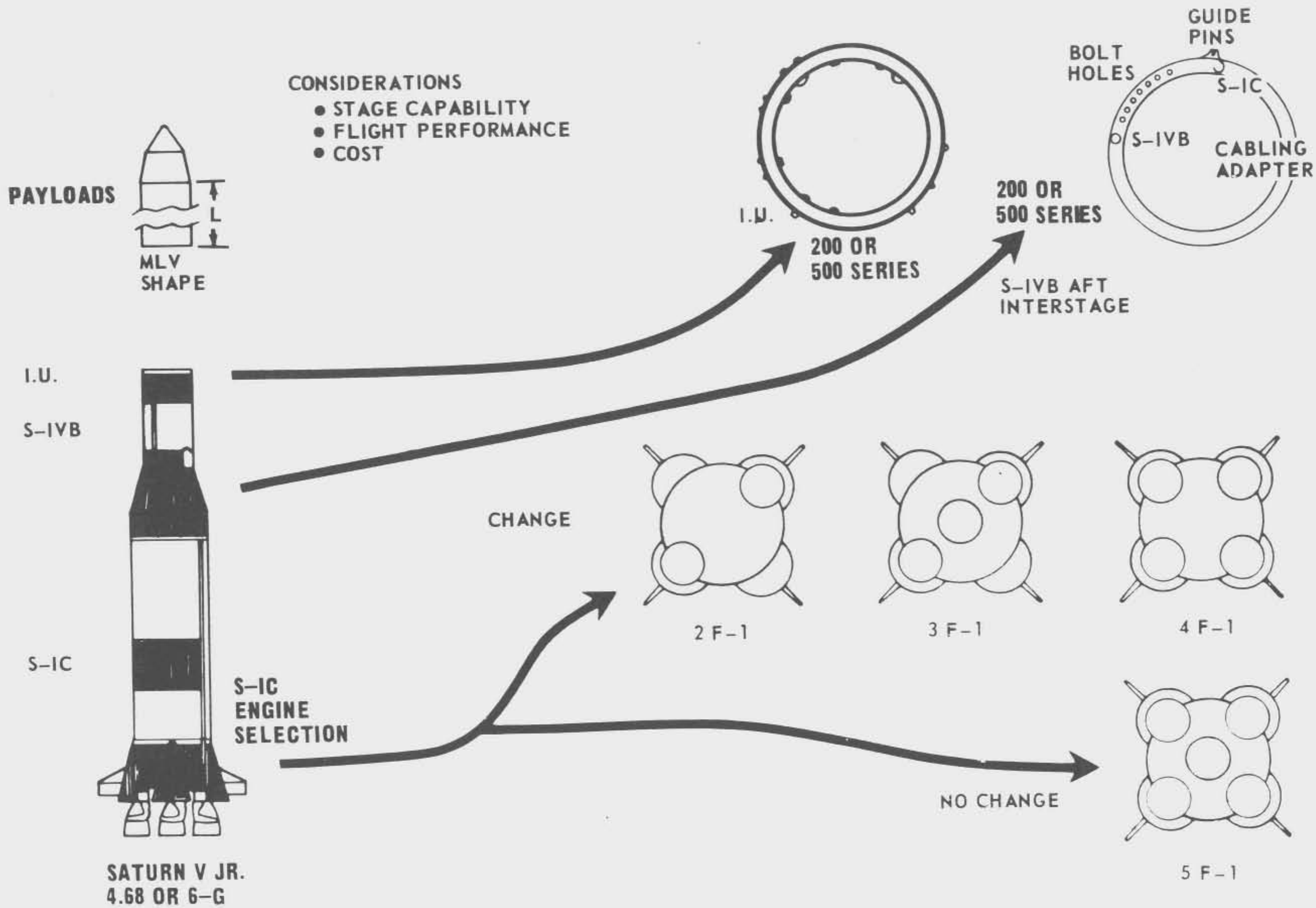


FIGURE 3.0-1 TRADE STUDY ALTERNATIVES

3.1.1.2 Stage Loads

a. Cylindrical Structure Loads

The S-IC cylindrical structure (unpressurized structure plus tank sidewalls) was analyzed at various times of flight to determine structural adequacy for INT-20 applications. The analyses showed that the S-IC cylindrical structure was adequate, using a factor of safety of 1.4, for those vehicle/payload combinations studied.

Figures 3.1.1.2-1 and -2 show combined compressive (N_c) loads for the 4 and 5 F-1 S-IC stages, respectively. The 4 F-1 stage data shown are for maximum acceleration (6-g) time of flight with a 138,000 lb. payload. The 5 F-1 stage data are for a 147,356 lb. payload and an acceleration of 3.66 g's (3-engine cutoff at t = 102.5 sec).

Analysis was limited to the 4 and 5-engine stages. Axial loads on the 2 and 3-engine stages would be less since payloads are much smaller. Bending moments would be smaller because the payload envelopes would be smaller for corresponding payload densities and flight trajectories. It was concluded that combined applied loads would be less on the 2 and 3 F-1 stages for the mission considered.

b. Propellant Tank Bulkhead Loads

Cursory loads analyses of the S-IC propellant tank bulkheads were made. These analyses revealed potential loads problems (for a factor of safety of 1.4) in the lower bulkheads of both 5 F-1 stage tanks, the 3 F-1 stage RP-1 tank, and the 4 F-1 stage RP-1 tank. These analyses also showed that it was possible to reestablish a 1.4 factor of safety in each bulkhead without structural modification. This could be done by reducing tank ullage pressure, decreasing F-1 engine thrust, or use of load-alleviating trajectories (or some combination of these).

Figures 3.1.1.2-3, 3.1.1.2-4, and 3.1.1.2-5 show the S-IC propellant tank limit operating pressure envelopes for the 3 F-1, 4 F-1, and 5 F-1 stages, respectively. The limit operating pressure envelope shows the maximum pressure experienced at various tank stations. The data shown are for 6-g trajectories. On the 3 F-1 stage, the factor of safety in the RP-1 tank bulkhead was less than 1.4 below Station 230. The 4 F-1 stage RP-1 tank bulkhead became critical below Station 270. The 5 F-1 stage became critical in both the RP-1 and LOX tanks. The RP-1 tank was critical below Station 360 and the LOX tank became critical below Station 815. Note that early AS-504 bulkhead capability data were used in these analyses, and that the Phase II data used generally showed higher capability.

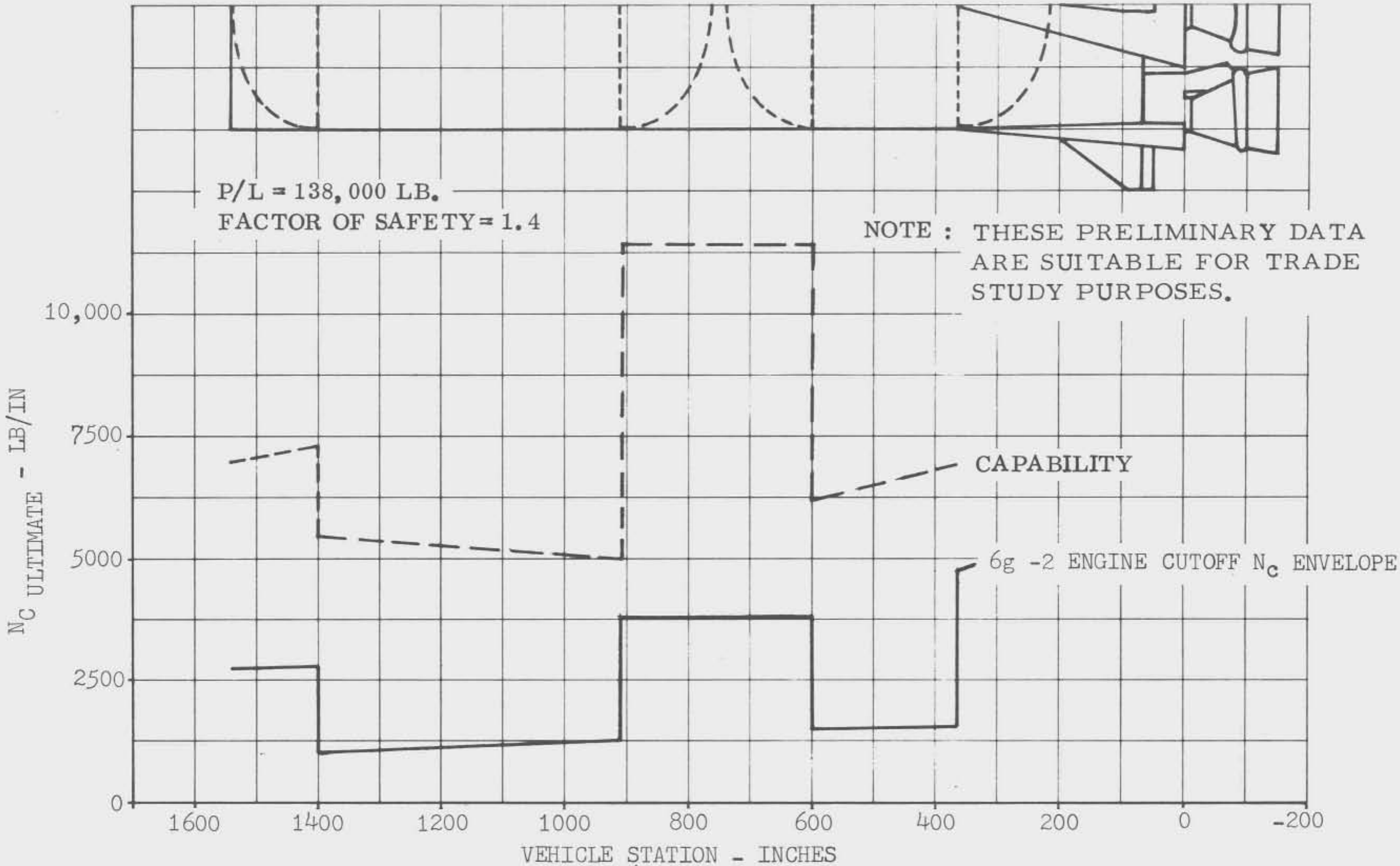


FIGURE 3.1.1.2-1 ULTIMATE COMBINED COMPRESSIVE LOADS FOR 4 F-1 S-IC

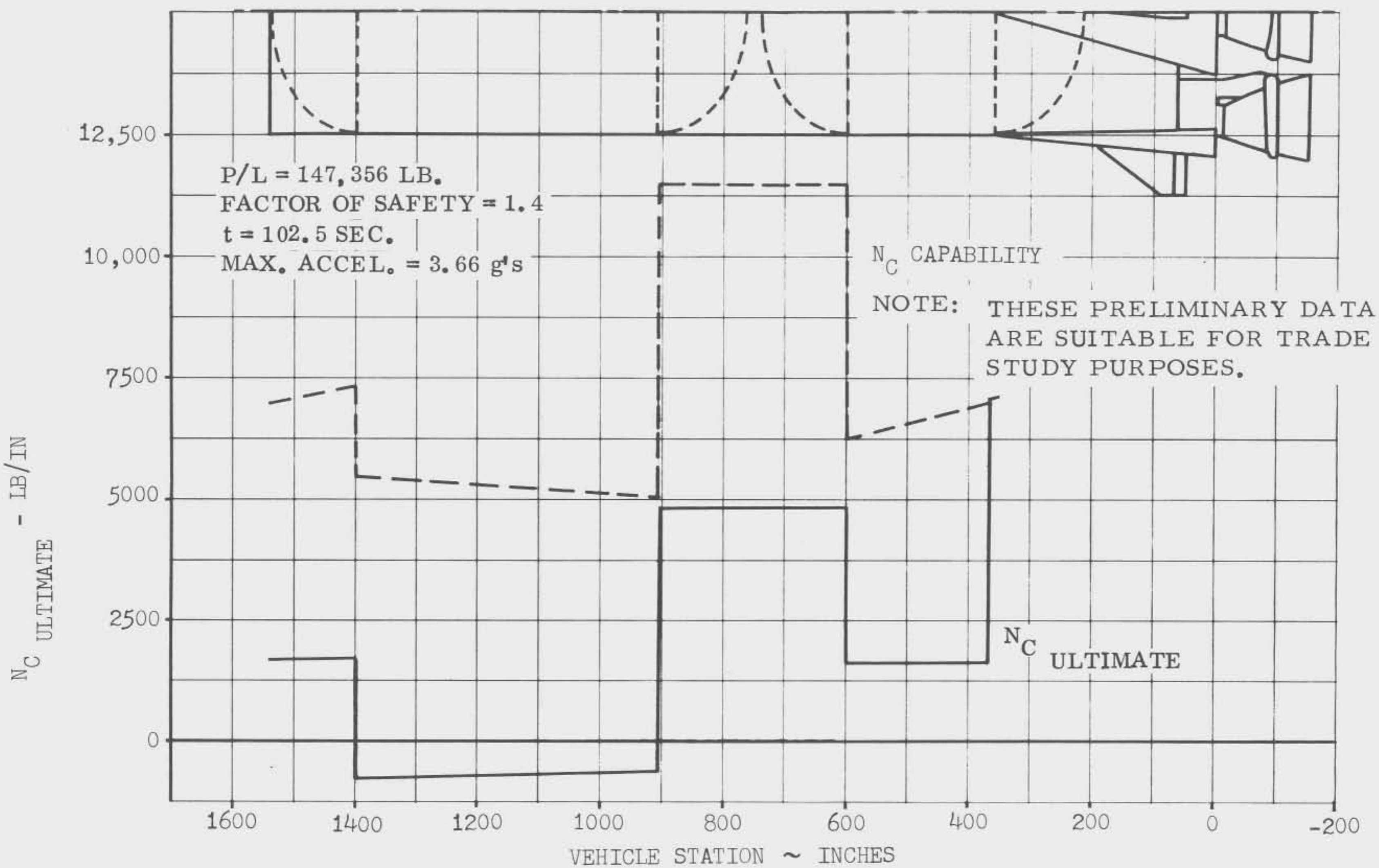


FIGURE 3.1.1.2-2 4 F-1 S-IC COMBINED COMPRESSIVE LOADS

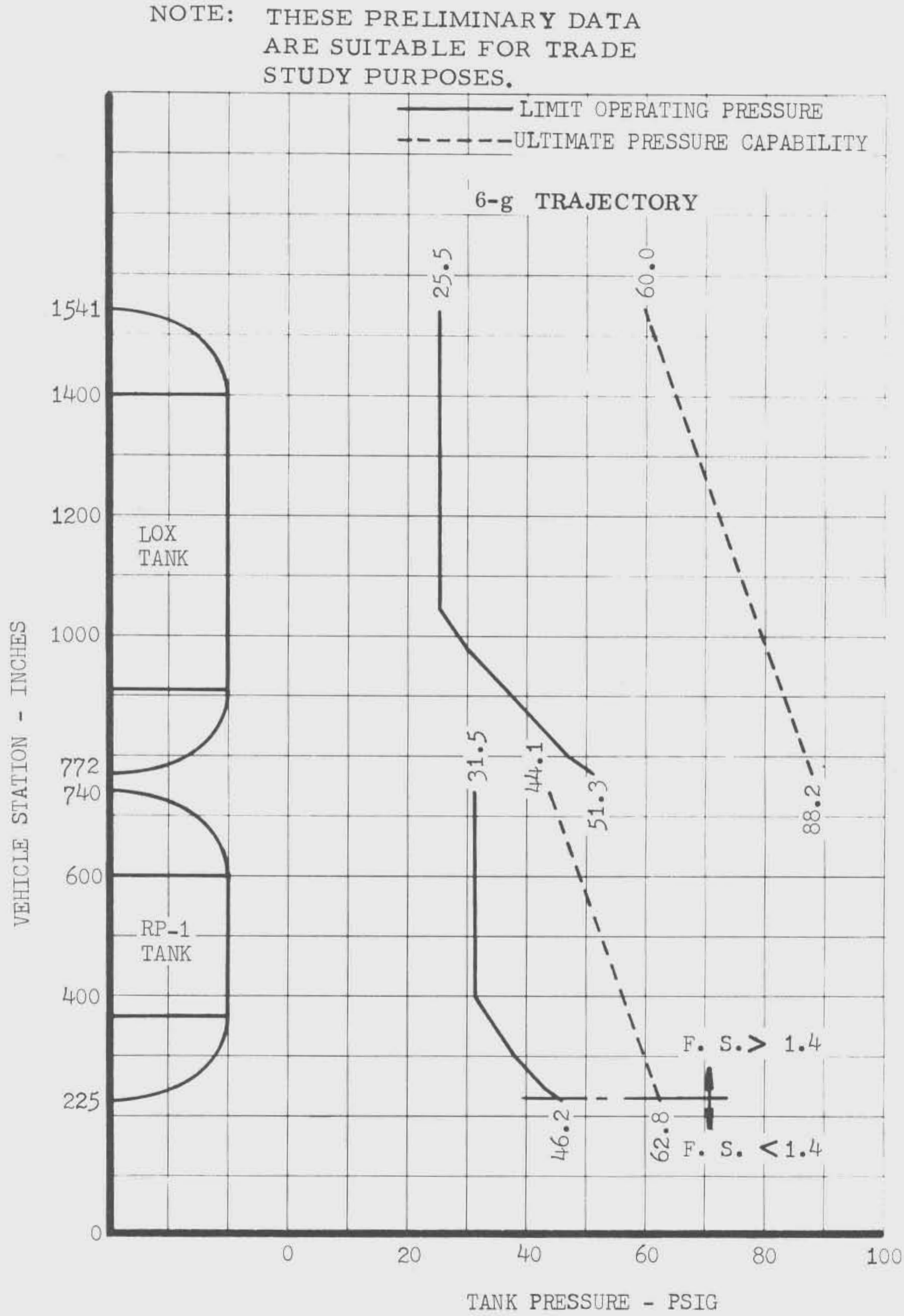


FIGURE 3.1.1.2-3 3 F-1 ENGINE VEHICLE S-IC TANK PRESSURES

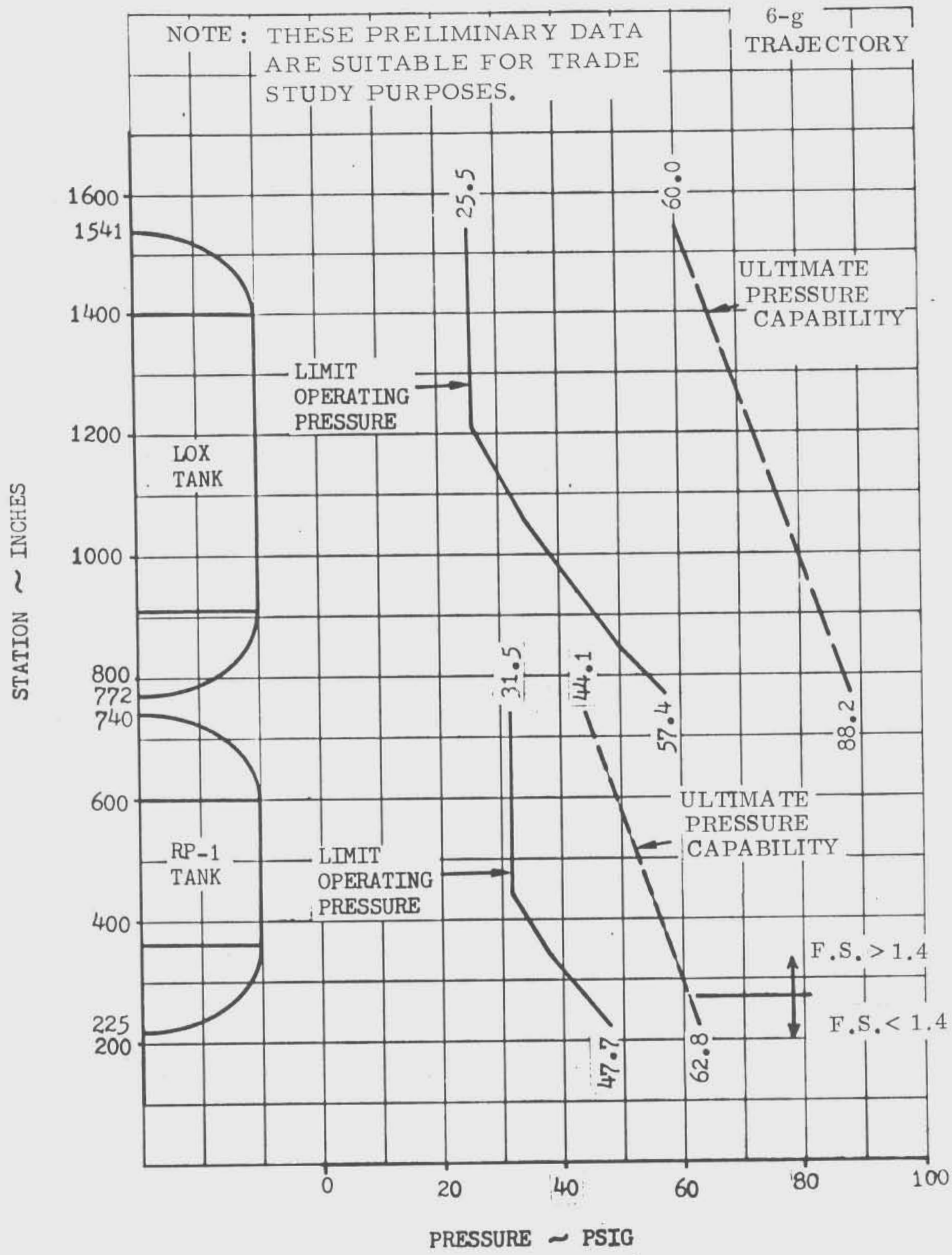


FIGURE 3.1.1.2-4 4 F-1 ENGINE VEHICLE S-IC TANK PRESSURES

NOTE: THESE PRELIMINARY DATA ARE SUITABLE FOR TRADE STUDY PURPOSES.

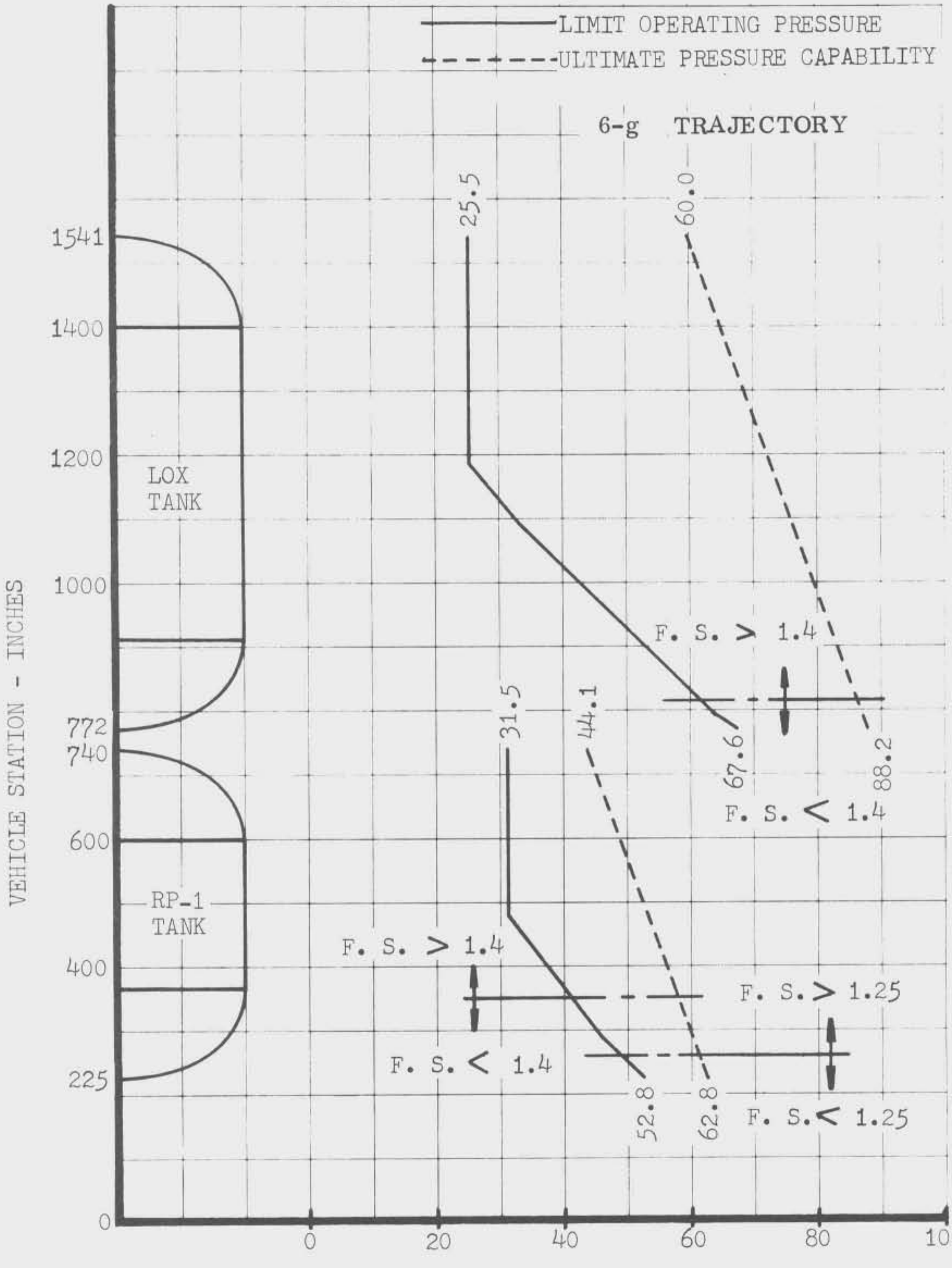


FIGURE 3.1.1.2-5 5 F-1 ENGINE VEHICLE S-IC TANK PRESSURES

3.1.1.2 (Continued)

Basically, the higher bulkhead loads on INT-20 stages occur because of a greater acceleration head contribution. (Ullage pressure schedules for this analysis were assumed to be the same as for Saturn V). The INT-20 thrust-to-weight at launch is higher than for Saturn V (1.25 versus 1.18), so the INT-20 experiences higher accelerations for comparable times of flight. For the stages having less than 5 engines, the propellant depletion rates are lower than for Saturn V, and although initial propellant masses may be lower, they will later in flight exceed those in the Saturn V S-IC. The mass of propellant in the 5 F-1 stage begins exceeding Saturn V levels when three engines are cut off (about 102 seconds into the flight). The difference in propellant mass and acceleration will be sufficient to cause excessive tank bottom pressures. RP-1 fuel weight versus time for the 2, 3, and 4 F-1, 6-g INT-20 S-IC configurations and the 5 F-1 INT-20 S-IC stage are shown in Figure 3.1.1.2-6. Comparisons of Saturn V and 4 F-1 INT-20 accelerations and RP-1 levels as a function of time are made in Figure 3.1.1.2-7.

3.1.1.3 Base Heating

The 1966 INT-20 Study base heating analysis (Reference 3.1.1.1-1) showed that the heating environment at the base of the four F-1 S-IC would be less than on the Saturn V S-IC. This analysis was reviewed and affirmed. The five F-1 S-IC base environment would be similar to Saturn V S-IC, and two and three F-1 S-IC correspondingly less because of the lower heating rates associated with the fewer engines. Accordingly, it was assumed that for the trades analysis, base heating was not a major consideration for baseline vehicle selection.

3.1.1.4 F-1 Engine Analysis

The Rocketdyne Division of North American Rockwell Corporation states (Reference 3.1.1.4-1 and 3.1.1.4-2) that the F-1 engine can be operated for the durations required for INT-20 applications without modification and without an engine qualification test series. However, verification tests are recommended to demonstrate long duration engine operation (see Table 3.1.1.4-I for typical 100 N.M. mission burn schedule for each configuration). Rocketdyne's remarks regarding engine testing and the restraints and requirements imposed by INT-20 application are summarized below.

a. Verification Tests

The F-1 engine long-duration verification firing could be done at MSFC. NAR/Rocketdyne would require only observers at such a test series, and it is anticipated that no more than one F-1 engine would be required for an extended-duration engine test program.

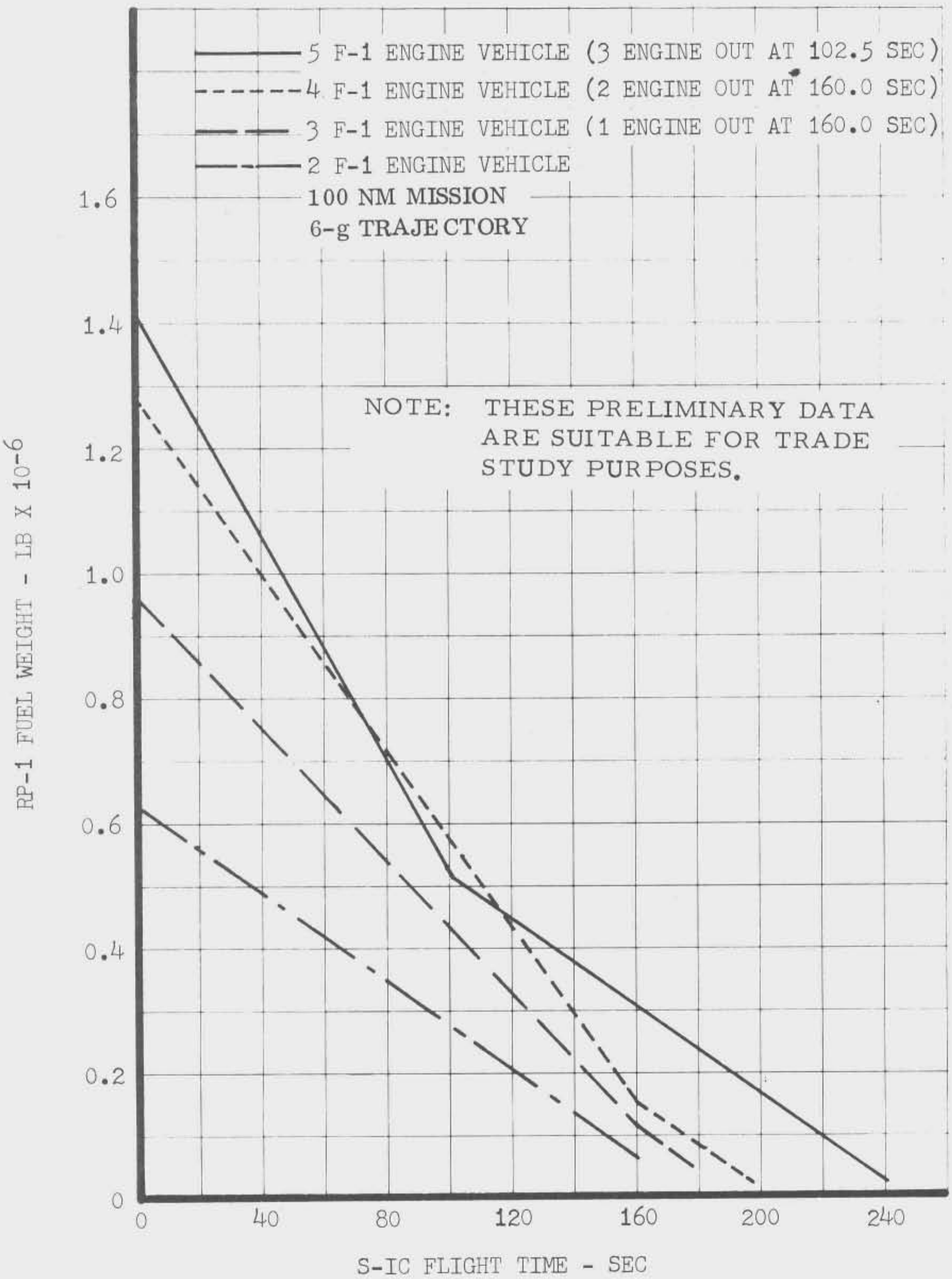


FIGURE 3.1.1.2-6 S-IC STAGE RP-1 FUEL CONSUMPTION

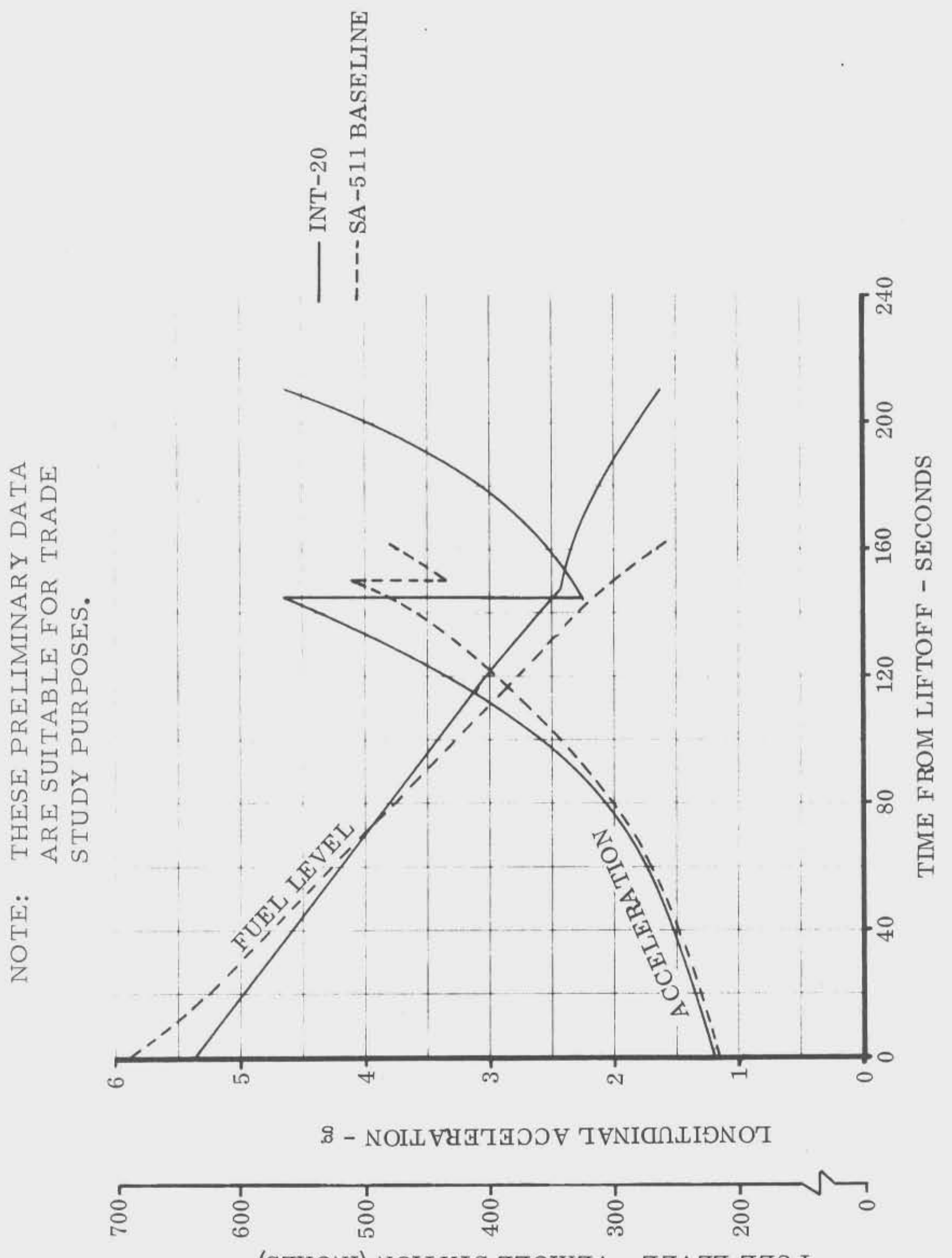


FIGURE 3.1.1.2-7 S-IC FUEL LEVEL AND ACCELERATION HISTORY FOR 4F-1/INT-20 AND SA-511 BASELINE VEHICLES

TABLE 3.1.1.4-1 PROGRAMMED F-1 ENGINE BURN TIMES
(100 NM MISSION)

	NO. OF S-IC ENGINES	TIME (SEC) FIRST CUTOFF	TIME (SEC) S-IC CUTOFF	CUTOFF SEQUENCE	FUEL BALLAST (LB)
4.68-g	2	146	146	2	175,781
	3	146	178	1-2	99,035
	4	146	211	2-2	19,403
	5*	103	241	3-2	-
6.0-g	2**	159	159	2	-
	3	160	178	1-2	-
	4	160	197	2-2	-

*950 PSF q MAX., 4.6-g MAX

**5.88-g MAX

NOTE: 5-F-1 VEHICLE WITH SMALL, HIGH-ENERGY PAYLOAD HAS MAXIMUM F-1 BURN TIME OF ABOUT 265 SEC.

NOTE: THESE PRELIMINARY DATA ARE SUITABLE FOR TRADE STUDY PURPOSES.

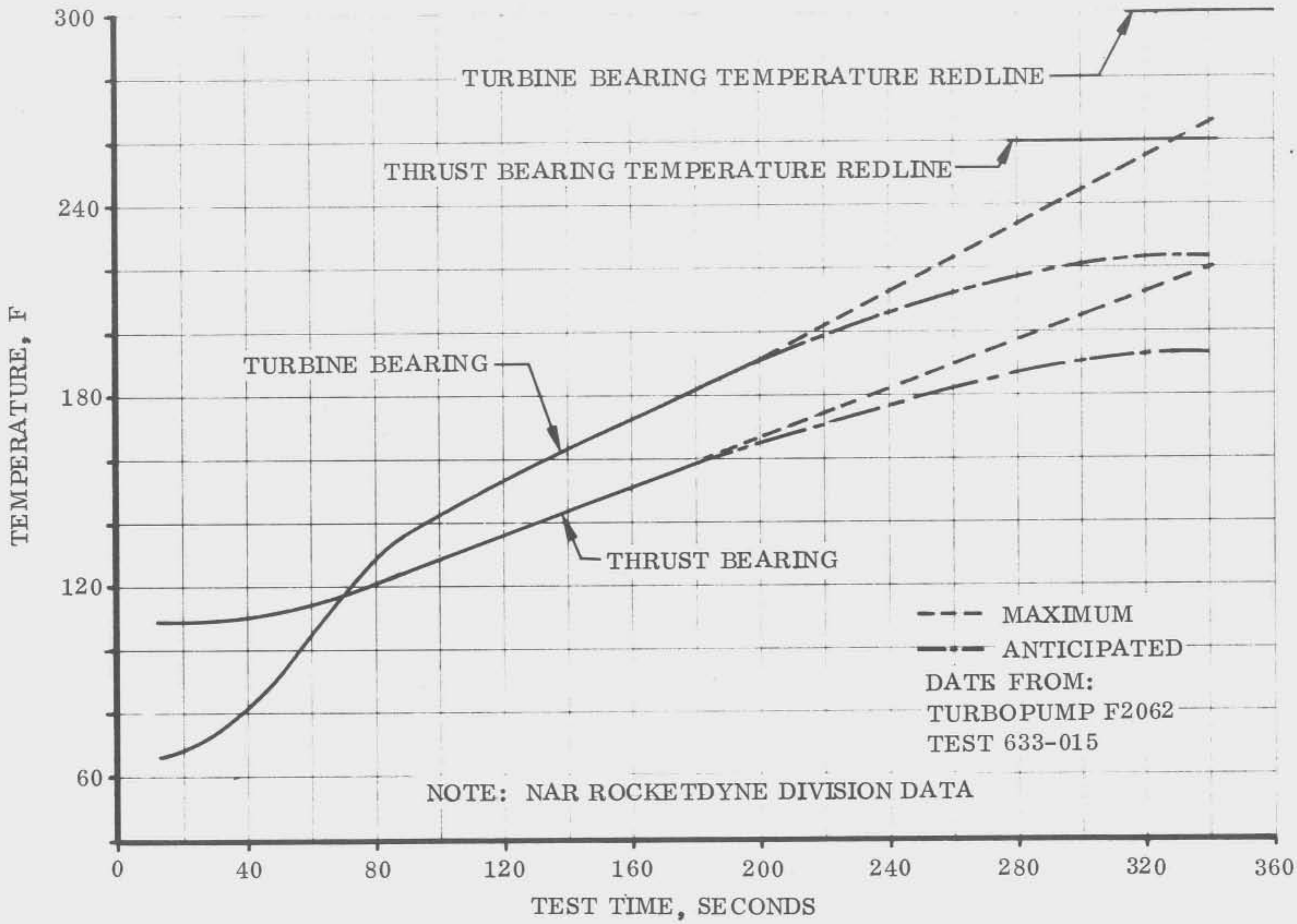


FIGURE 3.1.1.4-1 TYPICAL F-1 TURBOPUMP BEARING TEMPERATURES WITH EXTRAPOLATION TO 340 SECONDS

3.1.1.4 (Continued)

The F-1 engine has been run completely successfully for durations up to 194 sec. (maximum engine test stand duration) and turbopump tests have demonstrated completely successful operation at the 250-300 sec. operating duration range. Although these turbopump tests were at a 1300K equivalent thrust level, no operational difficulties are anticipated at the higher, 1522K thrust level for the longer required durations.

The verification of an extended duration continuous operation is required to insure that no operating limits are reached in the extended duration run associated with: (1) the bearing operation, (2) the seal operation, and (3) the turbine materials. These factors are reviewed in the following paragraphs:

1. During an engine firing the turbopump bearing temperature increases with time. Equilibrium conditions are not achieved. The rate of temperature increase decreases with time. This trend is shown in Figure 3.1.1.4-1. Based on extrapolation of test results, the maximum allowable (redline) bearing temperature should not be reached within approximately 340 sec. Thus the anticipated run duration should be entirely feasible.

2. Test results have shown that the wear rate of the turbopump rubbing seals (carbon) is constant with time. It is anticipated that this rate will not change during a longer run duration. Thus the operating life (qualification test demonstrated to 2250 sec) does not appear to be of consequence.

3. The turbine hardware experiences a temperature increase with time during engine operation in the normal 165 sec test runs. This causes some changes in the turbine blade impingement condition; actually the engine is designed for optimum performance under the heated conditions. Based on prediction of temperature rise in longer runs, extended operation (up to 340 sec) is not expected to effect engine operation.

b. Base Heating Considerations

Excessive temperatures may result on non-operating F-1 engine nozzle skirts and 4-way valve electrical connectors during any cluster static tests if protective measures are not taken. These components could be protected by either installation of the standard engine insulation (which is not normally used during static test), or by applying a water spray to the non-operating engines after their cutoff.

3.1.1.4 (Continued)

c. In-Flight Thermal Protection

Rocketdyne feels that the present thermal insulation should be completely adequate to prevent damage to a non-operating F-1 engine in a cluster during flight (i.e., F-1 engine shutdown prior to complete S-IC stage burnout).

d. In-Flight Purge

No in-flight purge is required for a non-operating engine if the engine is not to be re-used.

e. Fluid Power

There are no fluid power requirements for non-operating F-1 engines during flight.

f. Acceleration Effects

The F-1 and J-2 engines are designed to withstand 10 g's longitudinal acceleration. No problems are expected from the projected longitudinal acceleration for the Saturn V derivative (S-IC/S-IVB) launch vehicle.

3.1.1.5 S-IC Cost Analysis

The influence of varying the number of F-1 engines on S-IC stage development and production costs was determined. Stage development costs were found to be about the same for the 2, 3, and 4 F-1 engine versions. The total development cost for each of these versions is about $2.3 million. Development dollars required for the 5 F-1 version were found to be zero because no deletions are needed.

The estimated cost of performing 2, 3, or 4 F-1 S-IC stage development is basically engineering as follows:

Structures	462,000
Propulsion and mechanical	852,000
Electrical	113,000
Instrumentation and Misc.	626,000
	$2,053,000

3.1.1.5 (Continued)

Program support and management material, and other miscellaneous costs are about $247, 000, giving a total development cost of $2. 3 M. Each time an engine is omitted from the S-IC stage, the stage cost is reduced $0. 9 million by deleting the engine-related hardware. The cost of each F-1 engine deleted is $2. 0 million. The cost saving in engine support by using fewer engines is not well defined and is not included. The total cost savings for deleting F-1 engines from the S-IC stage is about $2. 9 million per engine omitted.

S-IC recurring costs vary with the production rate and with the configuration. The standard S-IC stage manufactured for 5 F-1 engines at 2 per year for 5 years is used as the reference and would have an average unit cost of $31. 4 million, (not including the costs of the F-1 engines). Increasing the production rate decreases the unit cost substantially. The reduction is 21% at 4 per year and 33% at 6 per year. The S-IC cost reduction for omitting engine related hardware is nearly constant at $0. 9 million per engine (3%). The standard S-IC stage is suitable for either 4. 68 or 6. 0 g max. acceleration (recurring cost is the same).

Rocketdyne recommends an engine verification test to demonstrate long-duration engine operation. It was concluded from the Rocketdyne data that the verification test program cost variation would not be significantly different between configurations.

3.1.2 S-IVB Stage

The following described data for the S-IVB stage were assembled and/or generated to support the selection of an INT-20 baseline vehicle configuration.

3.1.2.1 Stage Configuration

The stage configuration recommended for use on the INT-20 vehicle, and for which the strength, weight and cost data described in subsequent paragraphs were generated, is the Saturn V configuration, as pictured on Figure 3.1.2-1.

Configuration-wise, only the Saturn V is suitable for mating with the S-1C stage, having the 260-396 inch aft interstage. Structurally, the Saturn V skirts would certainly be required to withstand the expected environments of the intermediate vehicle. Current payloads, or weights above the S-IVB stage, are in the 40,000 lb class on the Saturn 1B and the 100,000 lb class for Saturn V. Thus, with the higher payloads expected for the INT-20, Saturn V structural elements would be necessary.

The propellant tankage is the area where configuration selection is not so obvious. Structurally, the tankage for the Saturn 1B and the Saturn V are identical; it is in the systems area that they differ, the major differences resulting from the restart capability of the Saturn V. Thus, the Saturn V requires among other things, more pressurant gases, in the form of both

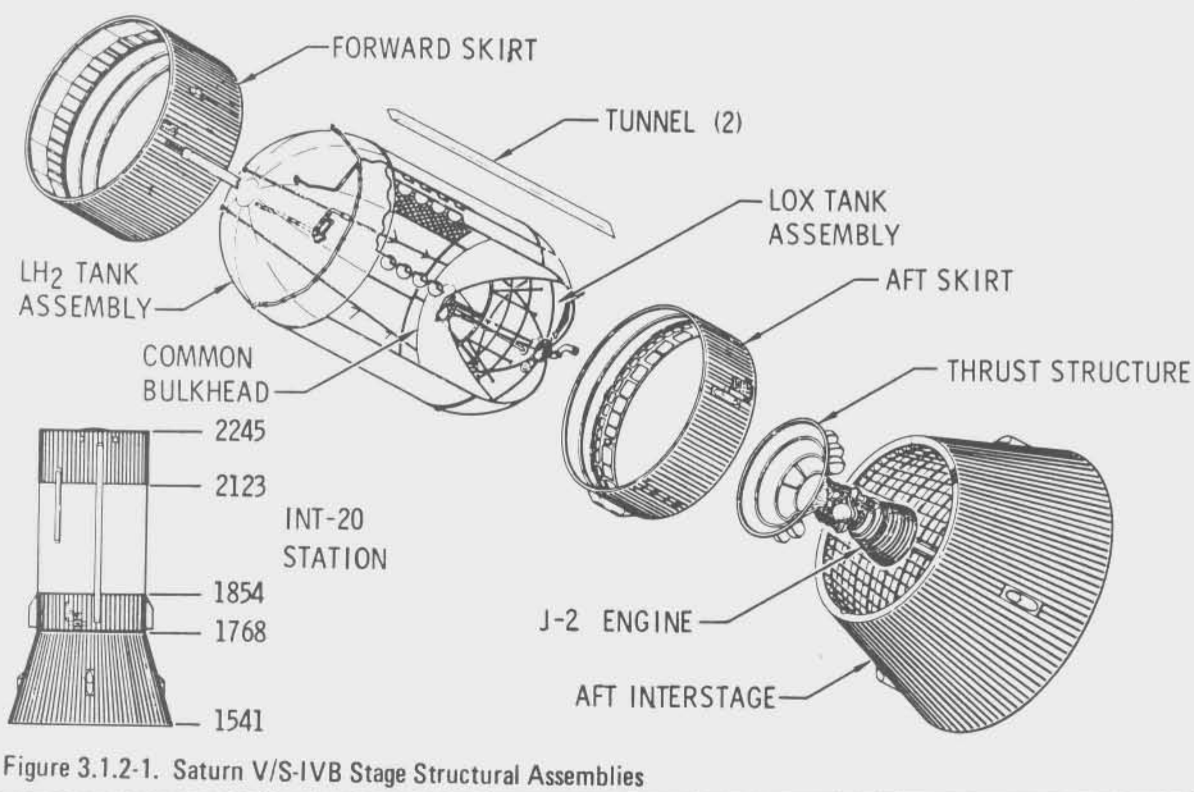


Figure 3.1.2-1. Saturn V/S-IVB Stage Structural Assemblies

added cold helium and ambient bottles, and a larger APS unit. Consequently, the stage weight is greater than that for the Saturn 1B.

The primary mission for the proposed vehicle -- resupply in low Earth orbit -- does not require restart of the second stage (S-IVB). Payload gains could result, however, from passing through a parking orbit at 100-nmi rather than ascending directly to a 270-nmi orbit. Further, alternate uses or missions, such as Hohmann transfers out of parking orbit, synchronous missions or lunar endeavors would certainly require the restart capability. Thus, it would be recommended that this versatility be retained. The cost difference attributed to the stage propulsion system is not that significant to consider a reconfiguration to a "mixed" stage. However, it would be possible to delete items not required on specific, non-restart missions.

Thus, it is concluded that stage development costs will be kept as low as possible by merely accepting the Saturn V stage. Also, only the Saturn V stage could be considered for a possible retro-fit for INT-20 use, considering propulsion systems (if restart would be required), replacement of skirts, new bolt patterns, new interstage required, etc.

3.1.2.2 Stage Weight Analysis

a. Baseline Stage Weights

Baseline S-IVB stage weight data are presented in Table 3.1.2-I. The first column presents the -511 basic stage data. This stage is primarily for the LOR mission and has a standard J-2 engine. The weight changes resulting from replacing the standard engine with the J-2S engine and performing those modifications required to support a synchronous orbit mission are given in the second column. These data were derived from the J-2S Improvement Study (Reference 3.1.2-1). The last column gives the resulting weights for modifying the S-IVB stage with standard J-2 engine to do the synchronous mission. These figures were taken from Reference 3.1.2-2. All the preceding weight figures were derived from the -511 baseline numbers.

Finally, the baseline aft interstage weights are presented on Table 3.1.2-II. These data are not affected by engine system. Note that the weights for the retro-rocket system are not included in the table.

TABLE 3.1.2-I

S-IVB Stage Dry Weight Data

	NASA Second Generation Breakdown	S-IVB-511 J-2 Baseline	S-IVB-511 J-2S Sync. Mission	S-IVB-511 J-2 Sync. Mission
W3.3	Propellant Container	8,933	9,232	9,628
W3.6	Forward of Tanks	1,242	1,242	1,342
W3.8	Aft of Tanks	1,816	1,801	1,829
W3.9	Thrust Structure	774	809	774
W3.10	Fairings & Assoc. Struct.	197	174	197
W3.15	Paint & Sealer	104	104	104
W3.18	Heat & Flame Protection	182	182	182
W3.0	Structure	13,248	13,544	14,056
W4.1	Engine & Accessories	3,572	4,073	3,572
W4.6	Purge System for Chilldown	272	0	272
W4.7	Fuel System	1,573	1,067	1,338
W4.8	Oxidizer System	1,264	1,111	1,264
W4.9	Cryogenic Repress. System	310	310	368
W4.10	Stage Control Sys. Hdwe.	284	284	284
W4.0	Propulsion System	7,275	6,845	7,098
W6.1	Equip. & Instru. Struct.	430	431	922
W6.2	Environ. Control System	231	268	231
W6.5	Control System Electron.	116	116	116
W6.8	Telemetry & Meas. Sys.	1,165	1,533	1,189
W6.10	P. U. System	175	175	175
W6.11	Electrical System	829	1,060	1,085
W6.12	Range Safety System	69	69	69
W6.15	Pneumatic System	298	269	298
W6.16	Auxiliary Prop. Sys.	855	829	855
W6.17	Separation System	117	117	117
W6.18	Ullage System	212	0	212
W6.20	System for Total Veh.	91	91	103
W6.0	Equipment & Instrumen.	4,588	4,958	5,372
WAD	Stage Dry Weight	25,111	25,347	26,526
	Change from S-IVB-511 Baseline	0	(+236)	(+1,415)

TABLE 3.1.2-II

S-IVB Aft Interstage Weight

NASA Second Generation Breakdown	S-IVB-511 Baseline
W3.13 Interstage Structure	5,678
W3.15 Paint & Sealer	49
W3.18 Heat & Flame Protection	523
W3.0 Structure	6,250
W6.2 Environ. Control Sys.	17
W6.8 Telemetry & Meas. Sys.	15
W6.12 Range Safety System	2
W6.17 Separation System	727
W6.20 System for Total Vehicle	10
W6.0 Equipment & Instr.	771
WBD Dry Weight	7,021

b. Baseline Stage Structural Capability

The structural capability of the baseline S-IVB-511 stage is indicated on Figure 3.1.2-2. These data were obtained from the J-2S Improvement Study results (Reference 3.1.2-1). Allowable compression loads in pounds per inch are shown for both the Max qα and the Max acceleration load conditions. Since the latter condition is also one of peak structural heating, temperatures are shown with the allowable loads. The liquid hydrogen tank sidewall is generally critical in the unpressurized, ground wind condition; thus that condition is shown on the chart.

The given value of aft skirt allowable was felt to be conservatively low, since it was based on local stringer allowables in the area of proturberance heating. For purposes of the trade study analysis, this allowable was revised to a value more in line with the expected degradation in compressive yield strength due to the indicated structural heating. Local effects would have to be considered in a detail design phase.

S-IVB-511 stage stiffness data are included herein as Figure 3.1.2-3. These data do not reflect any stage beef-ups as may be required for INT-20 use.

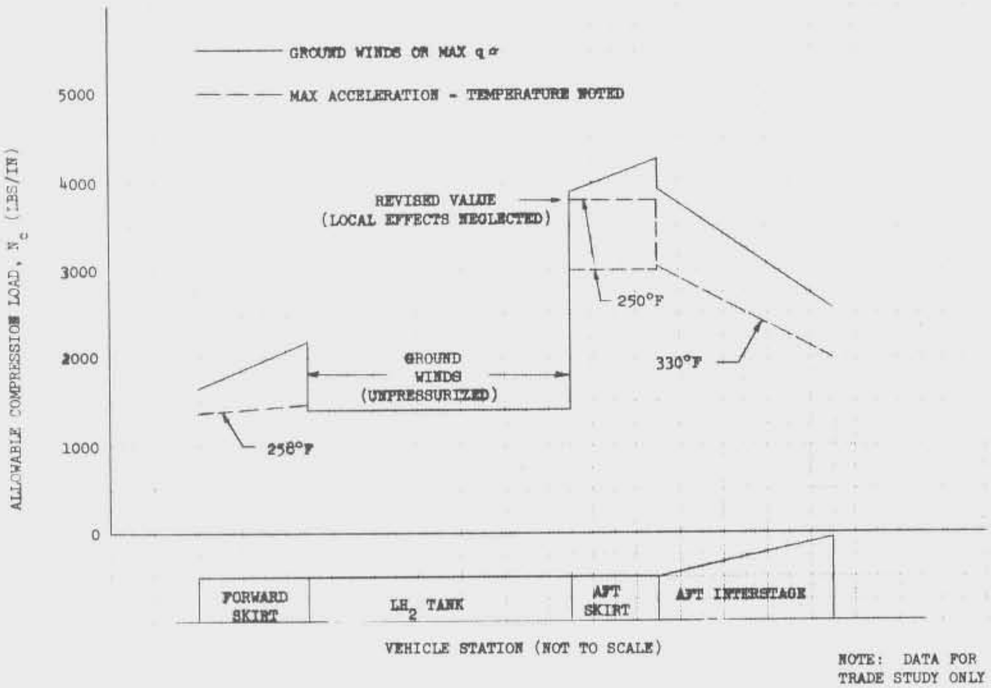


Figure 3.1.2-2. Saturn V/S-IVB Stage Structural Capability

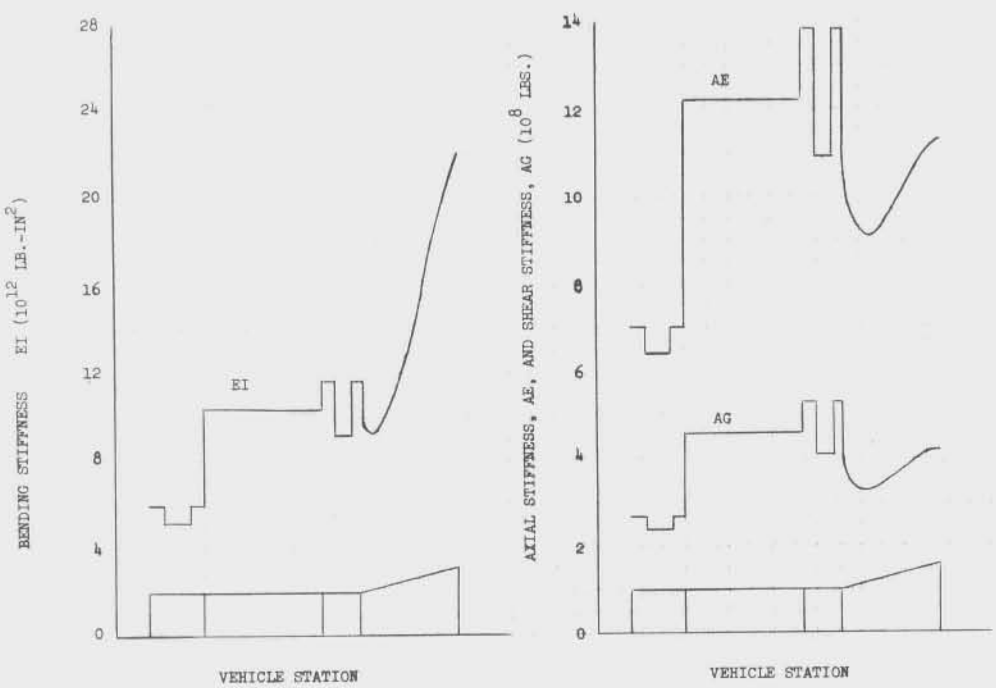


Figure 3.1.2-3. S-IVB Stage Stiffness

c. Baseline Stage Weight Changes

The changes in baseline stage weight were determined as a function of maximum acceleration (4.68 or 6.0 g's), safety factor (1.40 or 1.25) and payload weight (100,000 to 160,000 lbs). The approach taken was to first determine estimated structural loads for the various parameters and conditions, compare these with the stage allowables, and then estimate the weight change necessary if the allowables were exceeded. Details follow.

1. Design Loads

Design loads, in lbs/in. of compression, were calculated or estimated for two conditions, Max qα and Max a_x. For both conditions, the payload weight was varied from 100,000 lbs to 160,000 lbs and the safety factor was taken at 1.25 and at 1.40. In addition, Max a_x loads were calculated using both 4.68 and 6.0 g's. The Max qα condition was based on results from the previous INT-20 vehicle study (Reference 3.1.2-3). The vehicle bending moments and axial drag values were taken from those results (4F-1 engine baseline case) as were the axial acceleration values at time of Max qα to calculate total axial loads.

In order to calculate axial loads, weight above the S-IVB stage was considered to be comprised of an instrument unit (IU) weighing 3,850 lbs, a payload weighing from 100,000 to 160,000 lbs, and an LES weighing 8,200 lbs.

The results of these calculations are illustrated on Figures 3.1.2-4, 3.1.2-5, and 3.1.2-6, which present the design loads as a function of payload for the forward skirt, aft skirt and aft interstage, respectively. The left hand portion of the curves shows loads for the Max qα condition, safety factor 1.25 and 1.40. The right hand portion shows Max a_x condition loads for both safety factors and for 4.68 and 6.0 g's. Also indicated on each portion is the allowable load for that structural element. From these data it was possible to determine a limiting payload for each structure, each condition, and the amount of additional capability required for payloads above those limits.

Note that no load calculations were performed for the hydrogen tank sidewall section, as that structure has adequate strength to withstand greater increases in loading than will result in this proposed usage.

2. Comparison Results

The results of the load comparison indicated that in no case was the Max qα load condition critical. Further, for both the forward and aft skirts, no changes were necessary for loads at a max acceleration value of 4.68, whereas for the aft interstage, some

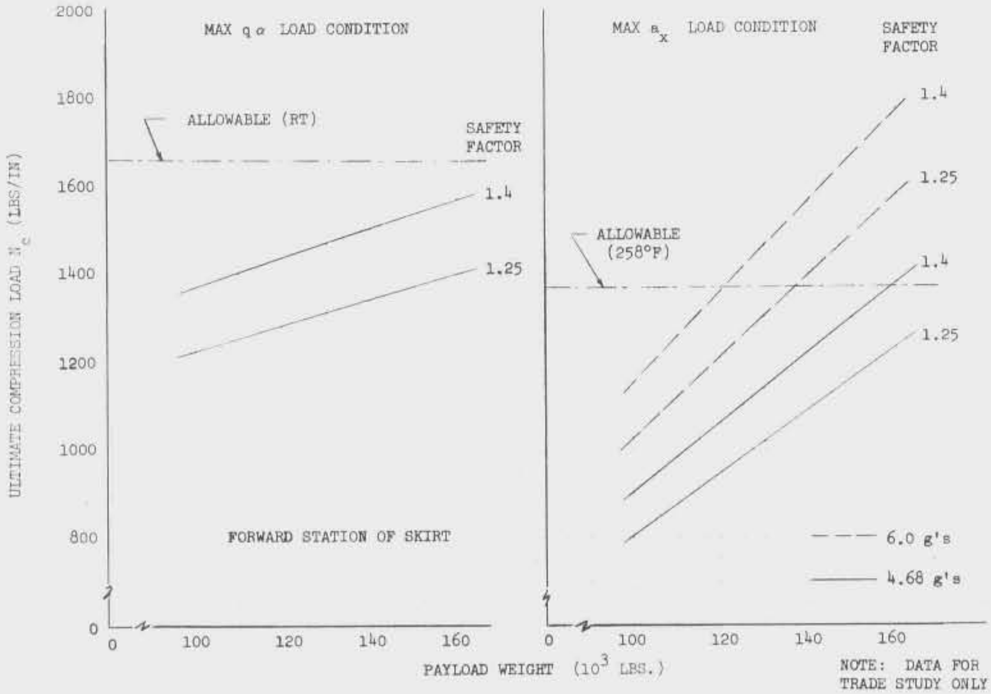


Figure 3.1.2-4. S-IVB Forward Skirt Design Loads

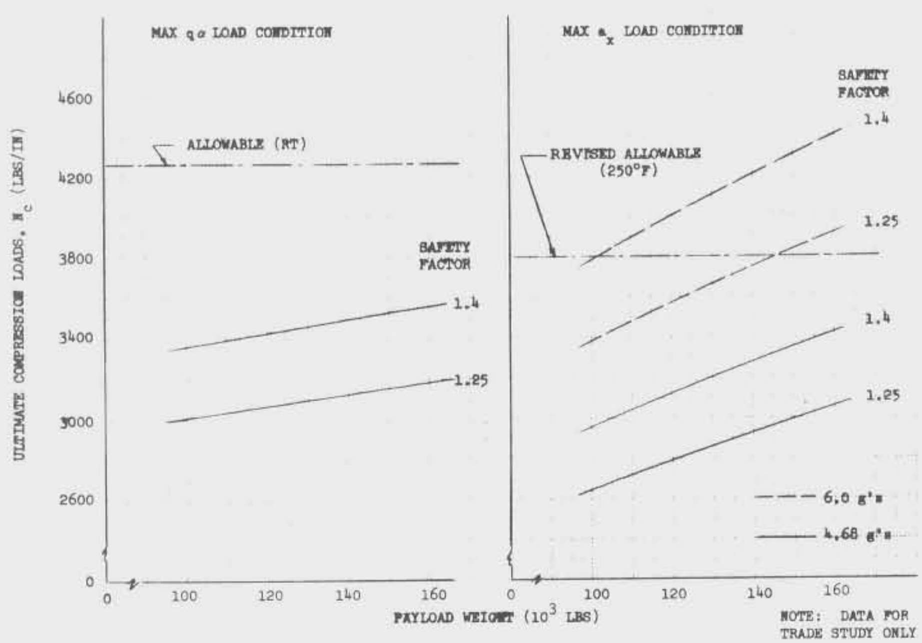


Figure 3.1.2-5. S-IVB Aft Skirt Design Loads

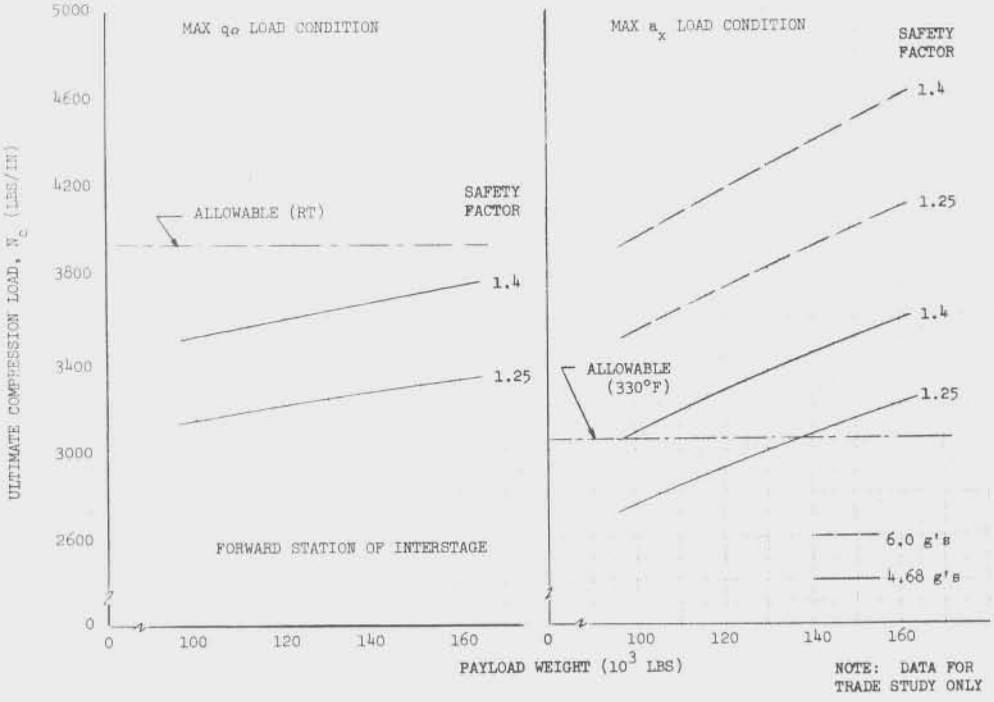


Figure 3.1.2-6. S-IVB Aft Interstage Design Loads

change was indicated for any payload in excess of 95, 000 lbs (F. S. = 1. 4). These results are illustrated on Figures 3. 1. 2-7 and 3. 1. 2-8.

Figure 3. 1. 2-7 shows the forward and aft skirt weight changes, which are only required for the 6. 0 g load condition. For the forward skirt, the limiting payload is 160, 000 lbs at 4. 68 g's and F. S. = 1. 4, 122, 000 lbs at 6. 0 g's and F. S. = 1. 4, and 137, 000 lbs at 6. 0 g's and F. S. = 1. 25. For the aft skirt, the limiting payloads for the 6. 0 g condition are 102, 000 lbs. (F. S. = 1. 4) and 145, 500 lbs (F. S. = 1. 25).

For the aft interstage, Figure 3. 1. 2-8, some weight changes are indicated for virtually all load and factor conditions except under 137, 500 lbs at 4. 68 g's and F. S. = 1. 25.

Figure 3. 1. 2-8 also summarizes S-IVB stage dry weight as a function of payload weight (interstage not included). This curve incorporates the tankage weight changes with those of the forward and aft skirts.

The propellant tankage is currently designed to withstand 4. 68 g's at a F. S. = 1. 4. The tankage can withstand 6. 0 g's if the safety factor is lowered to 1. 25. Thus, the only changes result when the g level is upped to 6. 0 with the current F. S. at 1. 4. The results in that case are a 100 lb weight increase in the hydrogen

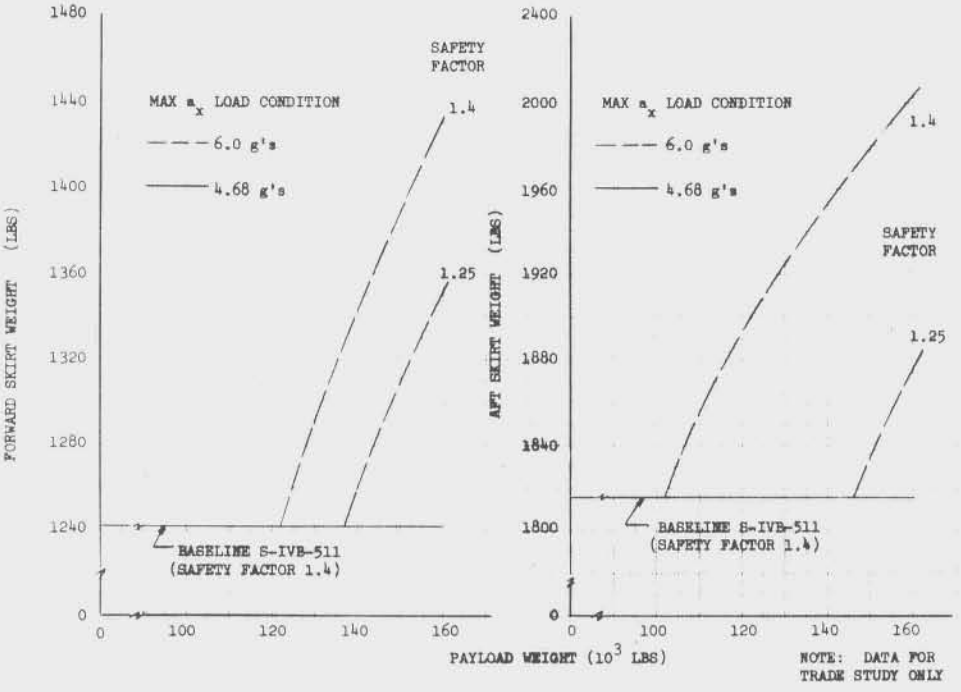


Figure 3.1.2-7. S-IVB Forward and Aft Skirt Weights

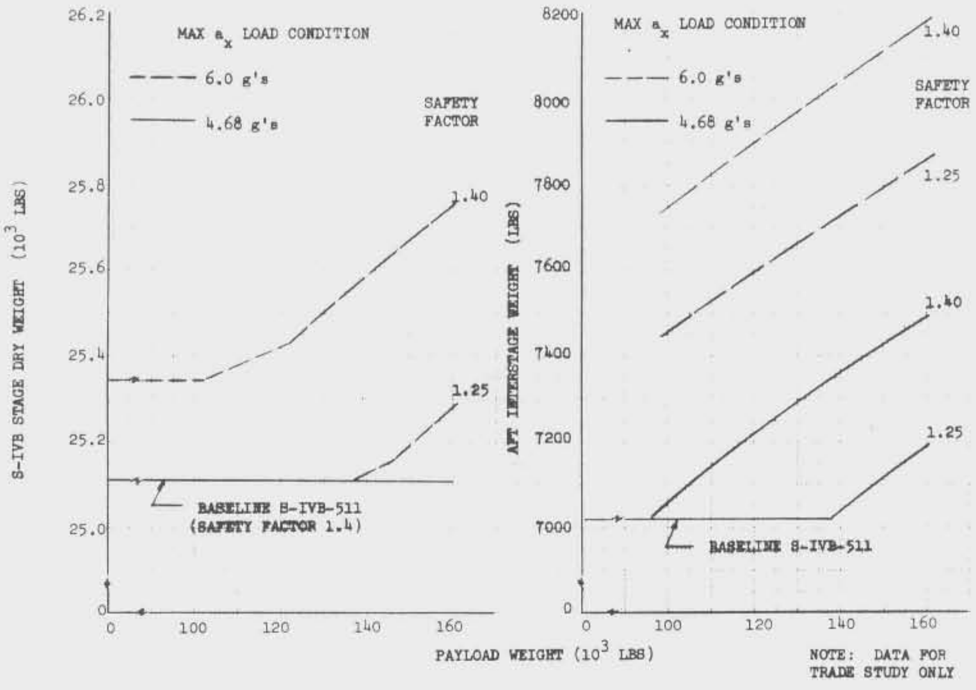


Figure 3.1.2-8. S-IVB Stage and Interstage Weights

tank and a 130 lb weight increase in the oxygen tank. Consequently, the weight curve for the 6.0 g, F.S. = 1.4 case on Figure 3.1.2-8 starts at a level 230 lbs higher than the S-IVB-511 baseline.

Note stage weight changes are only indicated for the 6.0 g conditions.

3. Summary

Some comments concerning the previous analyses must be offered.

Since the results shown are based on Max qα loads and/or parameters obtained from the previous study (Reference 3.1.2-3), they are subject to change as the payload configuration changes, i.e., the MLV payload shape, as proposed for this study would generally result in increased bending moments at Max qα as compared to the Apollo payload shape. Thus that load condition could become critical.

If structural temperatures at the time of max acceleration should exceed the values indicated on the allowable curve, further modification, or at least the addition of insulation, would be required. As previously mentioned, some modifications other than indicated may be required for the aft skirt due to local thermal load effects. These modifications would take the form of added thermal insulation.

One final comment is offered. In no case was there consideration of reducing stage weight due to reduced loads, e.g. reducing tank thicknesses due to a lowering of the safety factor to 1.25, 4.68 g's condition. This was not considered to be compatible with the overall study objectives.

3.1.2.3 Stage Cost Analysis

The following described cost data were assembled and/or generated to support the selection of an INT-20 baseline vehicle configuration. These cost data, which are development (non-recurring) costs only, were estimated in accordance with the ground rules applicable to the subject study (hence, no hardware costs associated with R&D flight vehicles are included).

a. Baseline Stage

The Saturn V/S-IVB stage S-IVB-511 is the stage from which the S-IVB-INT-20 baseline stage is derived. The development cost for modifying the -511 stage and interstage from their present configuration to the INT-20 configuration was estimated to be from a minimal amount which could readily be absorbed in the normal sustaining engineering base to a maximum amount of approximately \$7,000,000. The wide range was the result of having insufficient definition of the interface changes required to match the S-IVB and S-1C stages.

The lower bound is representative of merely having to adapt the S-IVB electrical interface to the S-IC stage, i.e., all changes required for structural interface would be accomplished on the S-IC stage. On the other hand, the maximum cost quoted (\$7M) represents the estimated costs involved with redesign and retest of the S-IVB aft interstage to accomplish all structural interface adaptions, i.e., adapting to the pattern of 216, 1/2-inch bolts at the present S-IC bolt circle diameter, as compared to the present S-IVB pattern of 288, 3/8-inch bolts.

The above quoted development costs are for the S-IVB-INT-20 stage designed for a maximum of 4.68 g's axial acceleration with a safety factor of 1.4 or 1.25, and assuming the shell structure capability is not exceeded. With the exception of the aft interstage, the stage as just defined is capable of carrying payloads over the range of 100,000 to 160,000 lbs (based on the previously described structural allowable data).* In order to qualify the aft interstage structure over this entire payload range, with either safety factor, an additional development cost of \$140,000 must be included. This figure would be valid for the entire payload range using a 1.4 safety factor, and for payloads in excess of about 132,000 lbs if a 1.25 safety factor were used.

b. 6.0 g Capability Stage

The additional development costs for providing the S-IVB-INT-20 stage with 6.0 g capability are summarized on Table 3.1.2-III.

Since conditions 1 and 2 specify that the shell structure is adequate, only tankage changes are involved, and as shown, only for the 1.4 safety factor condition. These changes involve slightly increased skin thicknesses on the hydrogen tank sidewall and the aft dome.

For conditions 3 and 4, wherein shell structure capability is exceeded, the total cost is obtained by including costs for modifying each of the primary structural elements, e.g., forward skirt, aft skirt, etc. According to the structural weight change analysis previously presented, these changes would occur in increments as the payload weight increased. Consequently, Figure 3.1.2-9 is included to reflect this incremental change in costs. As is shown, the entire increase in development costs as presented in Table 3.1.2-III are applicable for payloads over 122K and 145.5K for the conditions of 1.4 and 1.25 safety factor, respectively.

*Later study effort demonstrated aft interstage capability up to 132,000 lbs, safety factor of 1.4, with proper insulation.

TABLE 3.1.2-III

6 g DEVELOPMENT COSTS

Condition	Safety Factor	Structure	Cost
1	1.25	Shell Structure Capability Not Exceeded	0
2	1.40		$ 120,000
3	1.25	Shell Structure Capability Exceeded	$1,320,000
4	1.40		$1,440,000

c. Modified Engine Baseline Stage

The additional development costs required to provide the S-IVB-INT-20 stage with synchronous mission, three-start capability (two re-starts) using the standard J-2 engine are $2,200,000.

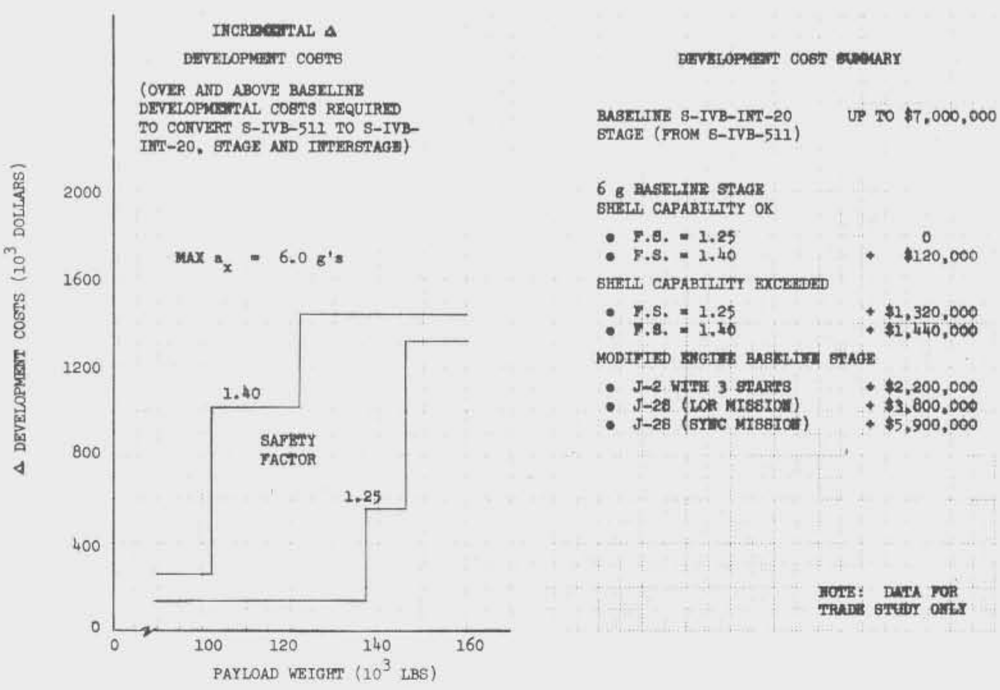


Figure 3.1.2-9. S-IVB-INT-20 Stage Development Costs

Replacing the standard J-2 engine with the J-2S engine on an INT-20 S-IVB stage requires an additional development cost of \$3, 800, 000 (not including engine development costs). This figure is for a typical two-start mission mode, i. e., LOR condition.

In order to replace the standard J-2 engine with a J-2S engine capable of three starts, and provide other modifications as required to accomplish a synchronous orbit mission, an additional development cost of \$5, 900, 000 will be required (not including engine development costs).

d. Summary

Each of the previous additional development costs, as summarized on Figure 3. 1. 2-9, are independently additive to the S-IVB/INT-20 baseline development cost. For example, an S-IVB stage with a three-start J-2S engine and 6. 0 g, 1. 4 safety factor and 160, 000 lb payload capability would incur a development cost of \$7, 340, 000 (\$5, 900, 000 for the engine adaptation and \$1, 440, 000 for the structural modifications) over and above the cost necessary to convert an S-IVB-511 to an INT-20 baseline (not including engine development costs).

All costs are quoted in 1968 dollars and include fee. They are preliminary and subject to refinement as the result of more detailed investigations later in the study. No provisions were made for start-up costs and it was assumed that no rate-type facility costs would be incurred.

NOTE

The data contained in this section are preliminary in nature, and were prepared for purposes of conducting the necessary Trade Studies. These data were subsequently superceded by detailed investigations in the latter phases of the study.

For example, it was determined in the vehicle design phase that the S-IVB aft interstage with an application of 0. 01-in. of Korotherm insulation would be satisfactory for INT-20 vehicle application. Further, the interface problem was investigated in some depth, and it was determined that the development costs for implementation would be quite low (as opposed to the "worst case" condition quoted here in the Trade phase).

3.1.3 IU

3.1.3.1 Summary

The Trade Studies conducted in Phase I indicate that the Saturn V IU is more suitable for conversion to an INT-20 IU than the Uprated Saturn I* IU. The impact of 4.68g and 6.0g in-flight acceleration and choice of 2, 3, 4, and 5 F-1 engines was investigated. The following paragraphs highlight the considerations in the choice of the Saturn V IU.

a. Load Relief

The INT-20, like the Saturn V vehicle, is not expected to require load relief during S-IC burn. Lateral accelerometers are not used in the Saturn V IU as they are in the Uprated Saturn I for load relief.

b. Command and Control System

The Saturn V Command and Control System (CCS) is not used in the Uprated Saturn I. Required replacement of the VHF with UHF by 1975 makes the Uprated Saturn I IU less attractive than the CCS system which is compatible with the Unified S-Band Systems. On the other hand, $102,000 per unit is saved by substituting an Uprated Saturn I Command System and UHF telemetry for the Saturn V CCS. The issue does not decide the choice of Saturn V IU per se but does offer a no cost impact with choice of the current Saturn V IU. The Synchronous mission requirement decides the issue in favor of the Saturn V IU.

c. Structures

The impact of loads and environmental effects for various INT-20 configurations favors the choice of the Saturn V IU because acoustic damping material and thermal protection (cork) has been added to the Saturn V IU and not to the Uprated Saturn I IU. The same treatment of the Uprated Saturn I IU would provide the same relative advantage with modest additional engineering.

d. Environments

Vibration exceedence in selected IU locations is common to Saturn V and Uprated Saturn IU and is not a factor in choice of IU.

*The Uprated Saturn I is another designation for the Saturn IB.

3.1.3.1 (Continued)

e. Flight Control

Prior to choice of the IU, a preliminary redesign of the Saturn V Flight Control Computer showed the feasibility of providing thrust vector control to any number of F-1 engines and providing reversibility. The requirement to drive four F-1 engines rather than four H-1 engines clearly decided the choice of the Saturn V Flight Control Computer.

f. Interface (IU/S-IVB)

From a networks standpoint, significantly lower level of effort would be required to modify a Saturn IU to the INT-20 configuration than to modify the Uprated Saturn I. Because S-II networks can be electrically open ended or in isolated cases provided with dummy loads, it is feasible to provide reversibility in networks.

3.1.3.2 Uprated Saturn I vs Saturn V S-IVB/IU Interface

a. Introduction

A comparison of the S-IVB/IU Interface between the S-IB IU and the S-V IU as presently designed reveals that the physical location of all nine interface connectors is the same for both IU's. Therefore, assuming a S-V S-IVB stage on the INT-20 vehicle, a S-IB IU or a S-V IU could be used without any relocation of connectors at the S-IVB/IU Interface.

b. S-IB IU Usage

If a S-IB IU design were used for the INT-20, as shown in Figure 3.1.3.2-1, all the S-IB functions would require deactivation, and all the S-IC functions would have to be added. These changes would require extensive networks design changes to allow a S-IB IU to function on the INT-20 vehicle.

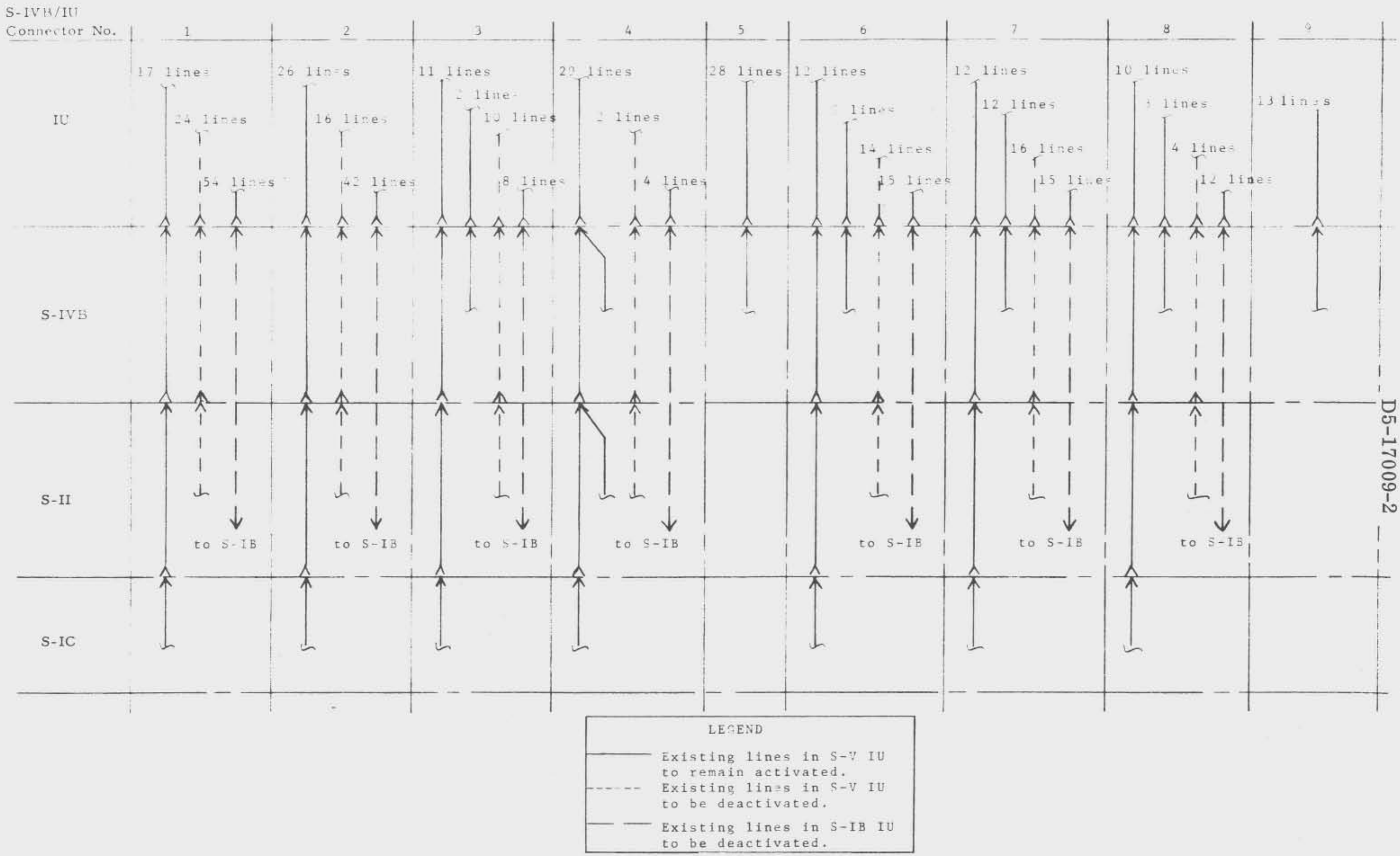


FIGURE 3.1.3.2-1. S-IVB/IU INTERFACE SCHEMATIC

3.1.3.2 (Continued)

c. S-V IU Usage

If a S-V design were used as a baseline, all the S-IC functions shown on Figure 3.1.3.2-1 would already exist. The only networks design changes necessary would be those required to deactivate the S-II functions existing in a S-V IU. The magnitude of these design changes is much less than that for the conversion of a S-IB IU for the same function.

d. Conclusion

Therefore, from a networks standpoint it appears that a significantly lower level of effort would be required to modify a S-V IU to the INT-20 configuration than to modify a S-IB IU to the same configuration. Using a S-V IU baseline, the networks design would be modified to deactivate the S-II functions not required for the INT-20 vehicle.

3.1.3.3 Uprated Saturn I Versus Saturn V IU Command System

a. Introduction

One distinct difference between the Saturn IB Instrument Units (200 series) and the Saturn V Instrument Units (500 series) is the Command System. The Saturn IB Instrument Unit utilizes a UHF Command Receiver (450 MHz) for reception of commands and a VHF Telemetry System for transmission of verification messages. The Saturn V Instrument Unit utilizes a CCS Transponder for both reception of commands and transmission of PCM telemetry data (includes verification message). The verification signal is also transmitted redundantly via VHF telemetry.

Four possibilities exist for selection of an IU/Command System for INT-20 vehicles. They are as follows:

Saturn IB IU and Saturn IB Command
Saturn IB IU and Saturn V Command
Saturn V IU and Saturn IB Command
Saturn V IU and Saturn V Command

This portion of the study effort will address only the Command Systems, not the Instrument Units.

Of major importance in the selection of a system are the vehicle missions. The missions are defined to be either low earth orbit or synchronous orbit missions.

3.1.3.3 (Continued)

b. Analysis

Factors affecting the selection of a system are communications capability, system function, cost, reliability, availability, ground support requirements, frequency assignment, RF interference, and input power requirements as well as the overall Saturn Communication Systems philosophy. Table 3.1.3.3-I gives a summary of the merits of each system related to the pertinent factors. Each of these factors will be evaluated separately.

1. Communications Capability

The Saturn IB Command System is not capable of operating at synchronous orbit altitudes. The UHF uplink needs an additional 14 db gain to ensure reliable communication with the vehicle. The VHF downlink needs an additional 13 db gain to supply an adequate signal to ground stations. For these reasons, only the Saturn V Command System will suffice for synchronous orbits. Even the CCS operation is somewhat marginal. A link margin of 1-7 db exists for the CCS downlink. The vehicle must be stabilized to achieve satisfactory communications using the CCS link.

2. System Function

The two systems perform the same command functions. The Saturn V System also handles telemetry transmission and has the capability of receiving and transmitting ranging information.

3. Cost

Saturn IB system components (Command Receiver) that are not common to the Saturn V system components cost $9,000/vehicle. An additional cost of $13,000 for UHF equipment will be required if NASA usage of VHF telemetry is phased out before 1975 as jointly agreed upon by the Department of Defense and NASA. The cost of the Saturn V CCS components not common to the Saturn IB is $124,000/vehicle. Substitution of a Saturn IB Command System and UHF telemetry for the Saturn V CCS would result in a savings of $102,000/vehicle. See Section 3.1.3.3c. for recommended approach.

4. Reliability

The reliability of the Command Receiver for Saturn IB is 0.999813 for a 4.7 hour orbital mission. The reliability of the Saturn V CCS Transponder Power Amplifier is 0.9987 for a 6.78 hour mission. No figures

TABLE 3.1.3.3-I. COMMAND SYSTEM SELECTION

	Saturn IB System		Saturn V System	
	Synchronous Orbit	Low Earth Orbit	Synchronous Orbit	Low Earth Orbit
COMMUNICATIONS CAPABILITY	UNSATISFACTORY, VHF-TM, UHF uplink inadequate	GOOD	SATISFACTORY, Requires vehicle orientation	GOOD
SYSTEM FUNCTION		COMMAND ONLY		COMMAND TELEMETRY RANGING
COST Components common to both systems not included		$9,000 +$13,000 IF UHF-TM REQUIRED		$124,000
RELIABILITY		VERY RELIABLE Minimum of circuitry		FAIR Relative to Saturn IB System, due to complexity and multi-function
AVAILABILITY		SOURCE AVAILABLE but new contract needed		SOURCE AVAILABLE but new contract needed
GROUND SUPPORT REQUIREMENTS		PRESENTLY AVAILABLE		GOOD Same equipment required as for Unified S-Band Systems
FREQUENCY ASSIGNMENT		VHF-TM USED FOR VERIFICATION NOT AVAILABLE BY 1975		GOOD
RF INTERFERENCE		FAIR		GOOD
POWER REQUIREMENT		3.5 WATTS		130 WATTS (95 watts required for power amplifier)
SATURN PHILOSOPHY		PLANNED TO BE PHASED OUT		CONFORMS TO 'UNIFIED' CONCEPT

3.1.3.3 (Continued)

are available on the CCS Transponder, but the reliability figure should be lower than that of the Power Amplifier due to the greater number of components in the CCS Transponder.

5. Availability

Procurement sources exist for both the Saturn IB and Saturn V system.

6. Ground Support Capability

Ground Stations exist for both systems, but it appears that it would be more economical to operate the CCS system. The Saturn V CCS system was developed in conjunction with the Unified S-Band Systems presently on Command Modules and Lunar Excursion Modules. The USB ground stations are capable of supporting either the CCS or USB. The USB systems will probably be used on most future payloads developed by NASA. Therefore, use of the CCS (Saturn V System) would reduce the operational requirements by deleting the requirement for separate 450 MHz ground transmitters and associated equipments.

7. Frequency Assignment

The Department of Defense and NASA have reached an agreement that NASA will vacate the VHF telemetry range (225 MHz to 400 MHz) by 1975 (NASA Memorandum NMI 1052.111, dated 30 August, 1968). If an S-IB System is used, this agreement will necessitate the addition of a UHF-TM System on the IU in order to transmit the required telemetry data (including command verification data) that are presently transmitted via VHF telemetry. This UHF Transmitter addition would increase the number of onboard systems if the Saturn IB Command System is selected. The CCS Link would not be impacted by this agreement as the telemetry downlink is in the UHF band.

8. RF Interference

No RF interferences to either system have occurred since the early days of the Saturn program; however, Saturn RFI Math Model (developed specifically for the Saturn Vehicle) Predictions indicate that the probability of interference is greater for the Saturn IB system (450 MHz) than for the CCS system.

3.1.3.3 (Continued)

9. Power Requirements

The Saturn IB Command System requires 3.5 watts input power for operation and the CCS requires 130 watts. The large power consumption difference is caused by the large power requirement of the transmitter power amplifier in the CCS. The Command function requires very little power. An additional 224 watts would be required by a UHF telemetry transmitter for the Saturn IB system should the VHF telemetry be removed from the vehicle.

10. Saturn Philosophy

The 'Unified' concept was developed in order to support lunar missions. Communications, telemetry, and tracking are incorporated into one system. A reduction in equipment and an increase in range capability is obtained. The Saturn V Instrument Unit Command System utilizes the CCS Transponder which has the capability of performing the same functions as the Unified S-Band Systems located in the Saturn Payload modules.

A selection of the Saturn IB Command System for follow-on vehicles would violate this 'Unified' philosophy. The other functions performed by the CCS (telemetry and ranging) would have to be performed by other equipments.

c. Conclusions

The Saturn IB Command System is superior to the Saturn V CCS in cost, reliability, and power requirements, but inferior in communications capability, system function, ground support requirements, frequency assignment and RF interference.

The Saturn IB Command System should be selected only if <u>all</u> of the following conditions are met:

The missions are to be low earth orbit missions only.

Reliability, power requirements, and on-board equipment cost are of major importance.

The missions are completed prior to 1975. Otherwise, UHF telemetry transmitters will be required.

3.1.3.3 (Continued)

The Saturn V CCS is preferred because of the limitations on the Saturn IB Command System listed above and the following additional factors favorable to the CCS:

The CCS Transponder used for Command signal reception also serves as a UHF telemetry transmitter (PCM Data including command verification) and has the capability of being used for ranging.

The CCS and Unified S-Band Systems used on Saturn V vehicles can use identical ground station equipment. Uniformity of the systems is more economical than using two different systems.

Only the CCS is capable of communications in synchronous orbit.

The CCS System on the Saturn V Instrument Unit and the Unified S-Band Systems on the Spacecraft conform to the 'Unified' philosophy. The systems were intended to reduce the amount of on-board equipment and increase the communications range. Selection of the Saturn IB Command System would be contrary to the philosophy.

3.1.3.4 Flight Control Computer (FCC) Modifications

a. Introduction

The FCC will require modification in order to meet the additional requirements of the INT-20 configurations. The basic requirements are:

Four S-IC Switchpoints.

No S-II Stage.

Elimination or modification of unused S-II hardware.

Modification of unused Servo Amps for two engine configuration.

The only constraint placed on the modification is reversibility. That is, minimum modification should be required to change any INT-20 configuration to the S-V configuration and vice-versa.

b. FCC Hardware Impact Assessment

In order to provide the four S-IC switchpoints in a manner that would produce minimum impact on the present S-V configuration, two presently unused

3.1.3.4 (Continued)

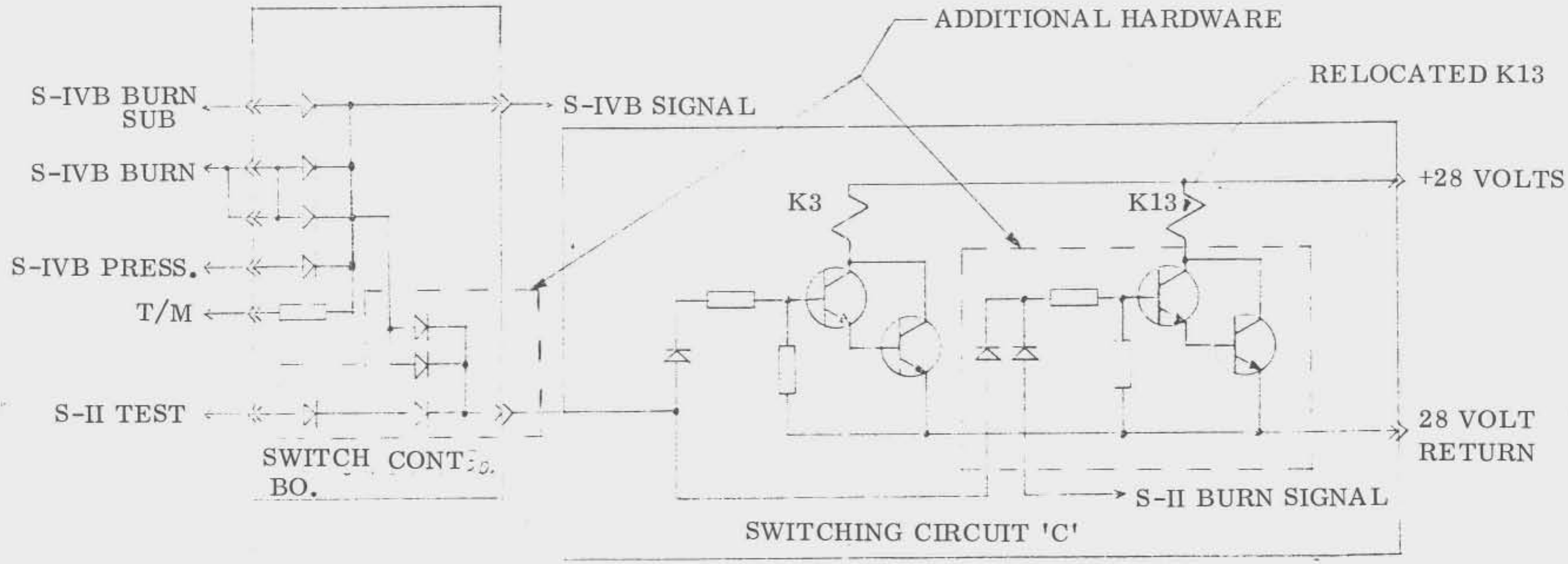
switchpoints will be utilized. The IU networks provide the FCC interface with nine switchpoints. The first six are presently used and the last three are terminated at the FCC interface. Therefore two of these will be routed to the S-IC filters. This will require four wires to be added to the FCC cable harness and Motherboards six and seven to be redesigned. All S-IC filters are located on Motherboards six and seven.

Since the present S-V configuration has an internal latching arrangement for the S-IC stage and only initiation of the S-II burn signal will release the latch, a redesign to the Switching Control Board and Switching Circuit 'C' will be required for the INT-20 configurations. The redesign will consist of diode-oring the S-IVB burn signal to two relays that presently release the latch. To insure that no S-II signal patch relays are energized by the S-IVB signal, one relay will be relocated on a new relay driver added to Switching Circuit 'C'. These changes are shown in Figure 3.1.3.4-1.

The above changes are sufficient to allow the present S-V FCC to command a four or five engine S-IC stage INT-20 mission. Additionally, the above changes will not impact a present S-V mission. The two engine S-IC stage INT-20 configuration, however, imposes an additional requirement on the FCC. The FCC output to each engine is derived from a Servo Amplifier. The two engine S-IC stage will require four Servo Amplifier outputs (two for yaw, two for pitch). However, the S-IVB burn portion of the INT-20 mission will require six Servo Amplifier outputs (triple redundant in yaw and pitch). To insure minimum transients at staging, all six outputs should be loaded during S-IC stage burn. This means dummy loads will be required on the two unused Servo Amplifiers for S-IC burn.

The present S-V configuration has eight Servo Amplifiers (four for yaw, four for pitch). It was assumed for this study that the two yaw, two pitch, four yaw, and four pitch outputs would be used during the S-IC burn portion of the two engine INT-20 mission. These were chosen because they are positioned diagonally opposite (see Figure 3.1.3.4-2), and require the minimum modification. If the above outputs drive the S-IC stage, then the one yaw, one pitch, three yaw and three pitch outputs require dummy loads.

Dummy loads are presently in the FCC for six of the eight Servo Amps. However, the S-IC burn signal opens relays in series with the loads for all six. Therefore, Switching Circuit 'A' will require redesign to block the S-IC stage signal from energizing the relays in series with the one yaw, one pitch, three yaw and three pitch dummy loads. This can be accomplished as shown in Figure 3.1.3.4-3. The switching has been arranged to where a



NOTE - ONLY AFFECTED CIRCUITRY SHOWN

FIGURE 3.1.3.4-1. FCC MODIFICATIONS

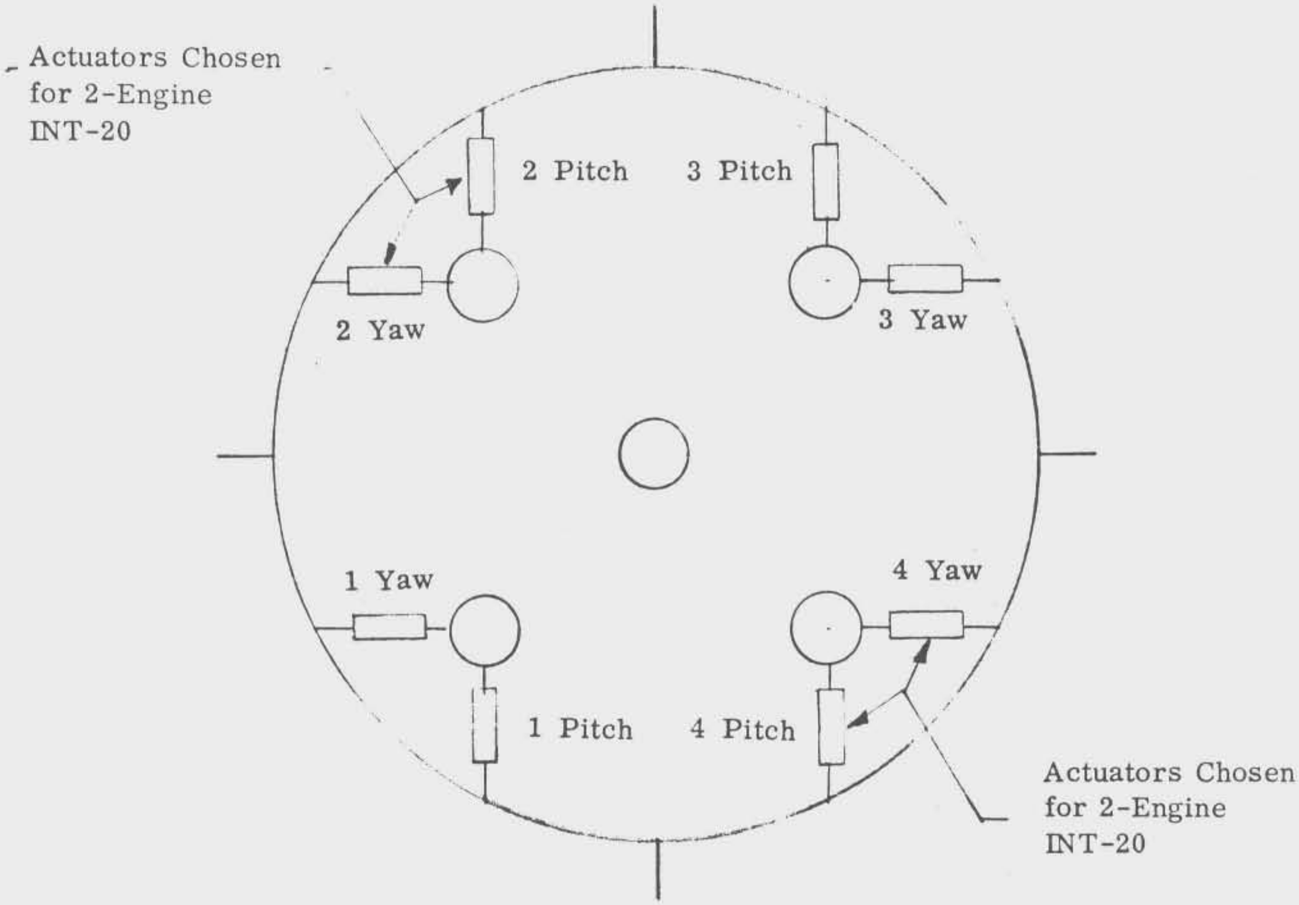


FIGURE 3.1.3.4-2. ACTUATOR PAIRS

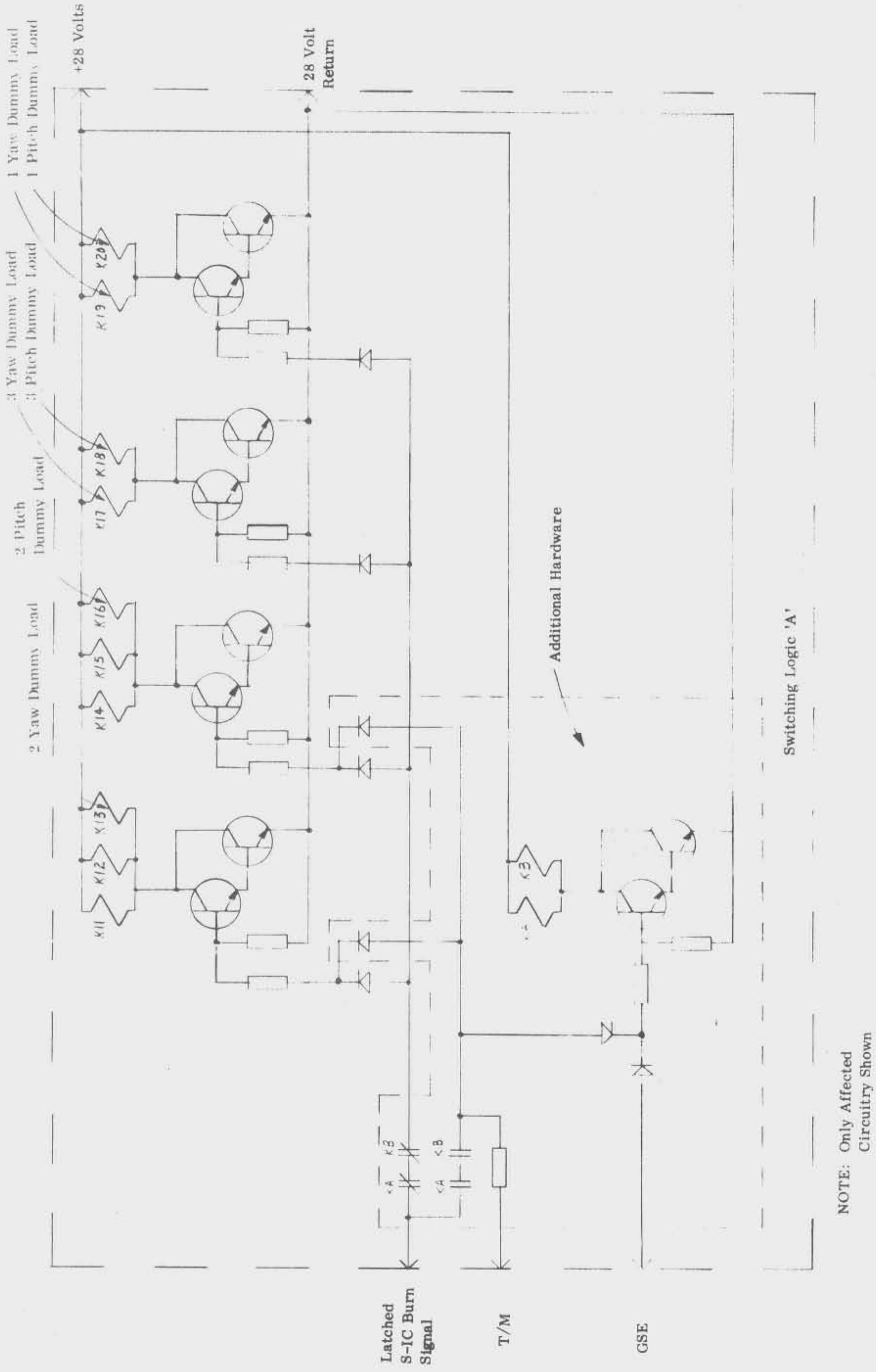


FIGURE 3.1.3.4-3. FCC/GSE MODIFICATION

3.1.3.4 (Continued)

GSE signal is required to initially energize the blocking relays but the normal S-IC burn signal, latched within the FCC, will maintain the blocking relays. The blocking relays will be de-energized by the unlatching of the S-IC signal.

A telemetry signal will be added as shown in Figure 3.1.3.4-3. This signal will verify that the FCC has received the GSE command and is configured for a two engine INT-20 mission.

c. Conclusions

An FCC, redesigned as described above, will be capable of commanding any of the three proposed INT-20 missions as well as the present S-V missions.

The fact that some unnecessary hardware (S-II filters, relays, etc.) is present in the FCC for an INT-20 mission was not discussed in the preceding section. It is recommended that this hardware remain in an FCC designated for an INT-20 mission for additional changes will be required for removal. These changes would severely impact the reversibility constraint placed on this study.

3.1.3.5 IU Environments

a. Acoustics

The five F-1 engine configuration of the INT-20 vehicle would apply acoustic pressures (PSI) on the IU approximately 12% more than a four engine configuration.

These increased pressure levels resulted in increasing the specified liftoff overall sound pressure level (OASPL) from 153.5 db to 154.5 db and the specified inflight OASPL from 155.0 to 156.0 db for the four to five engine configuration, respectively.

b. Vibration

The impact of the increased acoustic environment of the five engine configuration has increased the random vibration environment by 25% (PSD levels) which is a corresponding RMS acceleration intensity increase of 12%.

Due to the increased vibration environment, the projected flight random vibration at two IU locations (6 and 22) exceed the IN-P&VE-S-63-2 Random Specification. However, the Sinusoidal Specification (IN-P&VE-S-63-2) for these locations would exceed the random vibration peak excursions. This type of

3.1.3.5 (Continued)

comparison is an approximation and further analysis will have to be performed during Phase II of the INT-20 study.

At present reliability testing is being performed on IU components to vibration levels in excess of specified values. A comparison of the specified vibration environments and the reliability testing levels of the components at locations 6 and 22 (Figures 3.1.3.5-1 and -2) will be performed during Phase II study, to determine the impact of the higher vibratory environments.

The only other apparent problem area appears to be location 21 (ST-124 area). The increased acoustic and vibratory environments may cause malfunctioning of the ST-124 component. At present the ST-124 appears to be marginal with respect to higher vibration and acoustic environments.

A test program is presently in progress to evaluate the effectiveness of X-306 Damping Compound in the ST-124 area. An analysis of test data will be available in the near future which will indicate the vibration attenuation of X-306 Damping Compound. This data will be utilized in Phase II of the INT-20 study to determine the vibration input to the ST-124 and the vibratory effect on the component.

c. Acceleration

The INT-20 study has increased static acceleration requirements from 4.68 to 6.0 g's.

A summary of the components which are not qualified for 6 g's acceleration are shown in Table 3.1.3.5-I. These components are not presently qualified to this flight acceleration level and will require further analysis and possible requalification of questionable components.

d. Combined Vibration and Acceleration

Any more severe than the Saturn V combination of acceleration and vibration environments at the Max Q period of flight could be significant in determining the loads imposed on the structure and the operating condition of IU components at this flight time.

At the present phase of this study, Max Q data is not available to perform this analysis. Therefore, during Phase II of the INT-20 study this condition will be analyzed.

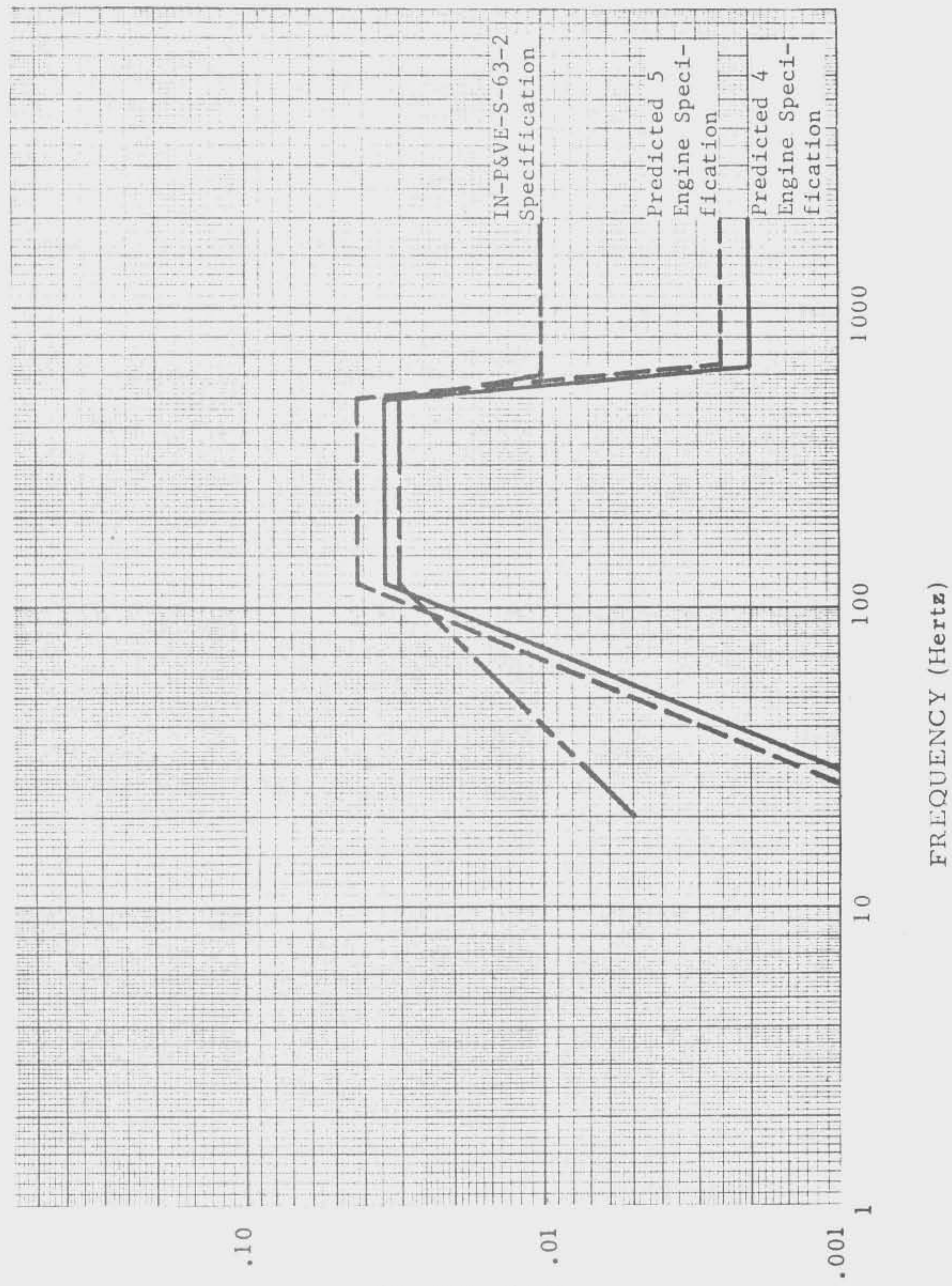


FIGURE 3.1.3.5-1. INT-20 STUDY IU LOCATION NUMBER 6

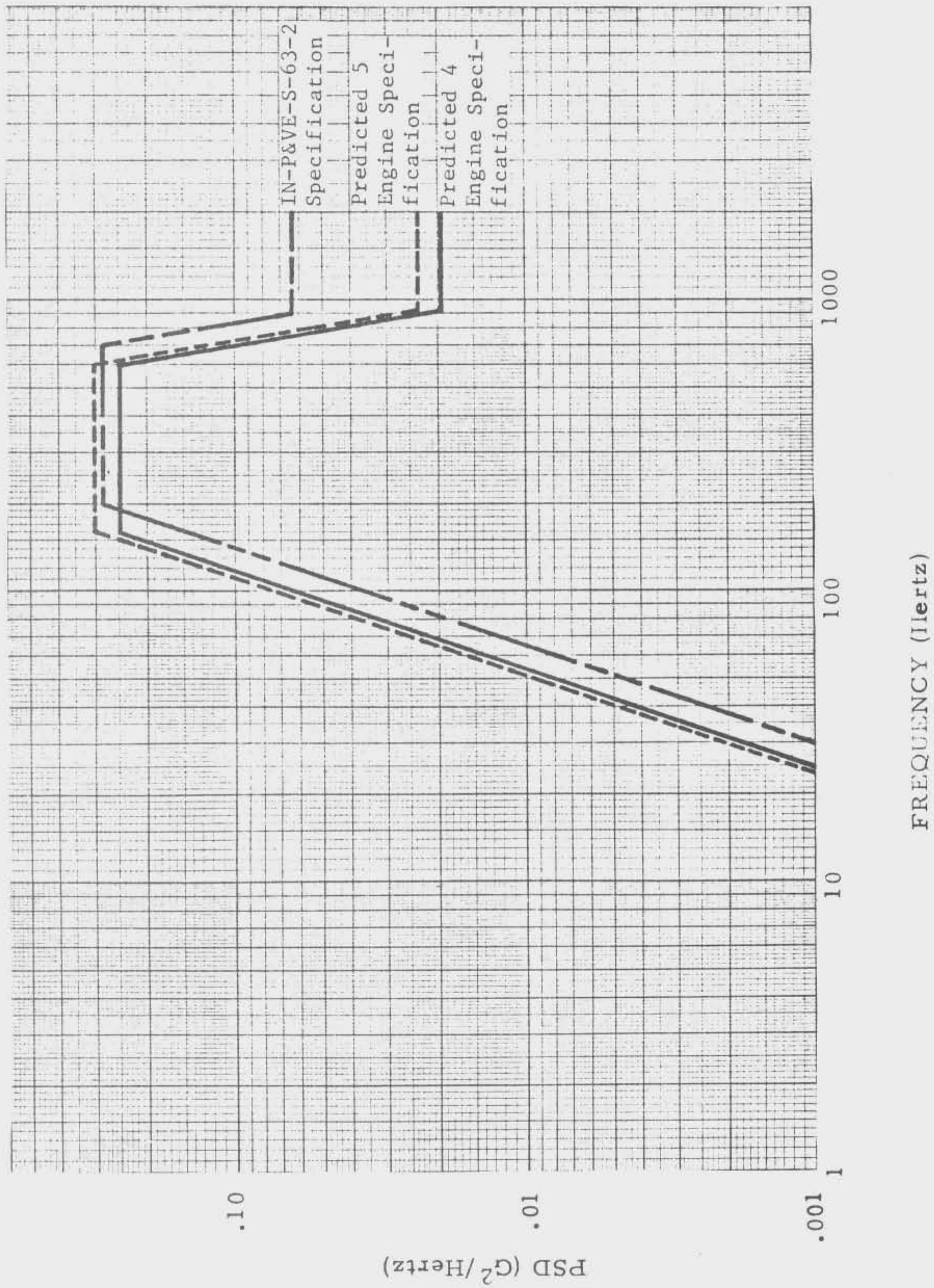


FIGURE 3.1.3.5-2. INT-20 STUDY IU LOCATION NUMBER 22

TABLE 3.1.3.5-I. COMPONENT ACCELERATION QUALIFICATION

COMPONENT	COMMENTS
Co-axial terminator assy	used only for ground checkout.
Electronic control assy	non-operational after 75 seconds flight time.
Thermistor	non-operational after 75 seconds flight time.
1000 PSI GBS switch assy	used only for ground checkout.
GBS panel assy	this panel assy was statically tested to loads in excess of 6 g (500 S-4 test).
First stage pres. regulator	tested to 5.3 g acceleration.
GN2 165 cu in storage sphere	not qualified but structurally capable of 6 g's.
GN_2 2 cu ft storage sphere	not qualified but structurally capable of 6 g's.
20 mu filter assy	not qualified but structurally capable of 6 g's.
GBS pres. regulator	qualified to 5 g's acceleration.
Thermal cond. panel	This panel with components has been statically tested to loads in excess of 6 g's. (500 S-4 test)
GB heat exchanger	qualified to 5.3 g's.
Bleeder assy	not qualified but structurally capable of 6 g's.
Hose assy, Flex IU	not qualified but structurally capable of 6 g's.
Orifice assy	not qualified but structurally capable of 6 g's.
GB solenoid valve assy	used only for ground checkout.
PCM co-axial switch	not qualified.
Ring Hybrid (CCS)	not used on IU 505 and subs.

NOTE:

The listed IU components were not qualified to 6 g's static acceleration in the flight axis.

3.1.3.5 (Continued)

e. Other Engine Configurations

The four engine configuration was discussed in Volume VII (November 13, 1967, "Selected Vehicle Configuration MLV-SAT-INT-20"). No apparent problems existed for this configuration and it is felt that a three engine configuration would result in a less severe dynamic environment during liftoff.

However, the inflight environment for the three and four engine configuration could pose some problems which cannot be determined until dynamic pressures, acceleration, payload configuration and payload weights are defined. This will be analyzed in Phase II of the study.

3.1.3.6 IU Structure

a. Configuration and Design Features

This section contains the IU structure and configuration analysis performed in the trades study of the INT-20 vehicle configurations of Figure 3.1.3.6-1. This effort consisted of:

Summarizing the present IU structural capability.

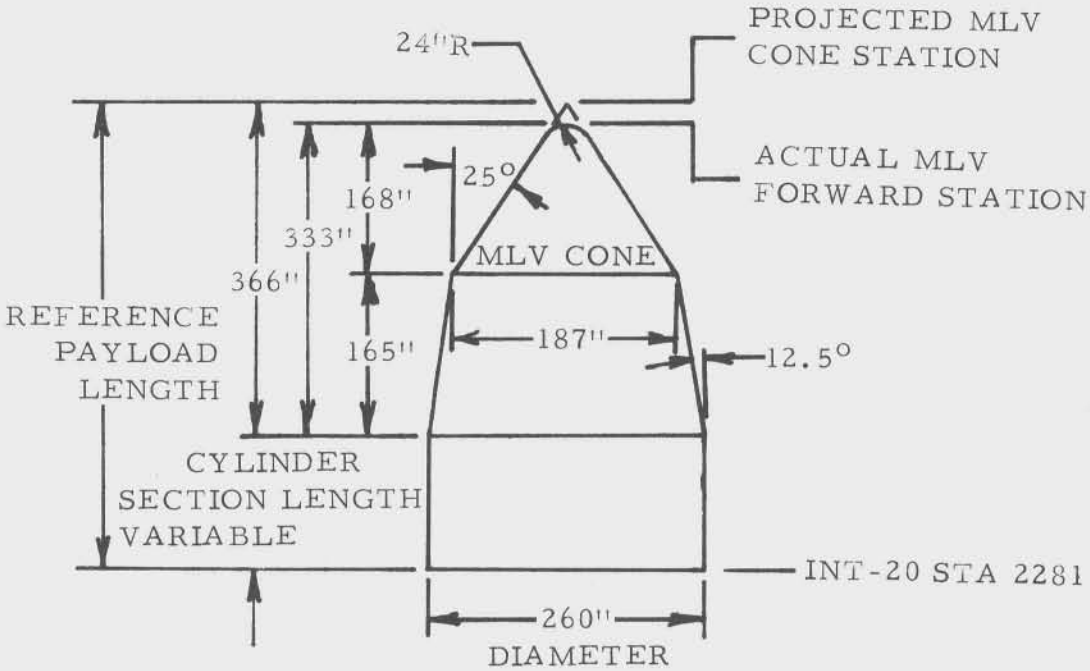
Determining the maximum IU loads and environmental conditions for the INT-20 vehicle.

Analyzing the IU structure using the maximum loading and environmental conditions at critical IU structural areas establishing the present structural qualification status and any design changes required.

Making feasibility studies of installation (placement) of added or changed IU components.

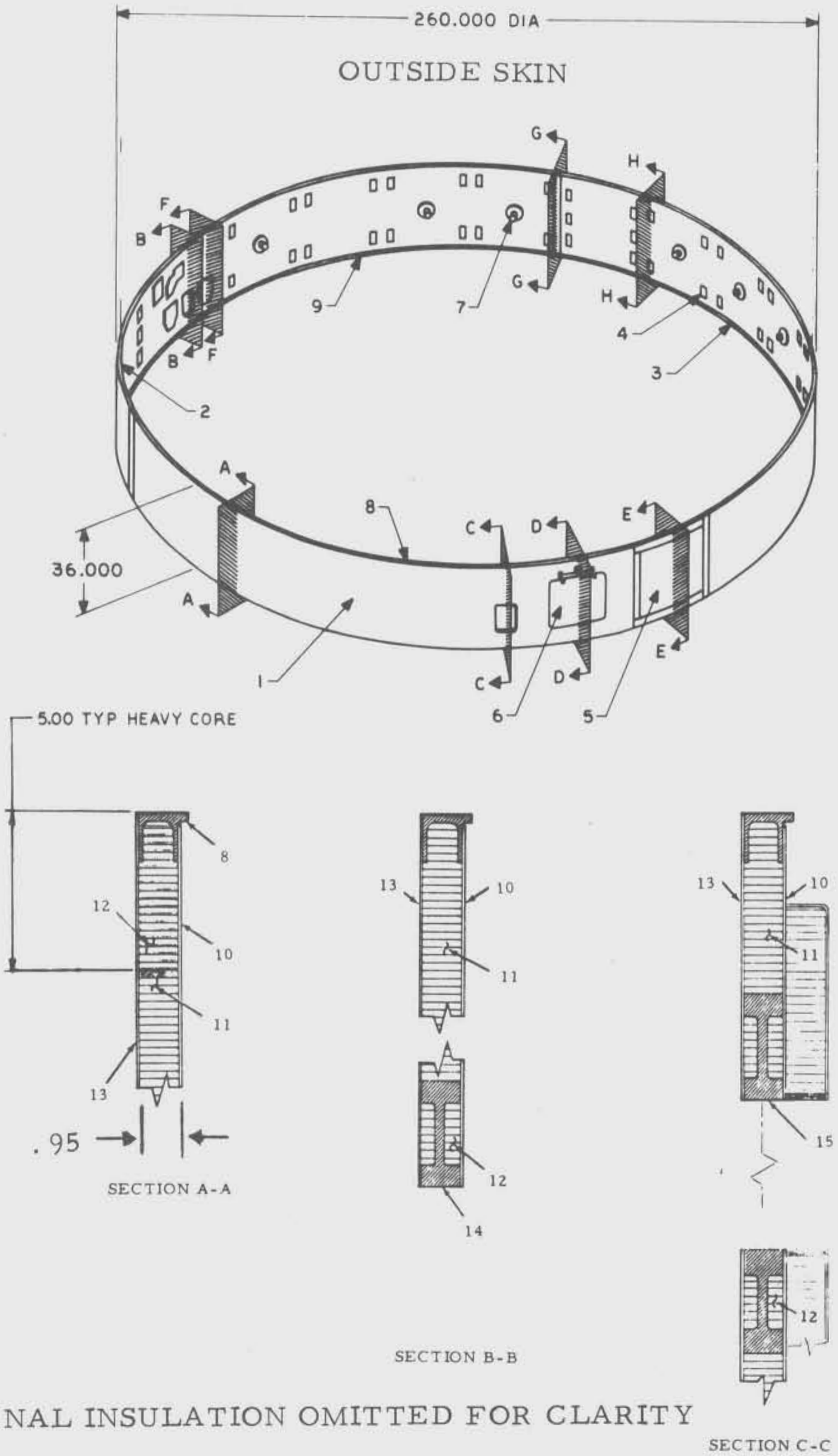
Estimating the IU weight for the new vehicle.

The present IU structure is defined as the 30Z13100-1 Structural Assembly. Principal features of this structure are depicted in Figures 3.1.3.6-2, -3, and -4. The IU structure is a cylindrical structure 260 inches in diameter and 36 inches high. The cylindrical structure consists of honeycomb sandwich construction 0.95 inch thick with upper and lower interface channel rings. It provides various pads, brackets, and inserts for component mounting, cutouts for antenna cables, an ST-124 viewport, Environmental Control System (ECS) panel, interface bolt access cutouts, an umbilical connection and a load carrying access door.



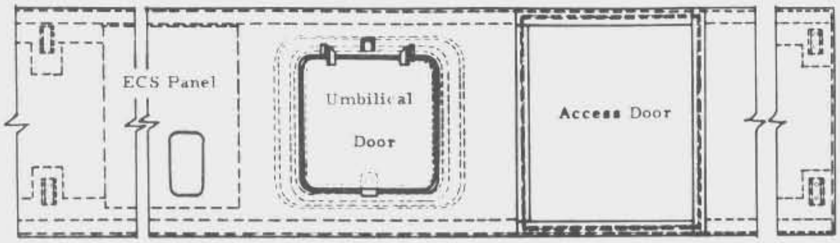
PAYLOAD CONFIGURATION			
Cylinder Configuration	Reference Payload Length		Projected MLV Cone Station
	(IN)	(FT)	
C	366	30.5	2647
B	600	50	2881
A	840	70	3121
D	1200	100	3481

FIGURE 3.1.3.6-1. INT-20 PAYLOAD CONFIGURATION ALTERNATIVES

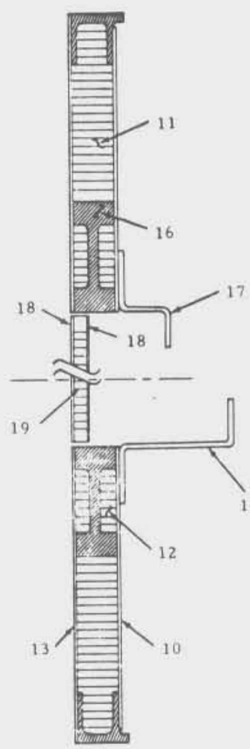


NOTE: EXTERNAL INSULATION OMITTED FOR CLARITY

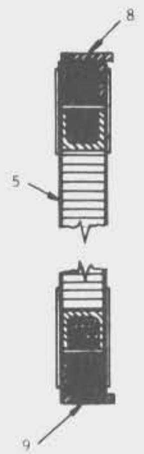
FIGURE 3.1.3.6-2. IU STRUCTURAL CONFIGURATION



View of Umbilical Door Area



SECTION D-D



SECTION E-E

Index No.	IBM Drawing No.	Description	Gage	Material	Material Specification
1	30Z13101	Segment Assembly		Honeycomb Construction	
2	30Z13103	Segment Assembly		Honeycomb Construction	
3	30Z13102	Segment Assembly		Honeycomb Construction	
4	30Z13004	Bracket (Weldment)		Honeycomb Construction	
5	30Z13109	Door Assembly		Honeycomb Construction	
6	30Z12008	Door Assembly		Honeycomb Construction	
7	30Z12050	Bracket		7075-T6	
8	30Z13110	Ring		7075-T6	QQ-A-277
9	30Z13111	Ring		7075-T6	QQ-A-277
10	30Z13105-7	Skin	0.020	7075-T6	QQ-A-283
11	30Z13031	Core	3.1 Lbs per Cubic Foot	Aluminum Honeycomb	MIL-C-7438
12	30Z13030	Core	8.1 Lbs per Cubic Foot	Aluminum Honeycomb	MIL-C-7438
13	30Z13105-5	Skin	0.030	7075-T6	QQ-A-283
14	30Z13059	Frame Assembly		7075-T6	QQ-A-283
15	30Z13037	Frame Assembly		7075-T6	QQ-A-283
16	30Z13021	Frame Assembly		7075-T6	QQ-A-283
17	30Z13005	Bracket		Reinforced Plastic	MIL-P-25421 Type II, Class I
18	30Z13105-6	Skin	0.015	7075-T651	QQ-A-283
19	30Z13035	Core	4.3 Lbs per Cubic Foot	Aluminum Honeycomb	MIL-C-7438
20	30Z13124	Filler Material		Polystyrene Beads	

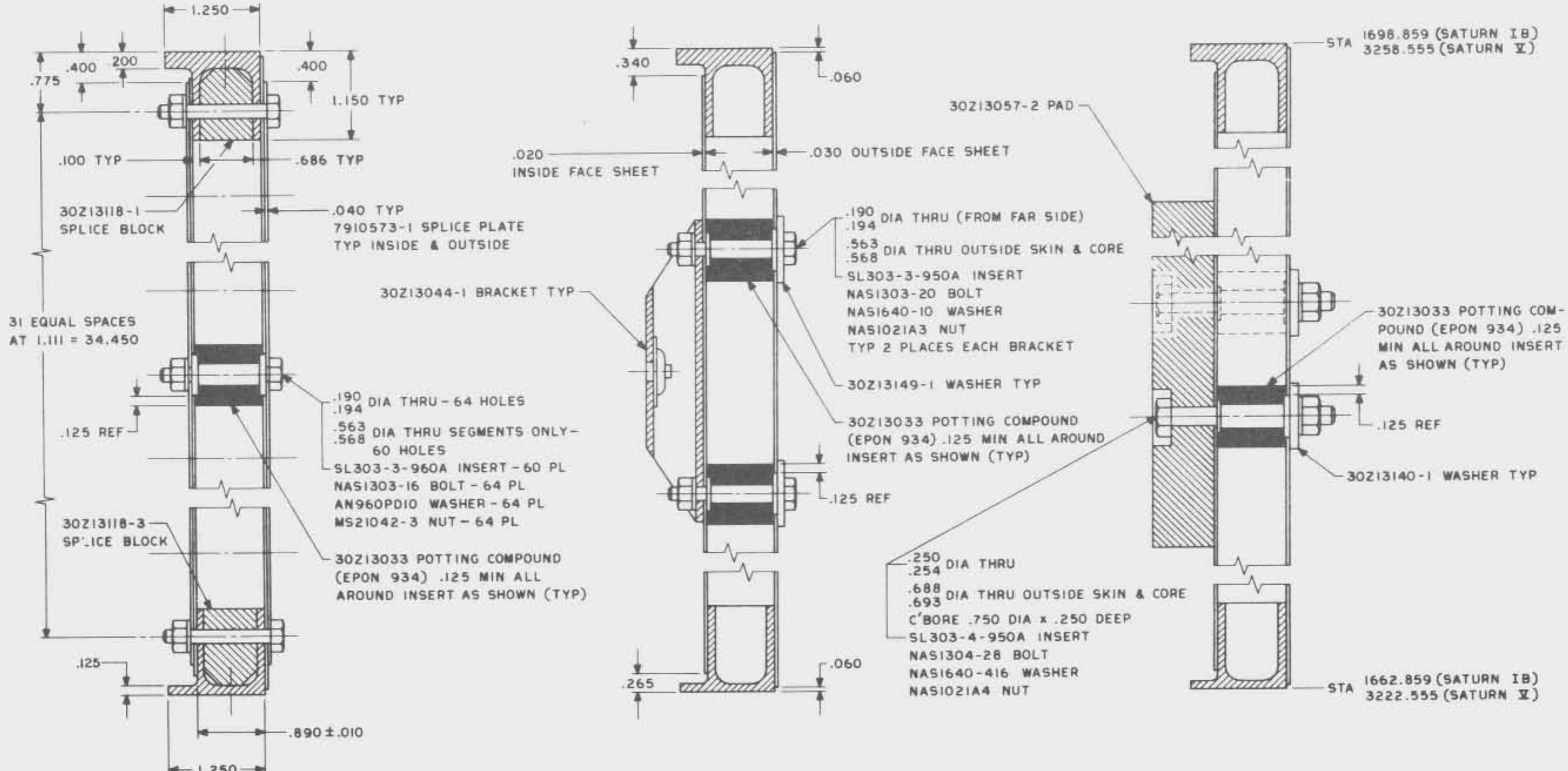
NOTES:

1. ADHESIVE BETWEEN METAL-TO-METAL AND METAL-TO-CORE IS NARMCO METLBOND 329.

2. ADHESIVE BETWEEN CORES IS EPO-CAST H-1310 MOD I.

3. INSERT INSTALLATION IS MADE WITH EPON 934 (SECTION F-F).

FIGURE 3.1.3.6-3. IU STRUCTURAL CONFIGURATION



SECTION G-G — CROSS SECTIONAL VIEW OF SEGMENT SPLICE PLATE INSTALLATION

SECTION H-H — CROSS SECTIONAL VIEW OF TYPICAL COLD PLATE (THERMAL CONDITIONING PANEL) BRACKET INSTALLATION

SECTION F-F — CROSS SECTIONAL VIEW OF THE LOWER RIGHT STABLE PLATFORM (ST-124) MOUNTING PAD INSTALLATION

FIGURE 3.1.3.6-4. IU STRUCTURAL CONFIGURATION

3.1.3.6 (Continued)

The honeycomb sandwich construction consists of 7075-T6 aluminum alloy face sheets bonded with METLBOND 329 adhesive to 3.1 or 8.1 pound per cubic foot core. EPOCAST H-1310, Mod. 1, is used to adhesively splice the core. The brackets and pads, bonded with METLBOND 329 and EPON 934, are in most cases also bolted to the basic honeycomb structure.

Certain salient features which must be carefully considered when evaluating the IU structure for new vehicle missions and configurations are:

The access door is load carrying and must be capable of being removed and reinstalled any time prior to flight. Also, the structure must be capable of sustaining the vehicle ground loads when the door is removed.

The 8.1 pound per cubic foot core is used to redistribute loads imposed by bracket, pad and mounting ring structural elements. Therefore, certain component additions or changes could require redesign of the core pattern prior to bonding or other local rework after bonding.

The structural buckling considerations must include not only the basic vehicle shell loads, but also the lateral loads imposed by the IU components attached to the basic honeycomb structure. The lateral loads are intensified by the dynamic environment imposed on the component and attachment dynamic response. Because of these complexities, structural capability values of the IU shell are subject to minor variations.

Hardpointing effects of stiffeners, rigid frames, etc., in structures above or below increase the equivalent running load per inch at the IU interfaces.

Any changes to interface hole patterns on adjoining stages from that currently used on the Saturn program would impact the IU.

The present IU configuration (installation and assembly criteria) is defined by the 10Z22501-1 drawing, Instrument Unit Assembly. This assembly defines each IU configuration by drawing revision level as required by the 10 IU system requirements (Navigation and Guidance, Attitude Control, Sequencing, Measurement and Telemetry, Radio and Command, Tracking, Power and Distribution, Emergency Detection, Environmental Control, and Structural) combined with alignment, interface control documentation, and parameters affecting component mounting surface requirements.

3.1.3.6 (Continued)

This assembly also defines the component mounting hardware, component location, and the IU stage total weight.

The only major structural differences between the Saturn IB Instrument Units (200 series) and the Saturn V Instrument Units (500 series) are that the Saturn V IU's have the following:

Cork insulation cold bonded on the IU outer skin except for the umbilical door, splice plates, and proturberance covers. The insulation is defined as the 7916352-1 Installation of Thermal Insulation. This insulation significantly increases the load carrying capability of the IU shell structure under End Boost loads by reducing the inflight temperature.

A pad of X-306 damping compound cold bonded to the outer skin in the ST-124 area (Position IV), and replaces steel channels previously used for the Saturn V IU configuration. This installation is defined as 7916344-1 Vibration Damping Pad. The damping compound is more effective in reducing local vibration induced loads than the previous damping system.

Antenna mounting provision.

By inspection, the Saturn V Instrument Units (500 series) will have a higher structural load carrying capability compared to the Saturn IB Instrument Units (200 series), and will be used in the current study. The cork or the damping compound, or both, might be found to be unnecessary in later studies, dependent upon the baseline vehicle selected.

b. Loads

The vehicle loads, environmental criteria and structural design criteria for the IU structure are defined in the following sections. Generally, the vehicle loads and environmental criteria were obtained from IBM analytical predictions and Boeing supplied reports.

1. Structural Design Criteria

The criteria is identical to that presently specified by NASA. The following safety factors are applicable to the IU structural design as minimum values.

3.1.3.6 (Continued)

Yield Load	= 1.1 times limit load
Ultimate Load	= 1.25 times limit load (unmanned flight) = 1.4 times limit load (manned flight)

2. Vehicle Environment and Loads

The preliminary vehicle loads and associated environments were obtained from a number of sources. The resulting loads for the various configurations/payloads are summarized in the following tables.

95% Ground Wind	Table 3.1.3.6-I
99.9% Ground Wind	Table 3.1.3.6-II
Launch	Not addressed in this study; definition of IU equipment loads required.
Max Q Alpha (30.5 ft Payload)	Table 3.1.3.6-III
Max Q Alpha (50 ft Payload)	Table 3.1.3.6-IV
End Boost g = 4.18	Table 3.1.3.6-V
g = 5.14	Table 3.1.3.6-VI
g = 6.00	Table 3.1.3.6-VII
Engine out S-IC Stage Separation	Omitted by oral direction of Boeing for this preliminary study.

These tables include conversion to maximum compression and tension running loads for specified factors of safety.

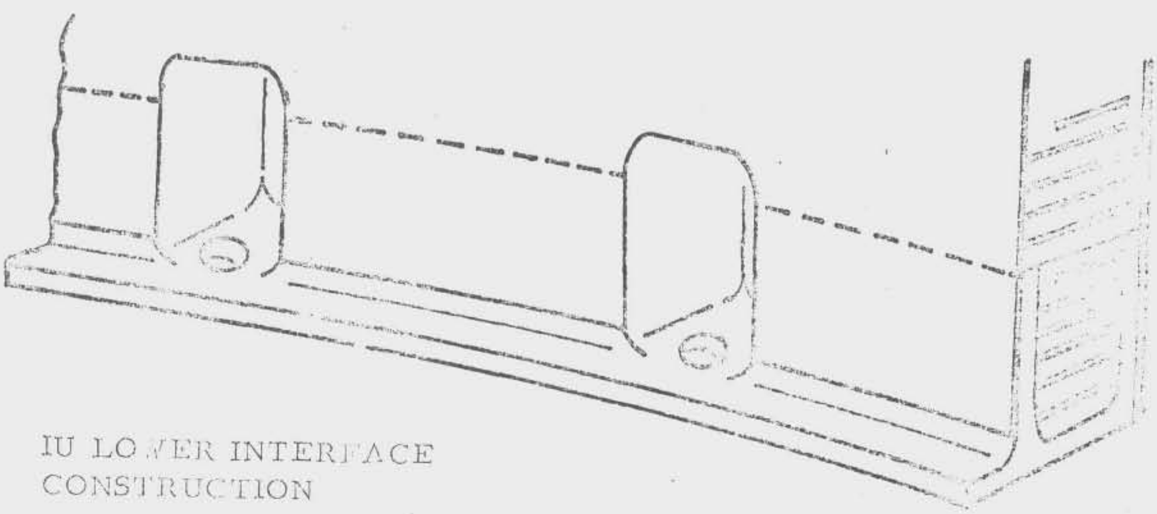
The maximum tension loads in these tables do not consider the delta pressure across the IU structure. This delta pressure consideration should add less than 26 lbs per inch to the limit tension load.

The unprotected IU structure maximum outer skin temperature with no insulation is estimated as approximately 210°F based upon the previous INT-20 study, pending further studies of aerodynamic heating.

TABLE 3.1.3.6-I. PRELIMINARY 95% GROUND WIND LOADS INT-20 STATION 2245

Payload Configuration		Limit Loads Station 2245		F.S. = 1.0	
Code	Payload Above IU (KIPS)	Axial Load (KIPS)	Bending Moment (In-Lbs x 10^{-6})	N_c (Lbs/In)	N_t (Lbs/In)
C	70	74.3	.3	96	N/A
	100	104.3	.3	134	N/A
	130	134.3	.3	171	N/A
	160	164.3	.3	208	N/A
B	70	74.3	1.1	111	N/A
	100	104.3	1.1	149	N/A
	130	134.3	1.1	186	N/A
	160	164.3	1.1	223	N/A
A	70	74.3	2.5	140	N/A
	100	104.3	2.5	178	N/A
	130	134.3	2.5	215	N/A
	160	164.3	2.5	252	N/A
D	70	74.3	5.7	197	17
	100	104.3	5.7	235	N/A
	130	134.3	5.7	272	N/A
	160	164.3	5.7	309	N/A

NOTES: 1) Number of S-IC engines do not affect above loads.
2) N_c is the actual compression running load.
3) N_t is the actual tension running load.
KIPS = lb x 10^3

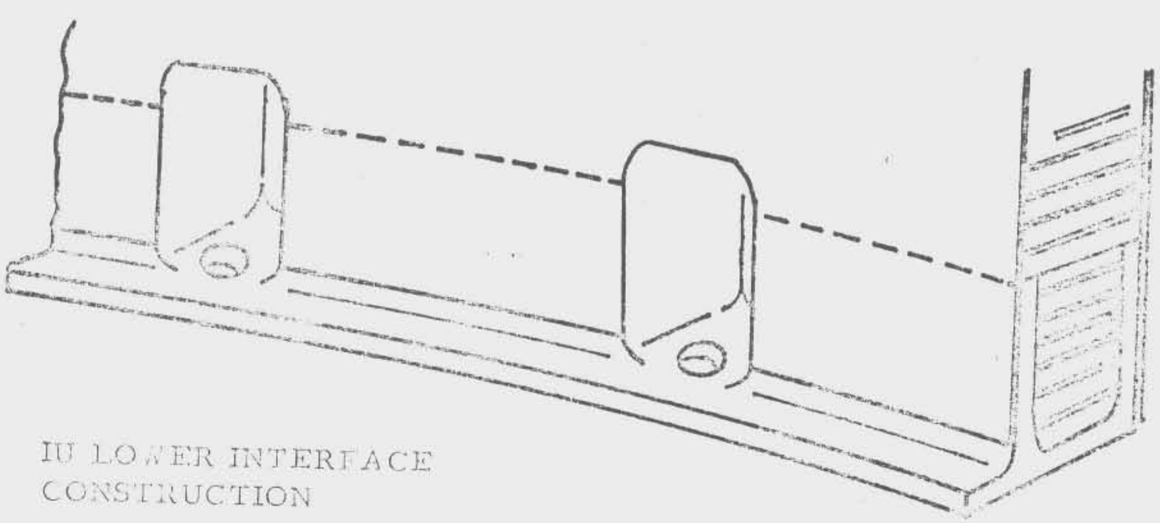


IU LOWER INTERFACE CONSTRUCTION

TABLE 3.1.3.6-II. PRELIMINARY 99.9% GROUND WIND LOADS INT-20 STATION 2245

Payload Configuration		Limit Loads Station 2245		F. S. = 1.4	
Code	Payload Above IU (KIPS)	Axial Load (KIPS)	Bending Moment (In-Lbs x 10^{-6})	N_c (Lbs/In)	N_t (Lbs/In)
C	70	74.3	.9	150	N/A
	100	104.3	.9	203	N/A
	130	134.3	.9	253	N/A
	160	164.3	.9	306	N/A
B	70	74.3	3.1	207	N/A
	100	104.3	3.1	260	N/A
	130	134.3	3.1	312	N/A
	160	164.3	3.1	364	N/A
A	70	74.3	6.9	306	55
	100	104.3	6.9	360	1
	130	134.3	6.9	411	N/A
	160	164.3	6.9	467	N/A
D	70	74.3	15.6	536	284
	100	104.3	15.6	590	231
	130	134.3	15.6	640	179
	160	164.3	15.6	693	127

NOTES: 1) Number of S-IC engines do not affect above loads.
2) N_c is the actual compression running load.
3) N_t is the actual tension running load.

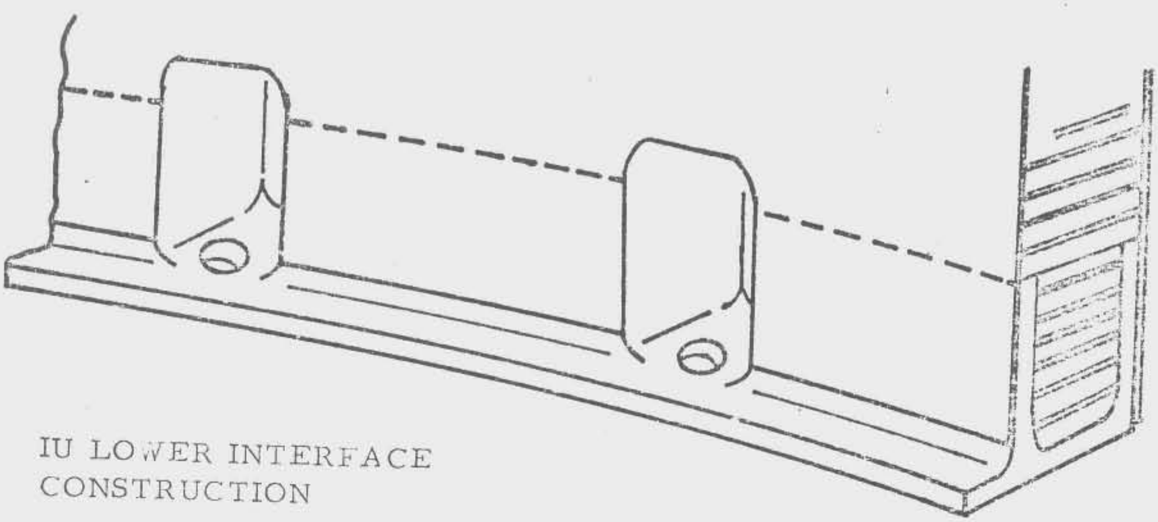


IU LOWER INTERFACE CONSTRUCTION

TABLE 3.1.3.6-III. PRELIMINARY MAX Q ALPHA LOADS
INT-20 STATION 2245
PAYLOAD HEIGHT 30.5 FEET

		Limit Loads		Ultimate Running Loads			
				F.S. = 1.25		F.S. = 1.4	
Number of Engines	Payload Above IU (KIPS)	Axial Load (KIPS)	Bending Moment (In-Lbs x 10^{-6})	N_c (Lbs/In)	N_t (Lbs/In)	N_c (Lbs/In)	N_t (Lbs/In)
2	70	215.0	15.0	683	22	764	25
	100	269.0	15.0	765	N/A	856	N/A
	130	324.1	15.0	850	↑	910	↑
	160	378.7	15.0	934		1048	
3	70	231.9	15.0	708		792	
	100	288.3	15.0	795		890	
	130	344.8	15.0	881		986	
	160	401.2	15.0	966		1081	
4	100	290.9	15.0	797		892	
	130	346.7	15.0	884		990	
	160	402.6	15.0	970		1088	
5	100	319.9	15.0	841		941	
	130	379.0	15.0	934	↓	1048	↓
	160	438.1	15.0	1022	N/A	1146	N/A

NOTES:
1. Payload height above IU 30.5 feet (for Bending Moment determinations).
2. Number of engines assumed not to affect Max Q Alpha Bending Moment.
3. N_c is the compression running load.
4. N_t is the tension running load.

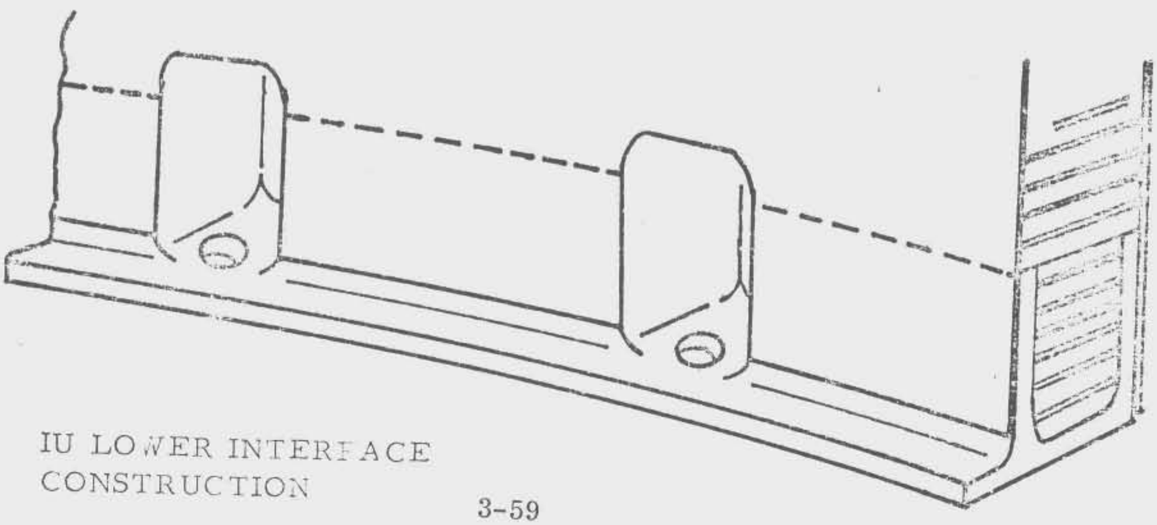


IU LOWER INTERFACE
CONSTRUCTION

TABLE 3.1.3.6-IV. PRELIMINARY MAX Q ALPHA LOADS
INT-20 STATION 2245
PAYLOAD HEIGHT 50 FEET

		Limit Loads		Ultimate Running Load			
				F. S. = 1.25		F. S. = 1.4	
Number of Engines	Payload Above IU (KIPS)	Axial Load (KIPS)	Bending Moment (In-Lbs $\times 10^{-6}$)	N_c (Lbs/In)	N_t (Lbs/In)	N_c (Lbs/In)	N_t (Lbs/In)
2	70	215.0	45.0	1389	727	1552	816
	100	269.6	43.1	1425	600	1598	671
	130	324.1	42.5	1495	501	1673	561
	160	378.7	44.4	1625	462	1820	518
3	70	231.9	45.0	1411	703	1581	787
	100	288.3	43.1	1458	570	1631	639
	130	344.8	42.5	1530	470	1712	526
	160	401.2	44.4	1658	430	1856	481
4	100	290.9	43.1	1458	566	1631	635
	130	346.7	42.5	1530	467	1715	524
	160	402.6	44.4	1660	426	1860	477
5	100	319.9	43.1	1502	521	1683	585
	130	379.0	42.5	1581	417	1770	467
	160	438.1	44.4	1715	374	1921	419

NOTES:
1. Assumed Payload Height above IU 50 feet (for Bending Moment determinations)
2. Number of engines assumed not to affect Max Q Alpha Bending Moment
3. N_c is the compression running load.
4. N_t is the tension running load.

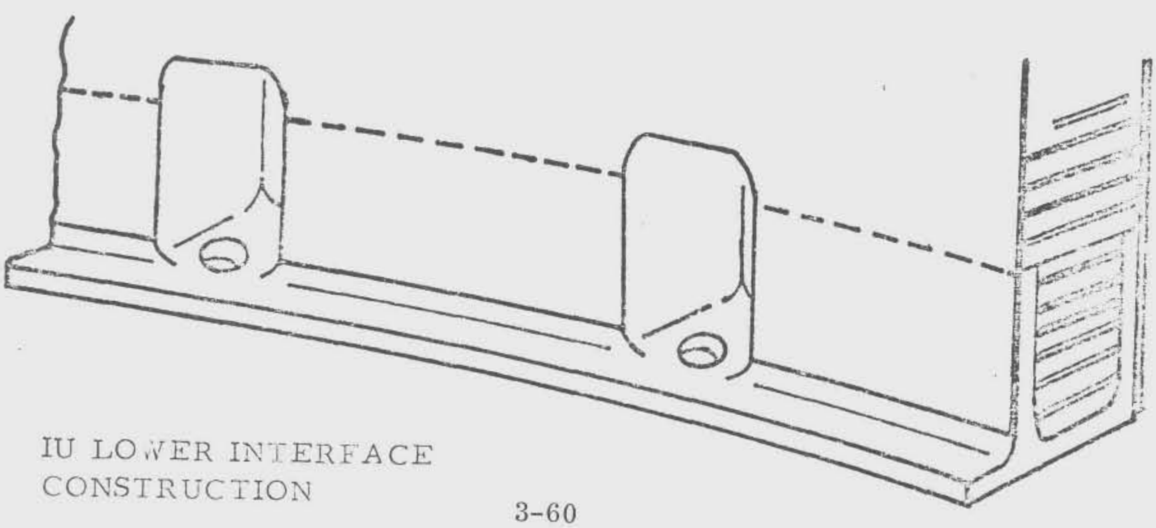


IU LOWER INTERFACE CONSTRUCTION

TABLE 3.1.3.6-V. PRELIMINARY END BOOST LOADS
INT-20 STATION 2245
g= 4.28

Number of Engines	Payload Above IU (KIPS)	Limit Loads		Nc	
		Axial Load (KIPS)	Bending Moment ($In-Lbs \times 10^{-6}$)	F.S. 1.25	F.S. 1.40
2	70	316.72	1.48	520	582
	100	445.12	2.08	731	819
	130	573.52	2.68	940	1052
	160	701.92	3.28	1152	1292
3	70	316.72	2.22	537	601
	100	445.12	3.12	756	847
	130	573.52	4.02	970	1088
	160	701.92	4.92	1190	1333
4	100	445.12	4.17	779	872
	130	573.52	5.37	1003	1124
	160	701.92	6.57	1230	1378
5	100	445.12	5.21	805	901
	130	573.52	6.71	1037	1160
	160	701.92	8.21	1270	1421

NOTES: 1. Payload configuration height assumed to be 50 feet.
2. N_c is the compression running load.
3. F.S. is the factor of safety.

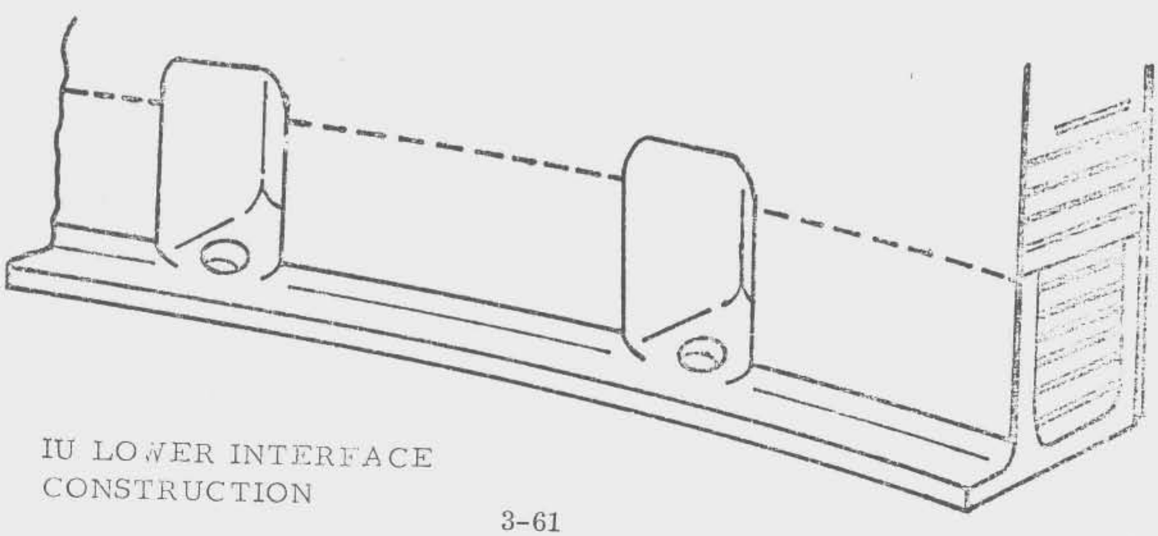


IU LOWER INTERFACE CONSTRUCTION

TABLE 3.1.3.6-VI. PRELIMINARY END BOOST LOADS
INT-20 STATION 2245
$g = 5.14$

Number of Engines	Payload Above IU (KIPS)	Limit Loads		N_c	
		Axial Load (KIPS)	Bending Moment (In-Lbs x 10^{-6})	F.S. 1.25	F.S. 1.40
2	70	380.36	1.48	617	690
	100	534.56	2.08	869	973
	130	688.76	2.68	1119	1251
	160	842.96	3.28	1370	1532
3	70	380.36	2.22	635	710
	100	534.56	3.12	894	1001
	130	688.76	4.02	1149	1288
	160	842.96	4.92	1409	1575
4	100	534.56	4.17	916	1029
	130	688.76	5.37	1181	1322
	160	842.96	6.57	1445	1620
5	100	534.56	5.21	941	1056
	130	688.76	6.71	1212	1360
	160	842.96	8.21	1482	1660

Notes:
1. Payload configuration height assumed to be 50 feet.
2. Nc is the compression running load.
3. F.S. is the factor of safety.

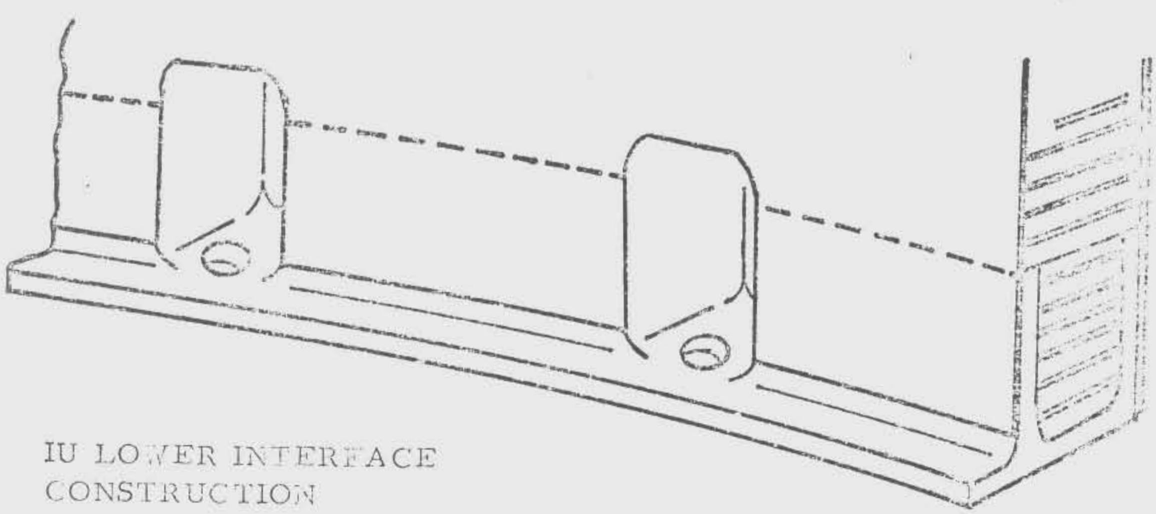


IU LOWER INTERFACE
CONSTRUCTION

TABLE 3.1.3.6-VII. PRELIMINARY END BOOST LOADS
INT-20 STATION 2245
g = 6.00

Number of Engines	Payload Above IU (KIPS)	Limit Loads Axial Load (KIPS)	Limit Loads Bending Moment (In-Lbs x 10^{-6})	N_c F.S. 1.25	N_c F.S. 1.40
2	70	444.0	1.48	715	800
	100	624.0	2.08	1003	1024
	130	804.0	2.68	1292	1450
	160	984.0	3.28	1589	1779
3	70	444.0	2.22	732	820
	100	624.0	3.12	1030	1151
	130	804.0	4.02	1328	1488
	160	984.0	4.92	1625	1820
4	100	624.0	4.17	1052	1180
	130	804.0	5.37	1358	1520
	160	984.0	6.57	1667	1864
5	100	624.0	5.21	1079	1209
	130	804.0	6.71	1391	1560
	160	984.0	8.21	1703	1910

NOTES: 1. Payload configuration height assumed to be 50 feet.
2. N_c is the compression running load.
3. F.S. is the factor of safety.



IU LOWER INTERFACE
CONSTRUCTION

TABLE 3.1.3.6-VIII. TRADE IMPACTS
INT-20 STUDY CONFIGURATIONS

			INT-20 CONFIGURATION LENGTH							
			C		B		A		D	
Number of Engines	Payload (KIPS)	End Boost g's	F.S. 1.25	F.S. 1.40	F.S. 1.25	F.S. 1.40	F.S. 1.25	F.S. 1.40	F.S. 1.25	F.S. 1.40
2	100	4.28	0	0	2	1,2	1,2	1,2,3	1,2,3	1,2,3
		6.00	0	0	2	1,2	1,2	1,2,3	1,2,3	1,2,3
	130	4.28	0	0	1	1,2	1,2,3	1,2,3	1,2,3	1,2,3
		6.00	0	1	1	1,2,3	1,2,3	1,2,3	1,2,3	1,2,3
	160	4.28	0	0	1	1,2,3	1,2,3	1,2,3	1,2,3	1,2,3
		6.00	1	1	1	1,2,3	1,2,3	1,2,3	1,2,3	1,2,3
3	100	4.28	0	0	1	1,2	1,2	1,2,3	1,2,3	1,2,3
		6.00	0	0	1	1,2	1,2	1,2,3	1,2,3	1,2,3
	130	4.28	0	0	1	1	1,2,3	1,2,3	1,2,3	1,2,3
		6.00	0	1	1	1	1,2,3	1,2,3	1,2,3	1,2,3
	160	4.28	0	0	1	1,3	1,3	1,2,3	1,2,3	1,2,3
		6.00	1	1	1	1,3	1,3	1,2,3	1,2,3	1,2,3
4	100	4.28	0	0	1,2	1,2	1,2	1,2,3	1,2,3	1,2,3
		6.00	0	0	1,2	1,2	1,2	1,2,3	1,2,3	1,2,3
	130	4.28	0	0	1	1,2	1,2,3	1,2,3	1,2,3	1,2,3
		6.00	0	1	1	1,2	1,2,3	1,2,3	1,2,3	1,2,3
	160	4.28	0	0	1	1,3	1,3	1,2,3	1,2,3	1,2,3
		6.00	1	1	1	1,3	1,3	1,2,3	1,2,3	1,2,3
5	100	4.28	0	0	1,2	1,2	1,2	1,2,3	1,2,3	1,2,3
		6.00	0	0	1,2	1,2	1,2	1,2,3	1,2,3	1,2,3
	130	4.28	0	0	1	1	1,3	1,3	1,2,3	1,2,3
		6.00	1	1	1	1	1,3	1,3	1,2,3	1,2,3
	160	4.28	0	1	1	1,3	1,3	1,3	1,3	1,2,3
		6.00	1	1	1	1,3	1,3	1,3	1,3	1,2,3

See next page for code used.

CODES FOR TRADE IMPACT TABLE

0 IU structure is adequate without modification.

1 IU structure must be modified for increased compression running load capability.

2 IU structure must be modified for increased tension running load capability.

3 IU access door area requires considerable modification from the present load bearing design to a non-load bearing concept.

4 IU Equipment Arrangement/Environment

No changes for this factor are assumed until acoustic/dynamic loads are determined in the preliminary design phase. These loads, if critical, could impact code Items 1 and 2.

5 IU Umbilical Plate Area

No changes for this factor are assumed until umbilical loads are determined in the preliminary design phase.

Values in the table represent trends which may change for borderline cases due to refinement of structural loads which will be determined in the preliminary design phase.

It is assumed S-IVB boost loads are not critical for IU structural considerations.

3.1.3.6 (Continued)

Since the payload is not presently defined, the load distribution on the IU structure was considered uniform. This does not allow the consideration of load concentrations from payload hard points or load application points such as the LEM load concentrations presently considered in the Uprated Saturn I and Saturn V IU structural design.

The IU structure, besides providing a load transfer path between adjacent stages, also provides a component mounting area for guidance and control, telemetry, supporting ECS, and other subsystems. The determination of component equipment loads to assess the capability of the IU structure requires definition of the acoustic/vibration environment, stiffness/transmissibility characteristics, and the arrangement of the component equipment. The determination of static and static equivalent dynamic equipment loads resulting from these factors to determine IU structure impact are not addressed in this study, but are assumed of the same order of influence as for the Saturn V/Apollo configuration in determining structural capability comparisons in the next section.

Umbilical plate loads on the IU structure due to the change of launch tower interface location to connect with external electrical, fluid, and gaseous systems during ground checkout and launch operations are not defined in this trades study, and they will be addressed during the preliminary design phase.

Since the design criteria for the IU specifies a minimum weight structure, no handling and transportation loads were considered critical in the original design.

Consequently, handling and transportation fixtures were provided which introduced loads into the IU in a manner compatible with the IU flight structure. The same handling and transportation design philosophy will be utilized for this study.

(a) Ground Winds

The results of the preliminary ground wind analysis are presented in Tables 3.1.3.6-I and -II based upon the following criteria:

Peak wind velocities for the 95% and 99.9% ground wind conditions, including influence of vehicle height above the natural grade, were estimated from Reference 3.1.3.6-1, pp 5.21 - 5.22.

3.1.3.6 (Continued)

$C_D = 0.7$ for all structures above the IU.

A factor of 1.55 represents an allowance for the dynamic response associated with vortex shedding and wind gusts.

Moment versus height for these ground wind conditions is plotted in Figure 3.1.3.6-5.

(b) Max Q Alpha

The results of the preliminary Max Q Alpha analysis are presented in Tables 3.1.3.6-III and - VIII based upon the following criteria:

Time versus vehicle mass, Mach number and dynamic pressure for representative payloads of each S-IC engine configuration were obtained from computer printout of preliminary trajectory and performance data received from The Boeing Company, Space Division, Launch Systems Branch, Huntsville, Alabama, for this study. A rigid body point mass is assumed without definition of angle of attack or engine gimbal angle(s).

The Max Q Alpha condition timepoint in the preliminary Boeing data was estimated as occurring approximately 7-9 seconds earlier than the Max Q timepoint based on IBM's experience for the three stage Saturn V/Apollo configuration.

The longitudinal g factor was determined as the total engine thrust minus the total weight drag referenced to the vehicle mas as the estimated Max Q Alpha timepoint.

The total vehicle drag and the drag forces for INT-20 Station 2245 at the estimated Max Q Alpha timepoint were estimated from aerodynamic data contained in Reference 3.1.3.6-2, pp 5-79 to 5-104.

The total vehicle drag versus Mach number was estimated from the graph on pp 5-83 of Reference 3.1.3.6-2 for the three stage MLV Saturn V-3B, three stage vehicle distribution of axial force coefficients for Mach numbers 1.30, 1.95, 3.0 and 5.0.

The data was replotted in terms of total aerodynamic axial force coefficient versus Mach number for that station, and is presented

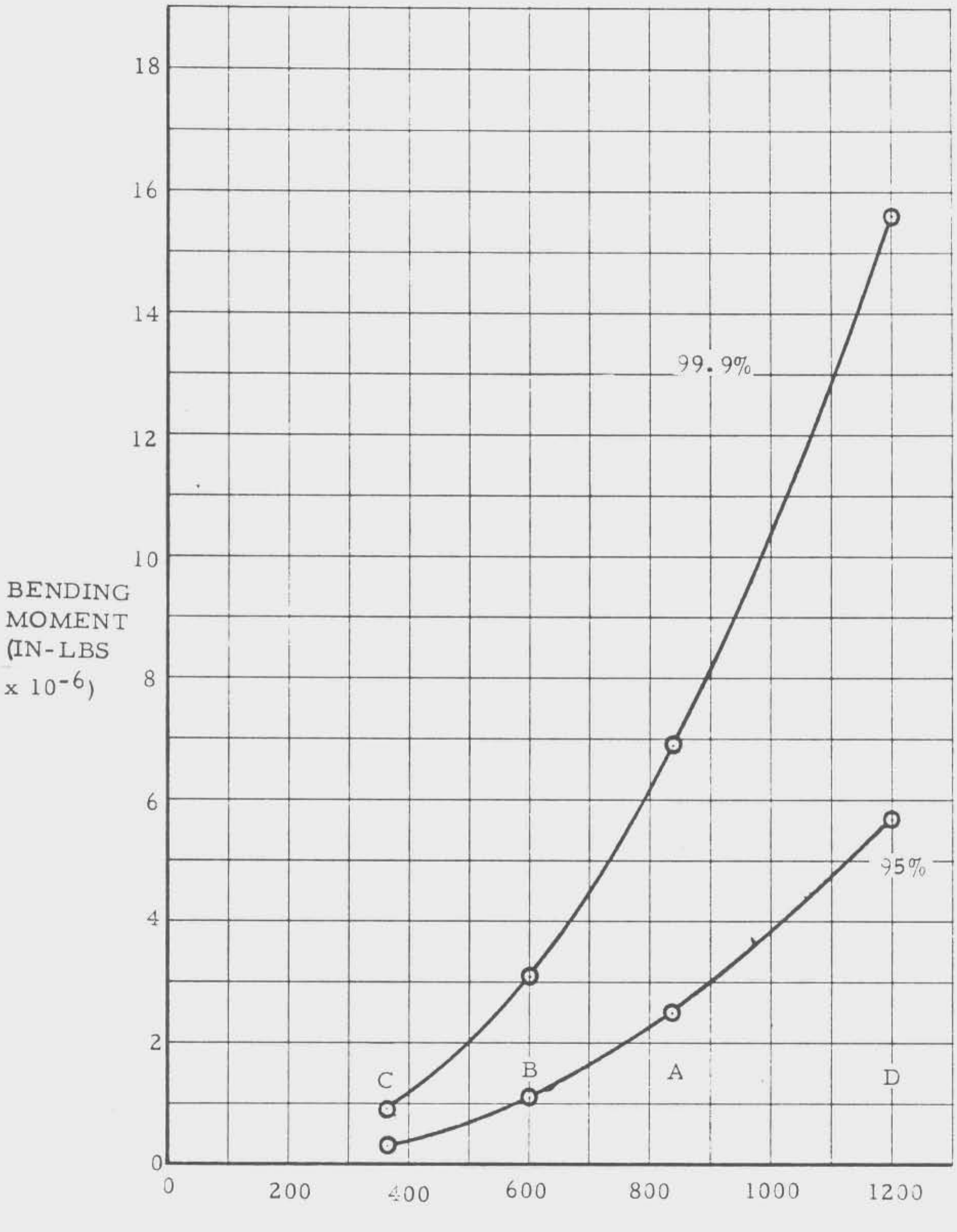


FIGURE 3.1.3.6-5. GROUND WIND BENDING MOMENT AT INT-20 STATION 2245

3.1.3.6 (Continued)

as Figure 3.1.3.6-6. Similar data for the MLV Saturn V-3B two stage vehicle from Reference 3.1.3.6-2, pp 5-93 and pp 5-94, is also presented as a reference for a 396 inch diameter payload.

The following assumptions were additionally employed:

The three stage vehicle payload configuration is substantially the same as the current study.

Small angle of attack.

Variation of cylinder payload length (i.e., INT-20 Payload Configuration A, B, C and D) may be neglected for the axial force determination since no boundary interactions appear to occur. Skin friction aft of the MLV double cone is neglected.

Axial force due to inertial loads was estimated at the Max Q Alpha timepoint by multiplying the longitudinal g factor by the payload plus IU mass configuration.

The bending moment for INT-20 Station 2245 was determined from preliminary data generated by The Boeing Company on January 3, 1969. This data is shown in different format in Figure 3.1.3.6-7. The Boeing Company presented the data as a plot of bending moment at Max Q Alpha versus payload length for payload weights of 40, 102, and 138 Kips.

Accurate bending moments for the various payload and engine configurations for each mission would have to be obtained through a computerized analysis.

The loads for specific configurations chosen for further study will be refined during the Phase II effort.

This preliminary data is reasonable for this trades study.

(c) End Boost

The preliminary End Boost loads for 4.28, 5.14, and 6.00 g's longitudinal acceleration, presented in Tables 3.1.3.6-V, VI and VIII, respectively, are based on the following assumptions:

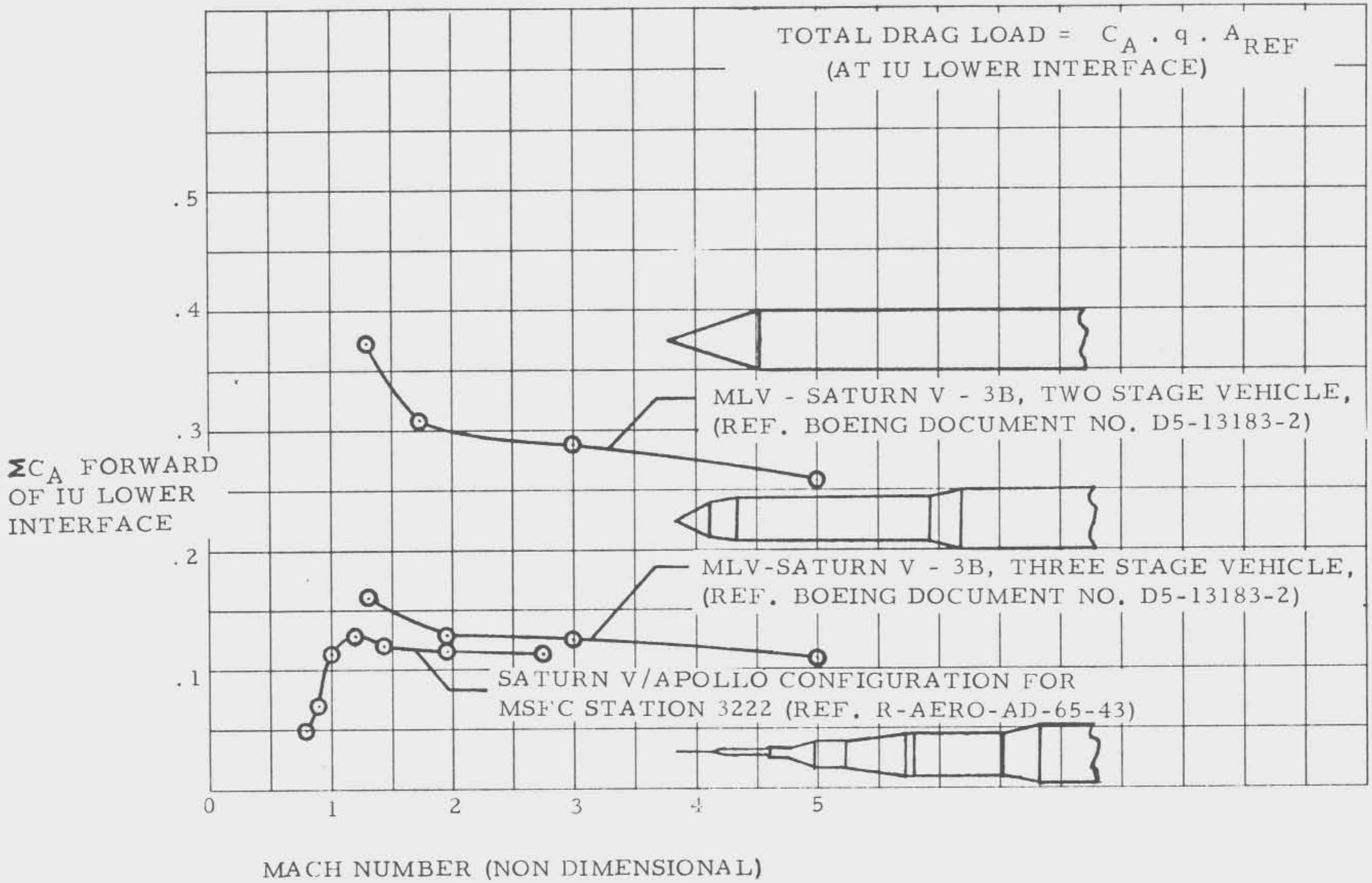


FIGURE 3.1.3.6-6. TOTAL DRAG LOAD COEFFICIENT AT INT-20 STATION 2245

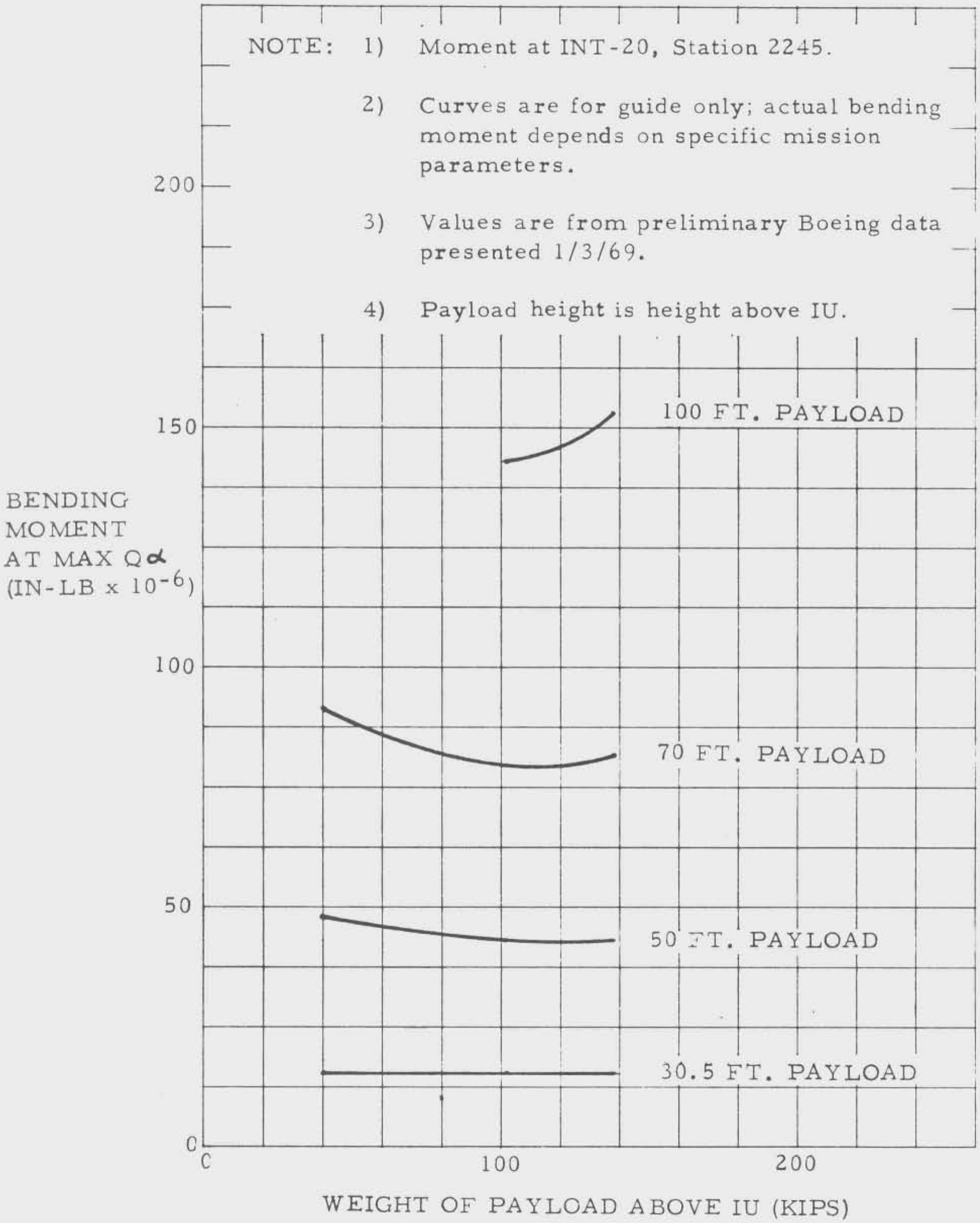


FIGURE 3.1.3.6-7. MAX Q ALPHA BENDING MOMENT AS A FUNCTION OF PAYLOAD HEIGHT AND WEIGHT

3.1.3.6 (Continued)

Axial drag load neglected. (There is a small air drag load.) Only inertial load influences are used for axial load. Pogo and other dynamic influences neglected.

The preliminary bending moment due to angle of attack and engine gimbal angle is estimated from the static portion of the End Boost bending moment calculated in Reference 3.1.3.6-3 of the Saturn V 505 and subs three stage Apollo payload configuration. The bending moment at Station 2245 for the present INT-20 configurations include consideration of the reference distance(s) to structures above the IU and the vehicle mass moment of inertia.

The bending moment is assumed proportional to the number of engines for the vehicle configuration, being higher for the greater number of engines used. This is not completely true because of actual angle of attack considerations yet to be defined.

(d) Dynamic Flight Loads

These loads, affecting rebound, Max Q Alpha, End Boost, and S-IC Stage Engine-Out conditions, were omitted by oral direction of Boeing for this trades study, and should be addressed in the preliminary design phase.

(e) EA, EI, and GJ Curves for the Instrument Unit

The following simplified assumptions are used for determining EA, EI, and GJ curves for the Instrument Unit structure for vehicle dynamic studies during the preliminary design phase . See Figure 3.1.3.6-8.

IU idealized as honeycomb shell with interface rails:

Structural discontinuities neglected honeycomb.

Honeycomb core and bond neglected. (This is not entirely accurate in that it is known some core and bond material act with the face sheets, but this will be reassessed during the preliminary design phase.)

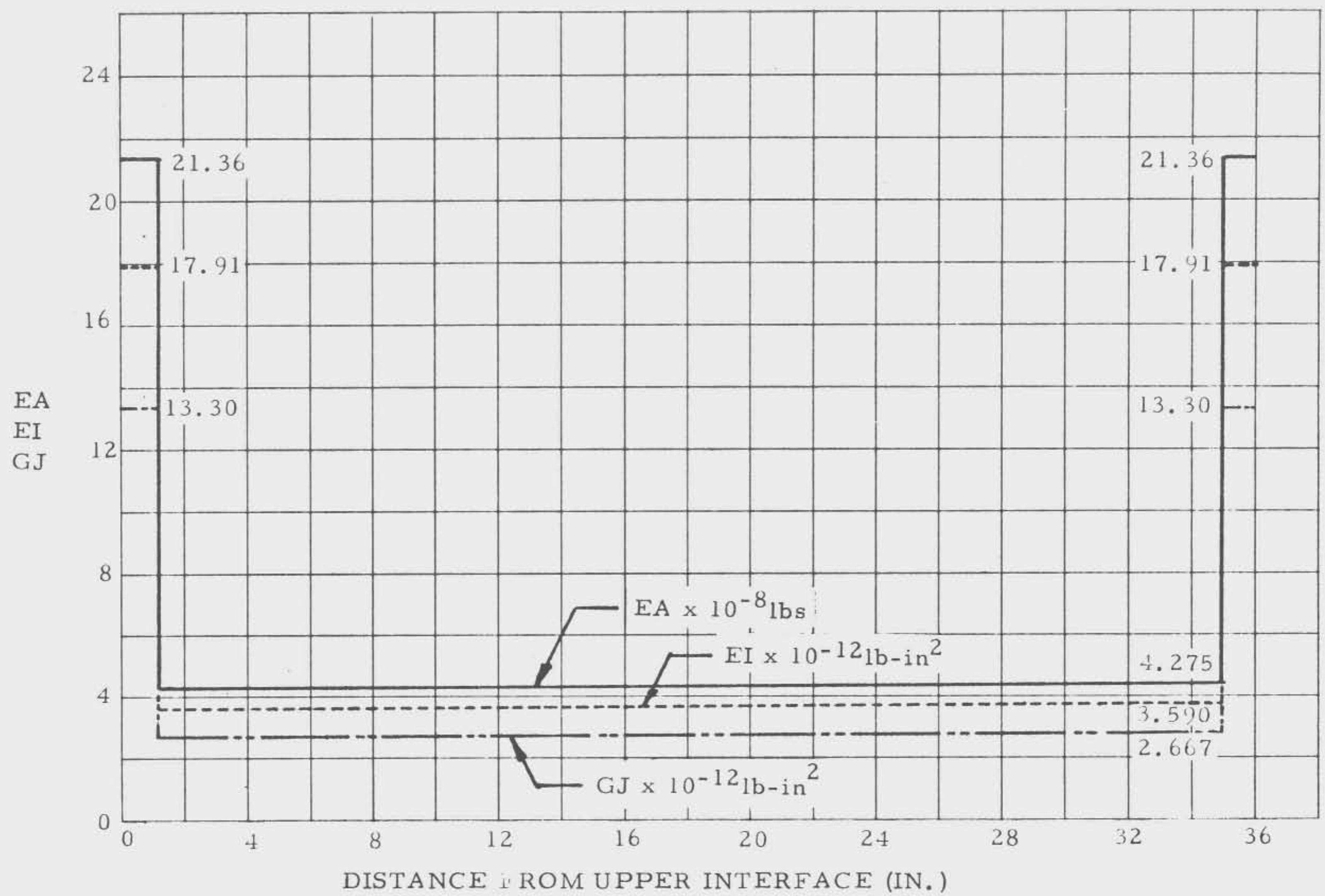


FIGURE 3.1.3.6-8. IU STIFFNESS DATA

3.1.3.6 (Continued)

Interface cutouts will be neglected. (This is not entirely accurate, either, particularly at the lower interface, but will be refined later.)

Radius to thickness ratio large. This allows the use of simplified formulas.

Nominal dimensions may be used with sufficient accuracy.

Compression modules of elasticity assumed since structure is principally under compression loads.

No thermal influences are assumed; i.e., values are for room temperature and without thermal load influences.

Neutral surface radii assumed.

Coldplate influences neglected.

Insulation, protuberances, etc., neglected.

c. Comparison of New Loads to Present IU Structural Capability

The present IU capability to resist structural loads compared to loads estimated in the previous section for the various study payload configurations/masses are shown in the following figures.

95% Ground Wind (Factor of Safety = 1.0)	Figure 3.1.3.6-9
99.9% Ground Wind (Factor of Safety = 1.4)	Figure 3.1.3.6-10
Max Q Alpha Flight Condition (Factor of safety - 1.25, 1.40)	Figure 3.1.3.6-11
End Boost Flight Condition for Insulated 500 Series IU (Factor of Safety - 1.25, 1.40)	Figures 3.1.3.6-12, -13, -14

The figures show running load capabilities used for these curves.

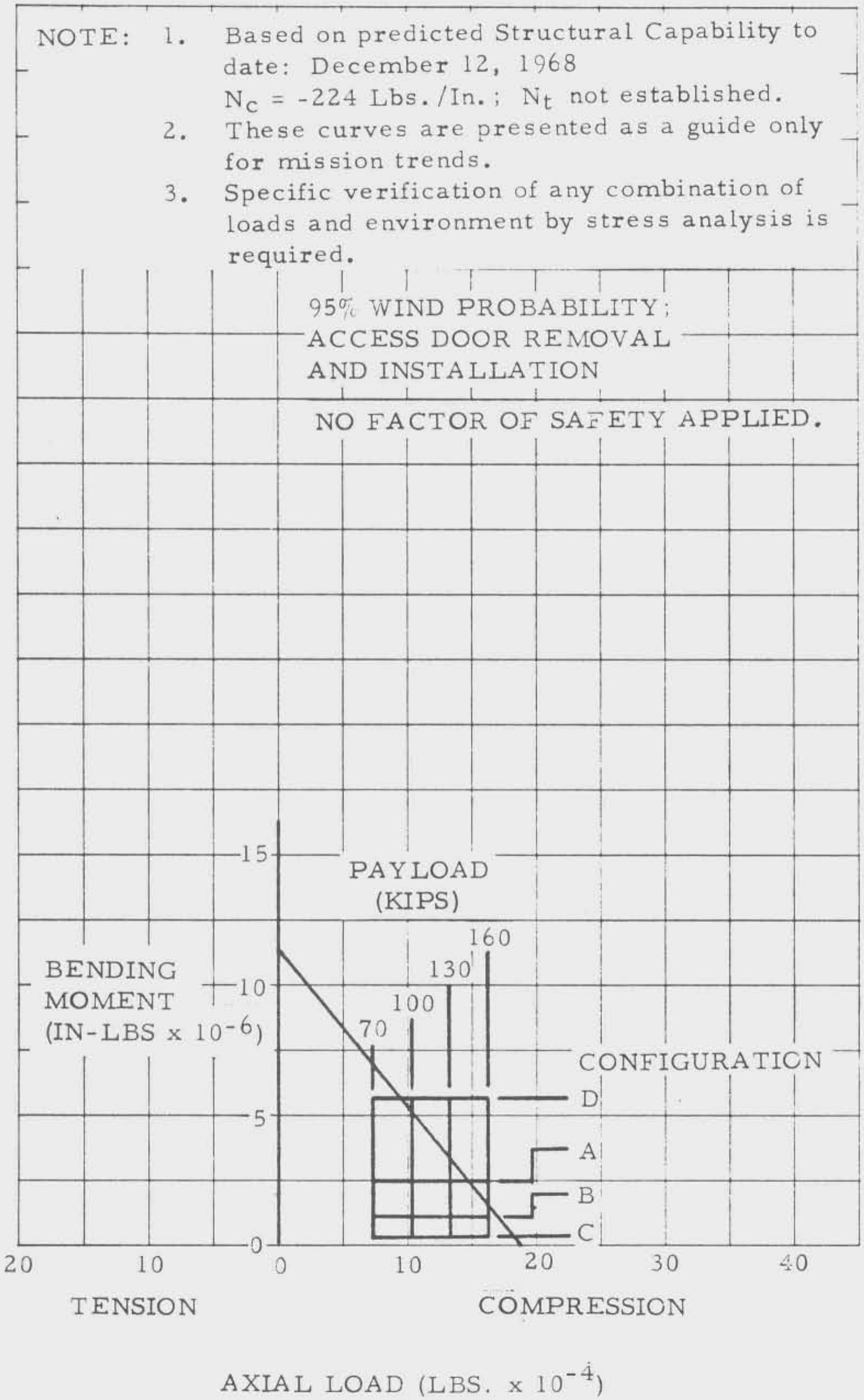


FIGURE 3.1.3.6-9. IU ON-PAD STRUCTURAL CAPABILITY, LOWER INTERFACE

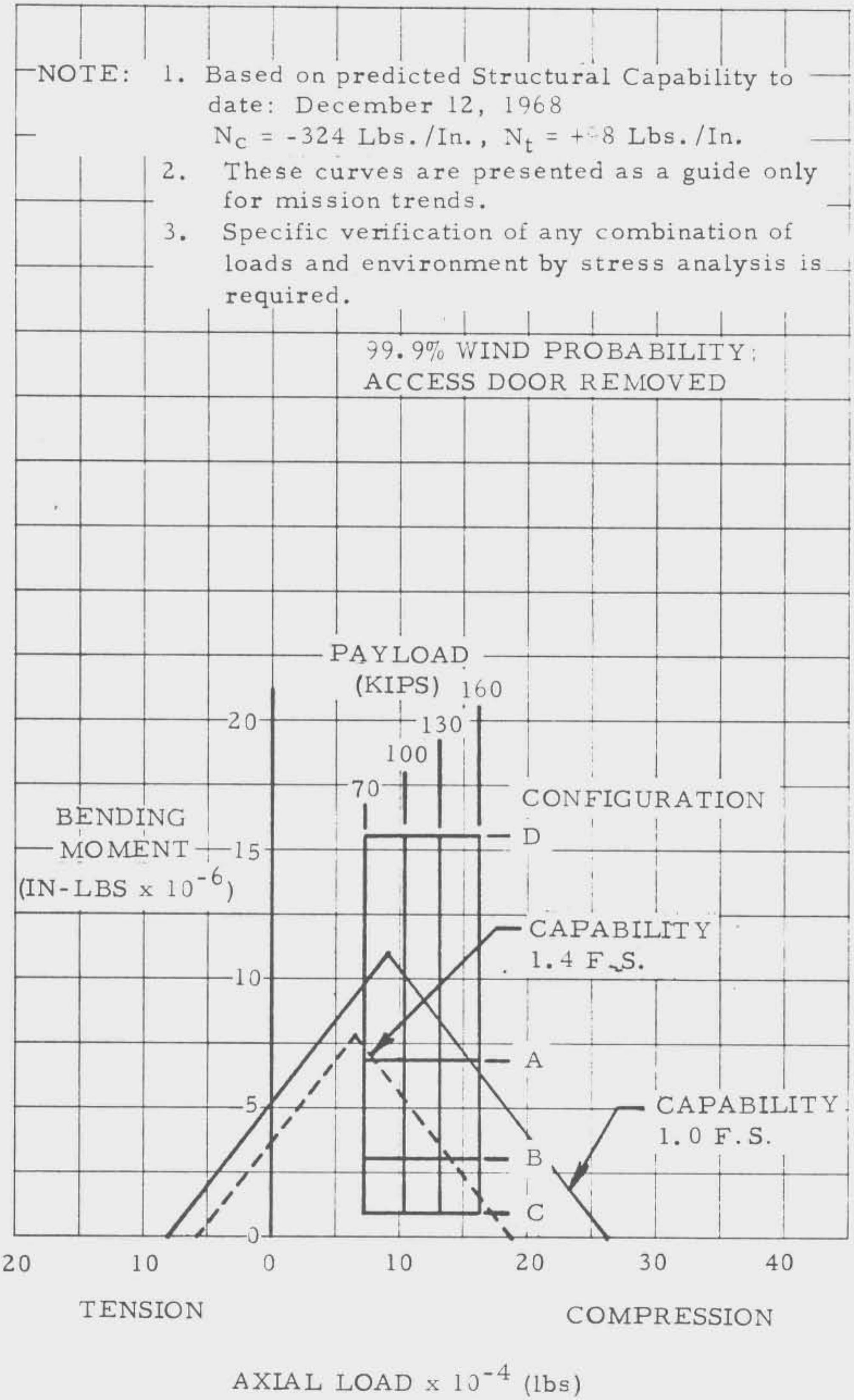


FIGURE 3.1.3.6-10. IU ON-PAD STRUCTURAL CAPABILITY, LOWER INTERFACE

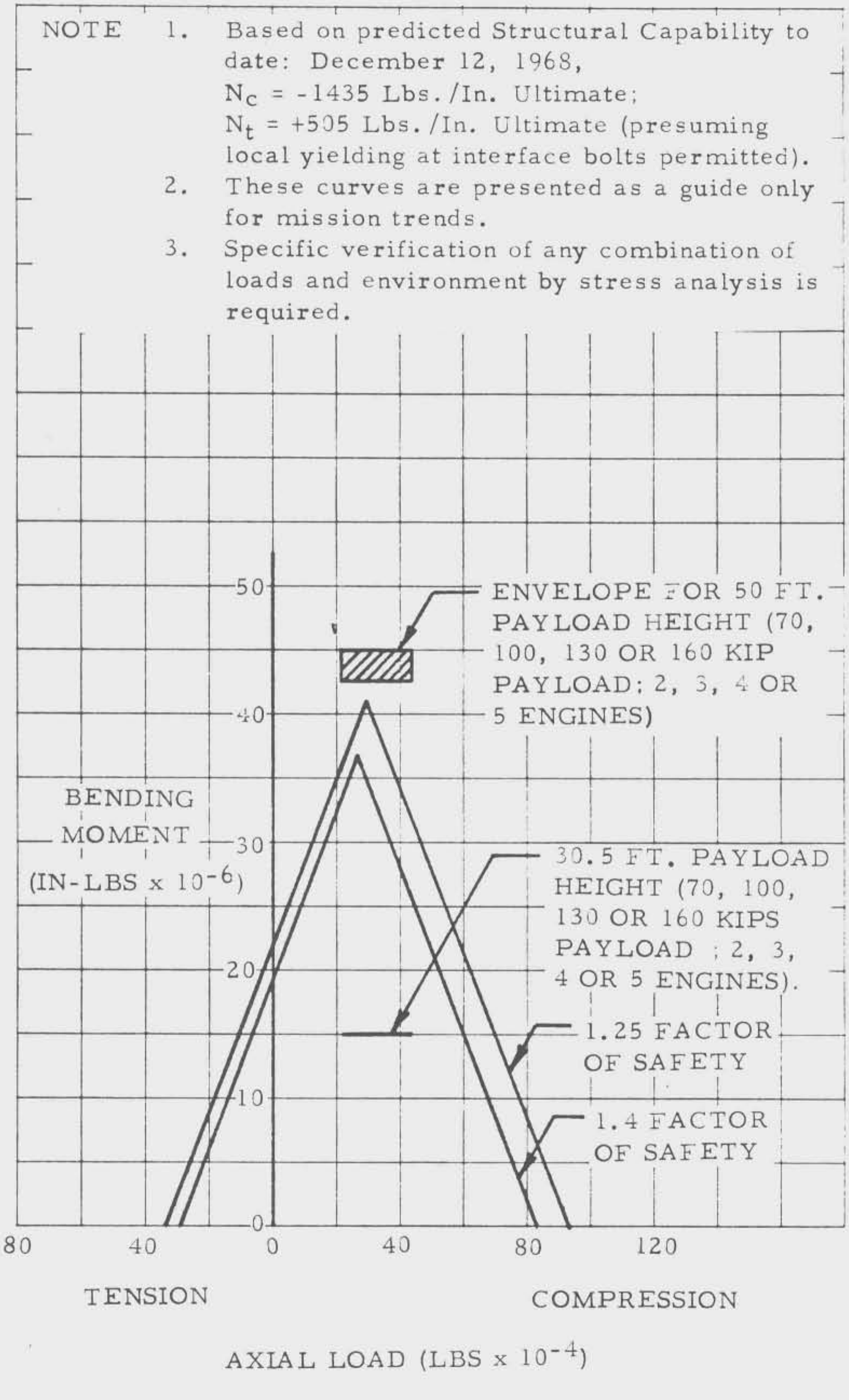


FIGURE 3.1.3.6-11. IU STRUCTURAL CAPABILITY, MAX Q ALPHA, LOWER INTERFACE

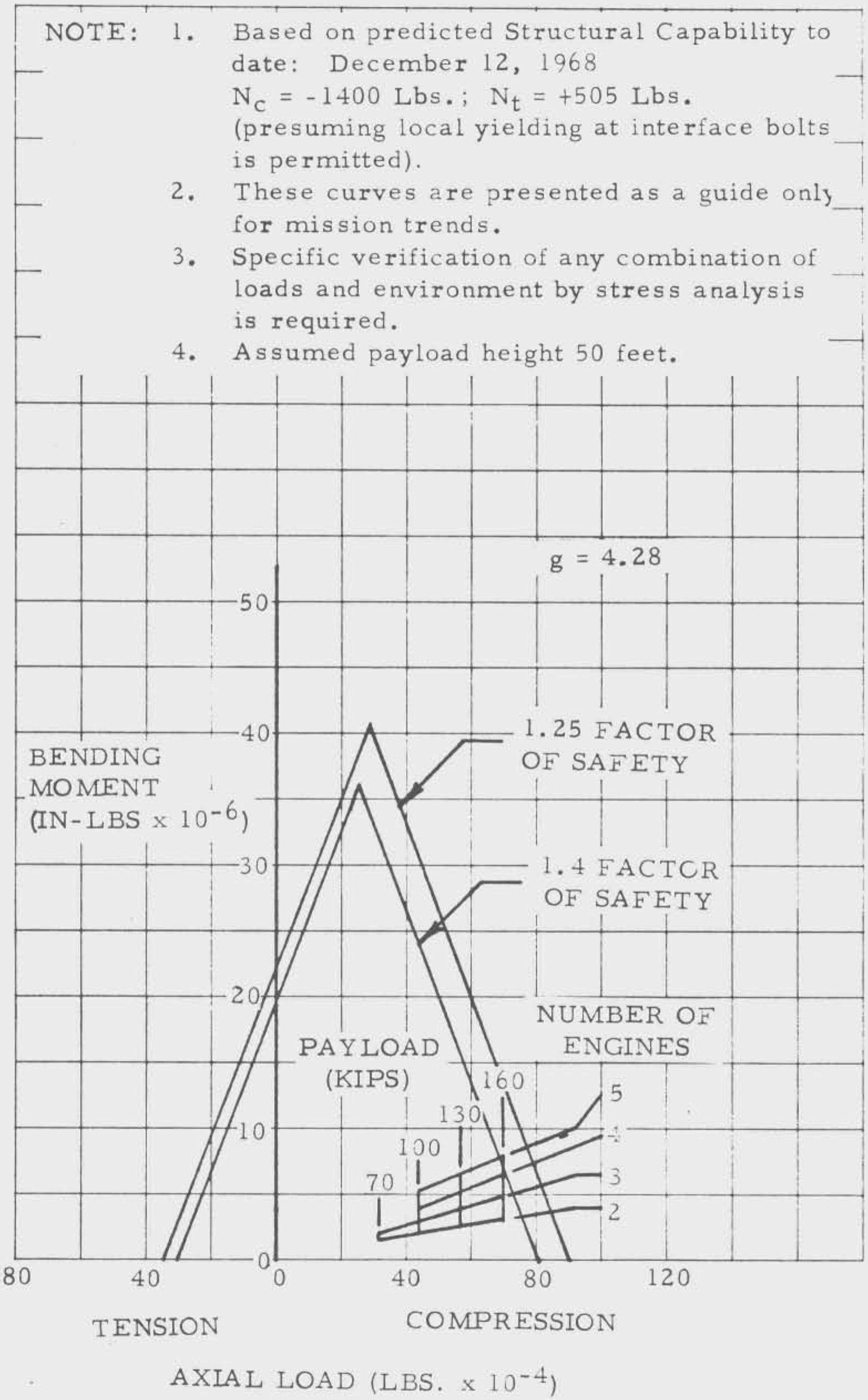


FIGURE 3.1.3.6-12. IU STRUCTURAL CAPABILITY, END BOOST, LOWER INTERFACE

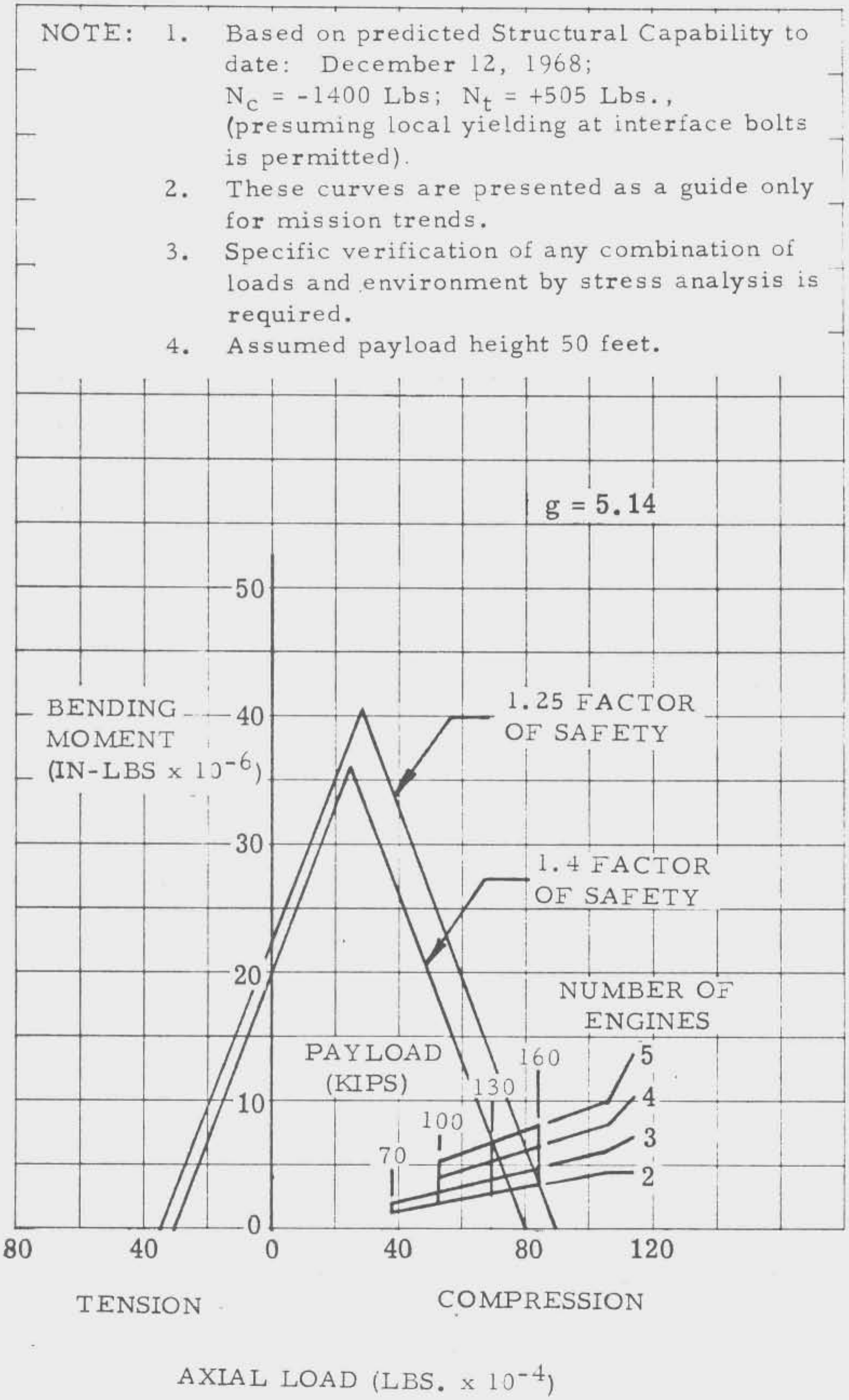


FIGURE 3.1.3.6-13. IU STRUCTURAL CAPABILITY, END BOOST, LOWER INTERFACE

NOTE:
1. Based on predicted Structural Capability to date: December 12, 1968; N_c = -1400 Lbs.; N_t = +505 Lbs., (presuming local yielding at interface bolts is permitted).
2. These curves are presented as a guide only for mission trends.
3. Specific verification of any combination of loads and environment by stress analysis is required.
4. Assumed payload height 50 feet.

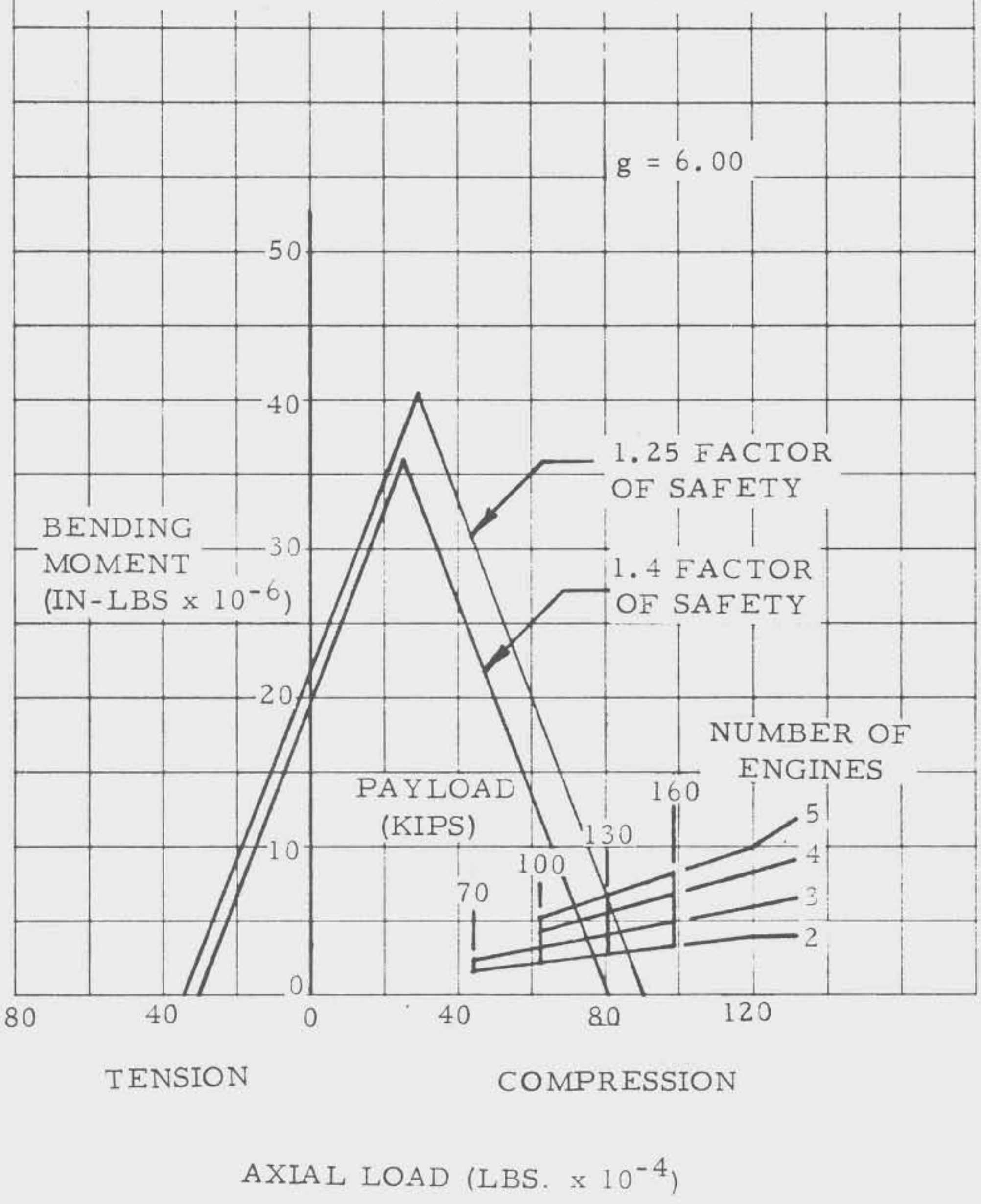


FIGURE 3.1.3.6-14. IU STRUCTURAL CAPABILITY, END BOOST, LOWER INTERFACE

3.1.3.6 (Continued)

The trade impact (no IU modification/modification) for the configurations under study is summarized in Table 3.1.3.6-VIII.

The prelaunch condition capabilities are based on IU test results summarized in Reference 3.1.3.6-4.

The flight conditions are based on updated capabilities reported by MSFC, with IBM concurrence, to Col. Lee James on September 27, 1968, superseding those capabilities previously reported.

The curves are approximate in that changes of adjacent structure and IU equipment static plus static equivalent dynamic loads may affect the compression and tension running load capabilities somewhat. This is due to the IU shell structural system being designed both for local component loadings (affected by equipment arrangement and masses) as well as adjacent stages. The curves presented are based on expected equipment loads for the Saturn V/ Apollo general equipment arrangement and acoustic/vibration environment.

Only the lower IU interface was considered in generating the curves. The lower interface loads govern in the absence of local peaking loads.

Values of estimated loads outside the required safety factor line with reference to zero axial load and bending moment indicate IU structure modification is required. No IU modification/type of IU modification for the various study configurations is coded in Table 3.1.3.6-VIII. Section 3.1.3.6 d describes the main modifications for those study configurations not meeting present IU structural capabilities, which fall in four categories:

Increased compression running load capability required.

Increased tension running load capability required.

Increased ground wind prelaunch influences causing the present structural access door area to be changed to a non-load bearing access door design.

Consideration of local effects (equipment component attachments, internal frams, antenna cutouts, splices, etc.)

The IU cork insulation may be unnecessary for some of the study configurations for which no modification is required. The weight delta is 75 lbs. This will be re-assessed during the preliminary design phase when structural loads and associated environment would be refined.

3.1.3.6 (Continued)

d. Modification Requirements

The following modification requirements must be addressed for those study configurations not within the current IU structural capability envelope. The estimated weight impact is summarized in Figure 3.1.3.6-15.

1. Increased Compression Running Load Capability Required

Figure 3.1.3.6-16 determines the IU honeycomb shell stability capability for a 260 inch diameter structure for a number of different 7075-T6 aluminum alloy skin thicknesses and two core densities. The upper curves are theoretical and based on simply supported end conditions at room temperature. The lower curve is established from a full-scale IU test involving the aft SLA and forward S-IVB stage assuming that shell stability failure was eminent. The same approximate semi-empirical curve would be required for local failure.

It was shown in the IBM First Performance Review for the INT-20, 18 July 1967, NAS8-21076, that a change in core density would have little influence on structural capability. Therefore, if no major component location changes are required, the present IU core configuration of approximately 85 percent 3.1 lbs per cu ft core would be kept unchanged.

The present IU construction total face sheet thickness is 0.050 inch. The theoretical curve indicates that the present IU structure is capable of resisting a 2770 lbs per inch compressive load. Using the semi-empirical curves indicates a 1435 lbs per inch capability.

The worst structurally loaded INT-20-IU has an ultimate design load requirement of 1921 lbs per inch. The semi-empirical curve indicates that this requirement total face sheet thickness of .071 inch is required.

There are several options to implement this, largely dependent on the amount of additional skin thickness required. For a small increase of total skin thickness on the order of 5%, as reported in the previous INT-20 study, the total design impact change would be optimum for the outer face sheet. For any greater required increase of skin thickness, it would be better to increase the thinner inner skin to maintain nearly equal face sheets and therefore better load path, but this would involve considerable relocation of internal IU component equipment attachment hardware even though the dimensional changes are small. For outer face sheet changes, there are changes involving antenna cutout dimensions, etc., but so far lesser extent than changing the inner skin thickness.

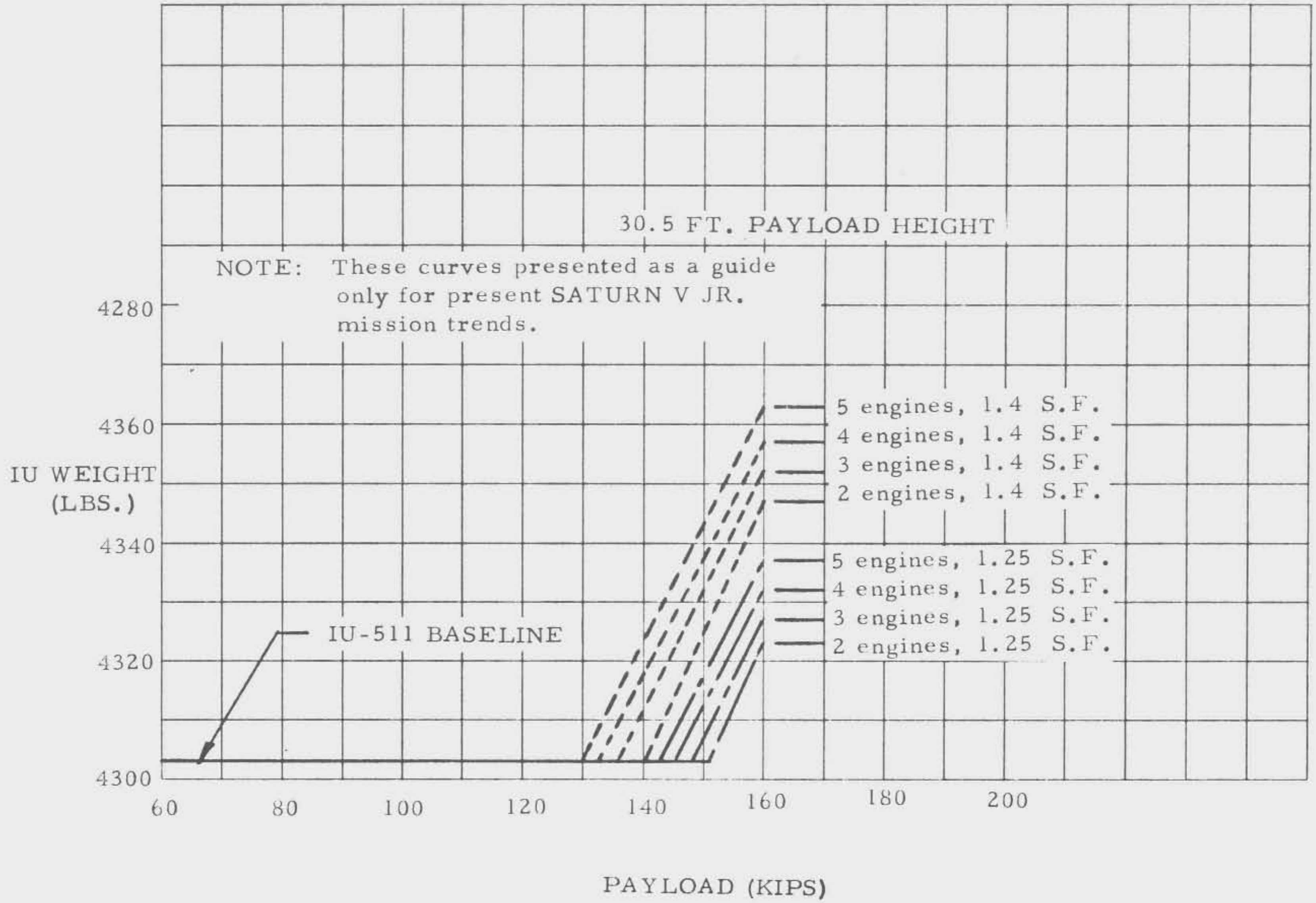


FIGURE 3.1.3.6-15. IU WEIGHT AS A FUNCTION OF PAYLOAD

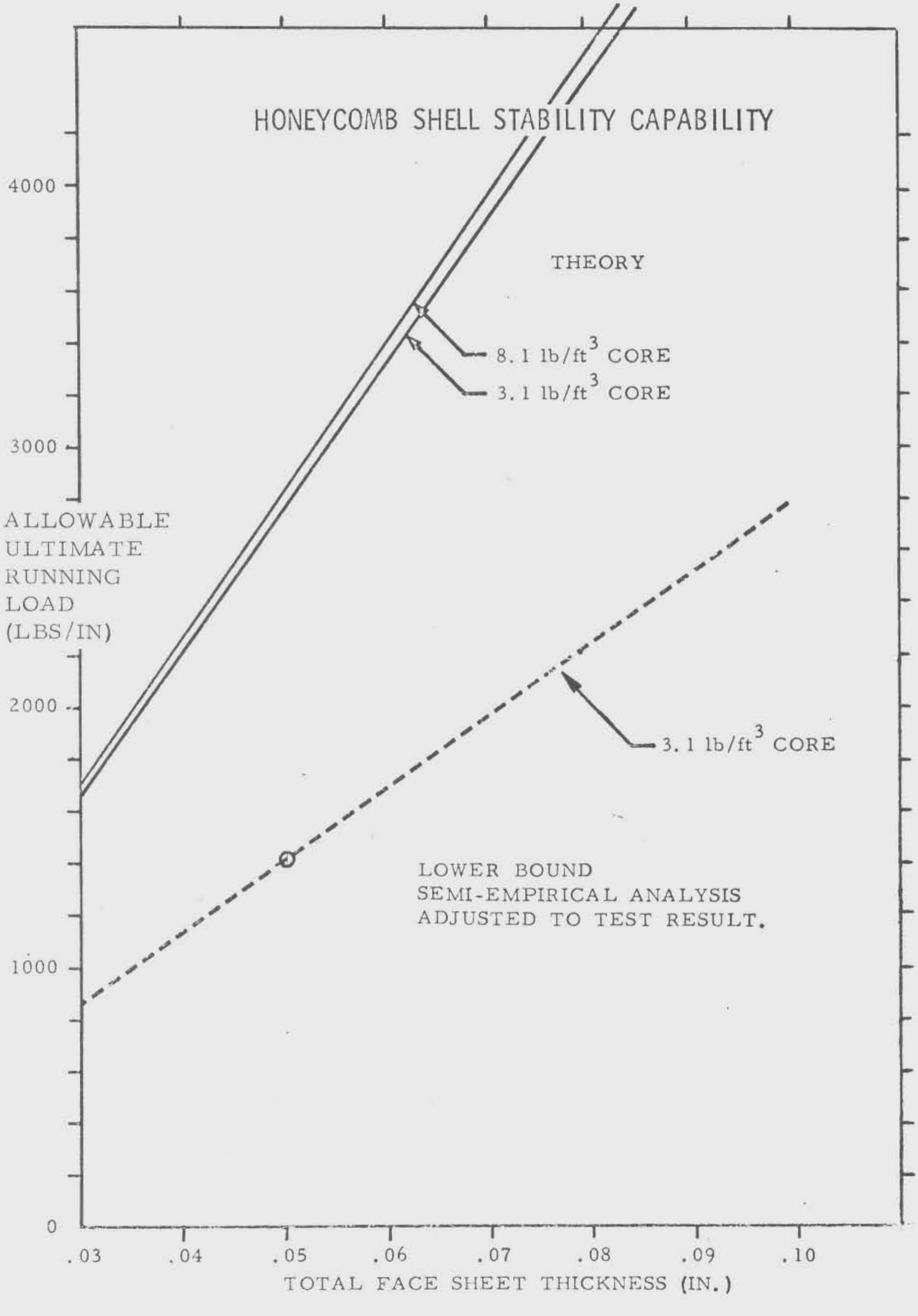


FIGURE 3.1.3.6-16. HONEYCOMB SHELL STABILITY CAPABILITY

3.1.3.6 (Continued)

This will be further examined during the design phase effort.

2. Increased Tension Running Load Capability Required

Figure 3.1.3.6-17 determines the IU lower interface rail capability for a 260 inch diameter structure for a number of different extrusion lengths to supply more tensile-shear bond area and flange thicknesses to increase local flange bending capability. If this type of modification is required, the shape of the IU interface bolted access hole cutouts would also be altered to reduce stress concentration factors from being a governing factor.

The upper IU interface rail capability is on the same order of magnitude as the lower, and similar changes could be assumed for preliminary weight impacts.

Above certain tension load levels, the inner skin thicknesses also would have to be increased, but it is not addressed in this trades study.

At certain levels of increased tension loads, the number/strength of interface bolts would have to be addressed. This impact is not described in the current study.

3. Access Door Modifications

The access door is load carrying and must be capable of being removed and reinstalled any time prior to flight. Also, the structure must be capable of sustaining the vehicle ground loads when the door is removed. This requirement is illustrated by Figure 3.1.3.6-18.

A major redesign is recommended for this area for loads much greater than present requirements (i.e., all D and several A configurations depicted in Figure 3.1.3.6-1). The present design of a structural load bearing access door would be modified to a non-load bearing door.

Since the access door is at the edge of one IU structural segment, redesign of this area on two structural segments is required. The interfaces, internal frames, and splices would be considerably changed. If increased compression and tension running load capability for flight loads is additionally required, maintaining the access door size may be a problem.

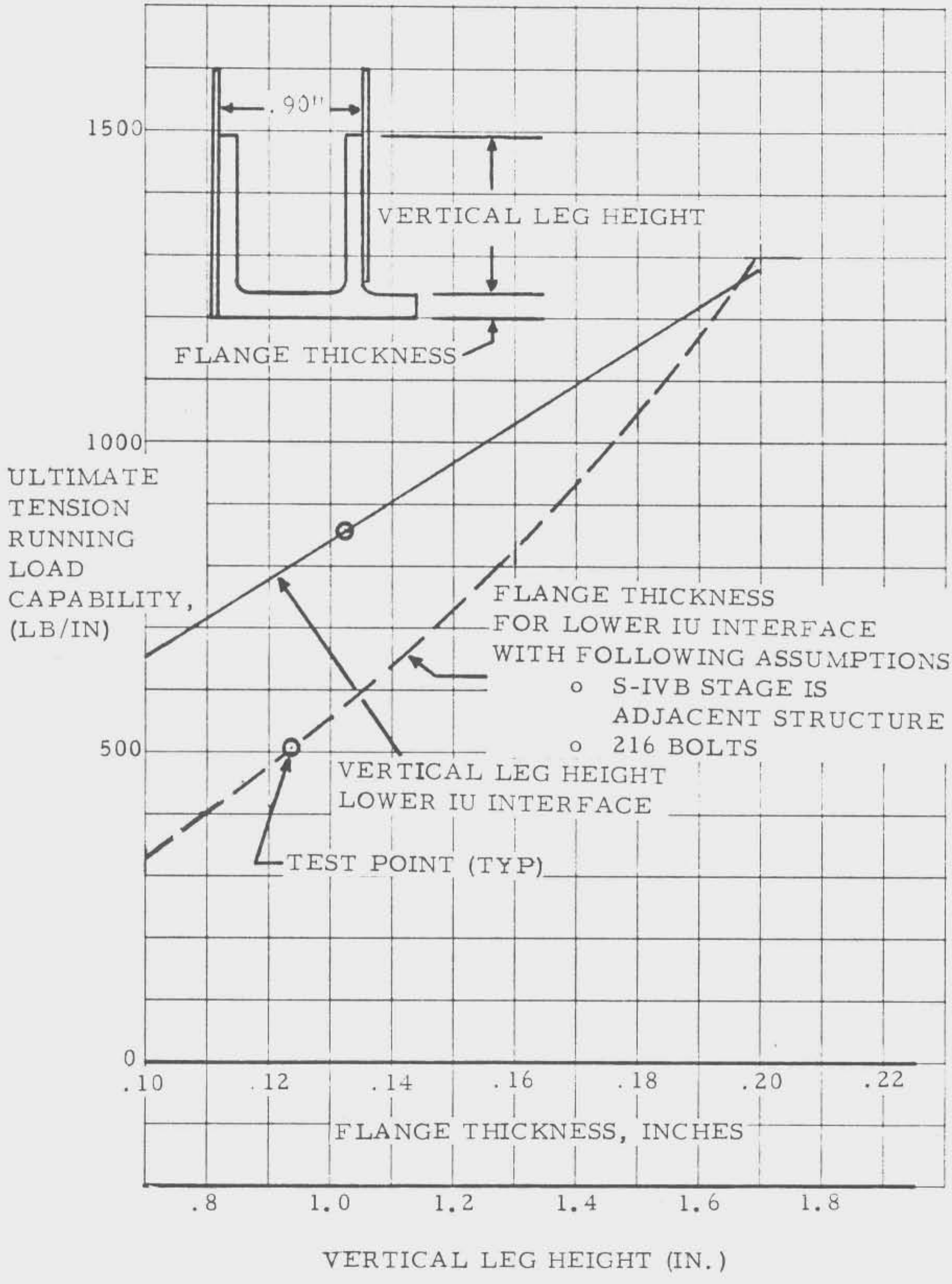
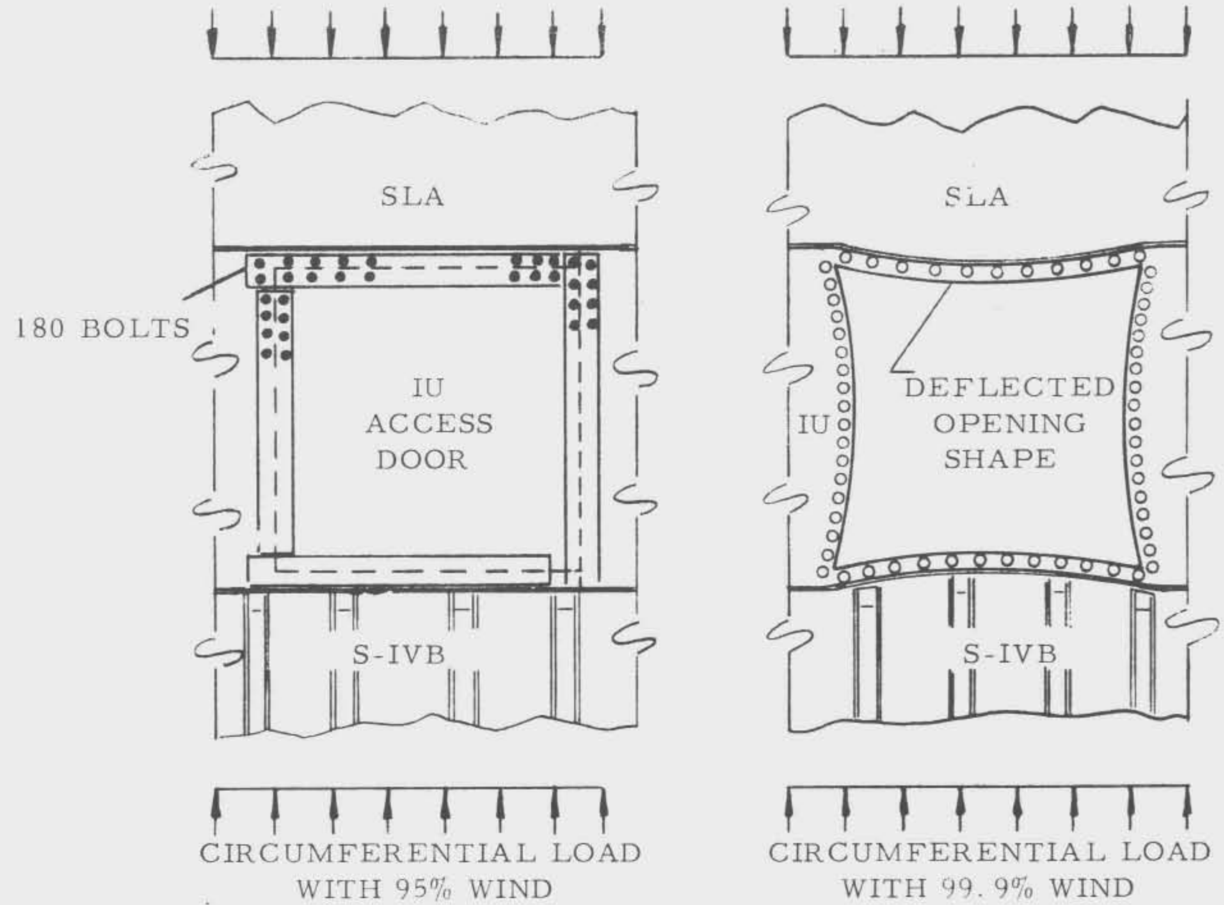


FIGURE 3.1.3.6-17. INTERFACE TENSION CAPABILITY



VEHICLE	INSTALLATION AND REMOVAL QUALIFICATION LOAD (lb/in)	IU STATIC QUALIFICATION WITH DOOR REMOVED (lb/in)
PRESENT IU	224	324

FIGURE 3.1.3.6-18. ON-PAD ACCESS DOOR DEFLECTION

3.1.3.6 (Continued)

4. Consideration of Local Effects

(a) Equipment Component Attachments and Beam Column Action

The local beam column considerations must include not only the basic vehicle shell loads, but also the lateral loads imposed by the IU components attached to the basic honeycomb structure. These lateral loads are intensified by the dynamic environment imposed and the component and attachment dynamic response. Also, equipment component attachments may be critical.

The dynamic equivalent loads will be defined and impacted in the preliminary design phase.

An important observation should be made when discussing the action of heavy components mounted to the IU structure, that 8.1 lbs per cu ft core is used to re-distribute loads imposed by brackets, pads and internal and external framing. Therefore, certain component additions or changes could require redesign of the core pattern prior to bonding. This does not appear to be a problem with the INT-20-IU.

(b) Close-Out Frames, Internal Frames, Cutouts

An evaluation of the adequacy of the present IU structure's internal and external frames and cutouts against the INT-20 design loads was made. This evaluation used the present IU stress analysis and considered all low margins of safety. These low margin areas were then compared to the measured stresses from the static qualification tests on the present IU structure.

It was concluded that for the 160,000 lb payloads (g = 5.14 to 6.00) or for payload heights in excess of approximately 45 ft, IU internal and external frames or cutouts may have to be modified for the INT-20-IU. This will be assessed for the preliminary design configurations in the next work phase.

(c) Placement of Added or Changed Components

In the consideration of the IU assembly configuration or component location, the approach has been taken to minimize the required component relocations versus the present IU assembly configuration.

3.1.3.6 (Continued)

This approach generally minimizes the cost of component changes but could result in increased IU weight caused by the use of longer cable lengths.

(d) Interface

Adjacent stages, such as the MLV payload, may require changes for compatibility.

e. Structural Qualification Test Considerations

For INT-20 configurations/payloads requiring IU structural modifications, if it is anticipated where the change is small (i.e., a skin thickness change of less than 5%) large scale requalification may be bought off by new strength analysis compared to IBM test experience.

In the event structural testing is required, and the design change not radical, one IU structural segment with simulated boundary structures would be tested on the basis of IBM test experience. This is because the manufacturing processes, specifications, and controls would be the same.

Where major redesign is required for the access door, the entire IU shell structure must be requalified.

Since structural testing may be required for the MLV payload, and an IU simulated structure required for such testing, an actual IU might be qualified in the same stack.

In addition, small specimen testing would be required to verify basic compression and tension configuration allowables.

f. Additional Considerations

1. Stage Separation

Although stage separation concepts/loads were not addressed for this preliminary trades study, such considerations may be important in the preliminary design phase.

IBM, in a company funded study, has developed a separation system for the IU/S-IVB interface, as depicted in Figure 3.1.3.6-19, which may provide this capability. Interface panel tension and panel separation test programs were successfully performed.

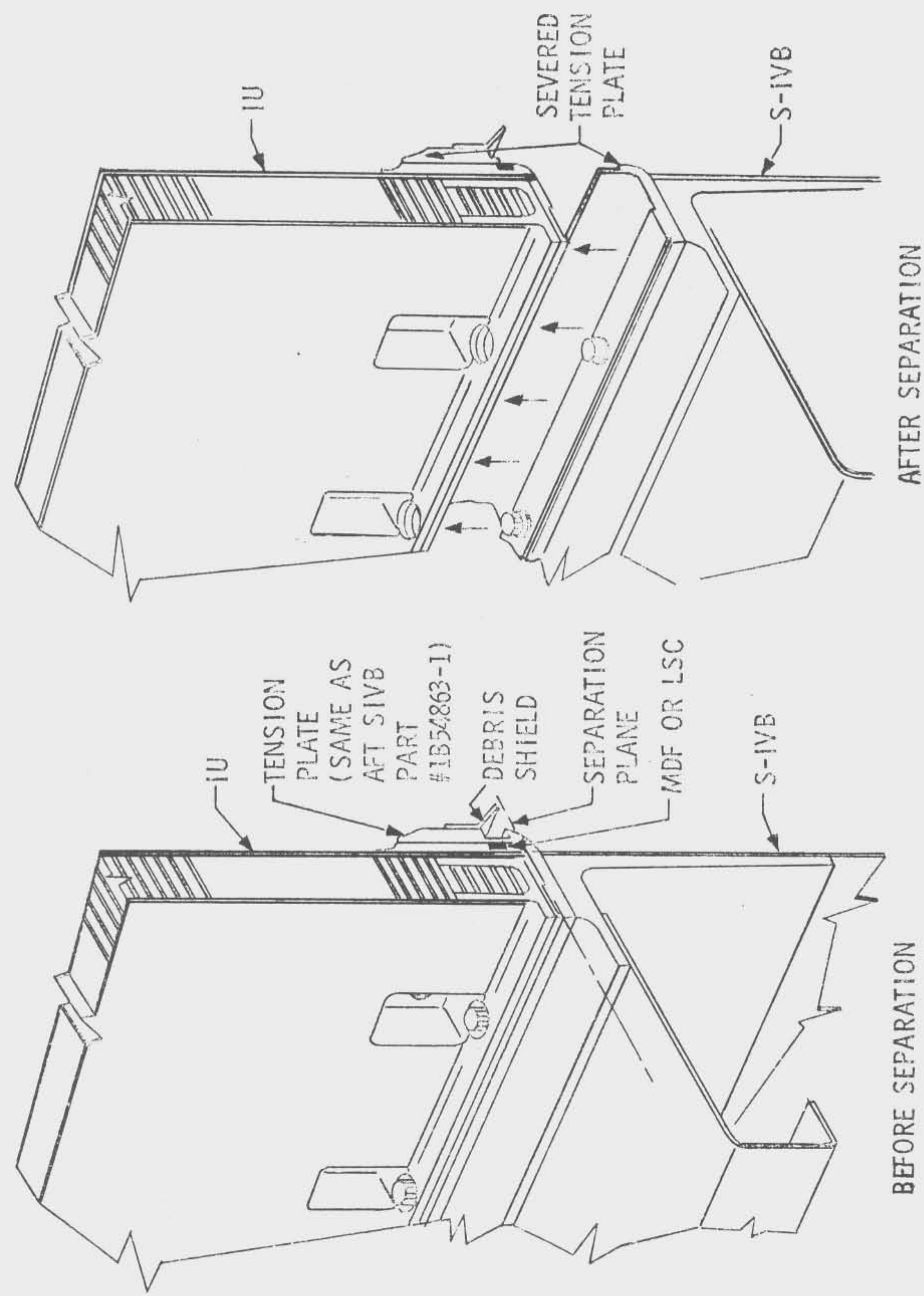


FIGURE 3.1.3.6-19. IU/S-IVB SEPARATION CONCEPT

3.1.3.6 (Continued)

Since the IU interface rings are essentially the same and the IU structure honeycomb depth is constant, such a separation system may be used for the upper IU interface as well.

2. IU Spacer

The AAP-4 Spacer consists essentially of an IU structure without equipment components (although coldplate mounting provisions to the IU ECS system could be easily accomplished), access door, and umbilical plate, and is 36 inches high.

The structure has been designed with all pertinent documentation, and is awaiting MSFC manufacturing turn-on.

The IU Spacer may have presently unforeseen advantages as a module with the MLV cone and cylinder payload in the INT-20 study program.

g. Conclusions

It is recommended that the Saturn V, or 500 series, IU structure be considered in INT-20 future studies rather than the Saturn IB, or 200 series, because of the higher capability at end boost and the ST-124 damping pad installation.

With the C configuration the current Saturn V IU is structurally adequate for some combinations of engine configuration, payload weight, and g level as shown in Table 3.1.3.6-VIII. If the B, A, or D payload height configuration is used, all combinations of payload weight, engine configuration, and g level would require modification to the IU structure.

Where modification is required to increase the IU capability for certain payload and engine configurations, the type of modification is indicated in Table 3.1.3.6-VIII. For load levels somewhat higher than capability, the structures impact can be minimal. However, for load levels much greater than current IU capability, the impact will be to a greater level. The estimated weight impacts of such modifications are shown in Figure 3.1.3.6-15.

Modifications for increased interface compression loads beyond present capability are easy to accommodate by increasing skin thickness. Increased tension capability can be accomplished by redesign of the interface rails combined with an increase in inner skin thickness. Neither of these modifications can be accomplished by retrofit.

3.1.3.6 (Continued)

Structural requalification is not considered mandatory for minor structural changes required. Where the access door must be redesigned for the ground wind loading of the B, A, and D configurations, a full-scale qualification program is required. This requalification should be in conjunction with an MLV payload test program.

3.1.3.7 Costs

The schedule of estimated costs for design and development and a delta recurring cost for production are based on the technical disclosures in other sections of this report. These estimates are presented in order under headings as follows:

a. IU Environments

The development costs under this heading are a function of the number of F-1 engines in the S-IC stage and accelerations of 4.68g and 6.00g. Four independent cost totals labeled A, B, C and D were derived and apply according to the combinations shown below.

No. Engines	3 F-1	4 F-1	5 F-1
Accel G=4.68	--	--	A, B, C
Accel G=6.00	D	D	A, B, C, D

ST-124 Vibration Analysis (A)		$ 3,086.00
Random Vibration Exceedance Analysis (B) (Structures Location 6 Hardware)		
Engineering	$ 2,315.00	
Material	55,475.00	
Testing	19,577.00	
Test Support	14,264.00	
Total		$91,631.00
Random Vibration Exceedance Analysis (C) (Structures Location 22 Hardware)		
Engineering	$ 2,315.00	
Material	10,496.00	
Testing	15,660.00	
Test Support	11,640.00	
Total		$40,111.00

3.1.3.7 (Continued)

Increased Acceleration Requirements (D)

Engineering	$ 7,330.00	
Material	41,812.00	
Testing	23,490.00	
Test Support	14,264.00	
Total		$86,896.00

For five F-1 engines under 4.68g conditions, the estimated development cost is the sum of A, B and C, or $134,828.

For five F-1 engines under 6.00g conditions, the estimated development cost is the sum of A, B, C and D, or $221,724.

b. Structures

Independent development cost estimates under this heading are coded as follows:

0 - "Analysis Only"

1 - "Increased Compression Design"

3 - "Access Door Design"

These codes correspond to those used in the technical discussion of impact on IU structures design. Since tension requirements do not control the design, development costs for code 2 "Increased Tension" were not derived.

0 - "Analysis Only" $ 3,921.00

1 - "Increased Compression Design"

Engineering	$64,429.00	
Material	43,000.00	
Test	110,925.00	
Test Support	30,236.00	
Total		$248,590.00

3.1.3.7 (Continued)

3 - "Access Door Design"

Engineering	$ 96,643.00	
Material	130,000.00	
Test Support	64,032.00	
Total		$290,675.00

The fabrication and assembly costs for the production of the IU shell for requalification testing of the redesigned structures are not included.

The estimated cost for performance of the IU requalification test is not shown above, however an estimate was derived. The total estimated cost for IU qualification testing is $326K which includes labor and an estimate of material costs for test instrumentation and fabrication of simulated upper and lower stage structures. Such test costs might be reduced if the requalification testing would be accomplished by the government using government facilities, or if the simulated structures were available, or if such testing were accomplished concurrent with MLV payload structures testing using the redesigned IU structures assembly as a boundary structure to that test.

c. Flight Control Computer (FCC) Modifications

Estimated non-recurring development costs:

Total	$33,657.00

d. Recurring Production Delta Cost

There is an estimated increase in the material cost of a set of redesigned structures segments of approximately $15,000 per IU. This represents approximately a 15% increase over the present cost of a set of IU structures segments.

e. Groundrules and Assumptions

Industrial base costs are not shown. Only non-recurring costs for development and recurring delta costs for the operational program are presented.

Maintenance of capability prior to the start of a development program is assumed.

Cost estimates are expressed in 1969 dollars without inflationary factors applied and include burden G&A, General Research, IRAD and 7% fee.

3.1.3.7 (Continued)

The estimates presented here do not represent actual costs, nor do they represent a commitment on the part of the IBM Corporation and use should be restricted to reference for the purpose of the Saturn Improvement Studies.

3.2 VEHICLE ANALYSES

Several vehicle-oriented analyses were performed to provide an understanding of the capabilities, limitations, and development requirements of each INT-20 configuration. The vehicle analyses did not differentiate between 200-series and 500-series S-IVB and I.U. stages, since weight differences are minimal and the other differences are mission-related. Some performance data was generated for the INT-20 with Centaur and Service Module injection stages, although these data were not needed for the baseline trades.

3.2.1 Performance Evaluation

A set of flight performance data to reflect performance variation was generated for each INT-20 configuration. A lift-off thrust-to-weight ratio of 1.25 was used for each configuration (except for the 5 F-1 version). Ballast, if required, was contained in the first stage as unused propellant (S-IVB propellant weight was held constant). Stage weights based on AS-516 were used in the trade studies and are shown in Table 3.2.1-1. A dynamic pressure limit of 950 lb/ft^2 was observed in shaping the trajectories.

The complete set of flight performance data generated during Phase I is contained in Appendix D, Part 1.

3.2.1.1 Generalized Performance (C_3) Comparisons

The capabilities of different INT-20 configurations for performing various missions can be directly compared by using generalized performance, or C_3 (twice specific energy). data. (Any mission can be equated to a C_3 value. For example, a 100 N.M. circular orbit mission has a C_3 value of about -60.8 km^2/sec^2, a 72-hour lunar transfer about -1.5, and synchronous missions range from about 16 for a 28.5^o inclination orbit to about 25 for a 0^o inclination orbit.) C_3 data were prepared for each engine option at peak acceleration levels of 4.68 g's and 6.0 g's.

a. Adding Engines to S-IC

The variation of payload capability through a C_3 range of -50 to 100 km^2/sec^2 is shown for 4.68-g-limited vehicles in Figure 3.2.1.1-1, and for 6.0-g vehicles in Figure 3.2.1.1-2. The addition of engines to the S-IC is reflected on the C_3 plot as an increase in vehicle capability throughout the C_3 range. The differential between the 4 F-1 and 5 F-1 vehicle performance is relatively small for the 4.68-g case. For the 6.0-g case, both vehicles

TABLE 3.2.1-1

INT-20 TWO-STAGE TRADE STUDY BASELINES

	NUMBER OF F-1 ENGINES		2	3	4	5
	LIFT OFF WEIGHTS	LBS	2,435.200	3,652,800	4,870,400	*
S-IC	SEA LEVEL THRUST	LBS	3,044,000	4,566,000	6,088,000	7,610,000
	SEA LEVEL I_{SP}	SEC	263.58	263.58	263.58	263.58
	PROPELLANT CONSUMED	LBS	*	*	*	*
	STAGE INERTS **	LBS	275,726	304,647	335.478	364,399
S-IVB	VACUUM THRUST	LBS	205,000	205.000	205.000	205,000
	VACUUM I_{SP}	SEC	426	426	426	426
	PROPELLANT CAPACITY	LBS	230,000	230.000	230.000	230.000
	STAGED INERTS	LBS	27,181	27.181	27.181	27,181
IU	ASTRIONICS EQUIPMENT	LBS	3,847	3.847	3.847	3.847

* VARIABLE WITH MISSION

** DOES NOT INCLUDE BALLAST FOR ACCELERATION CONTROL

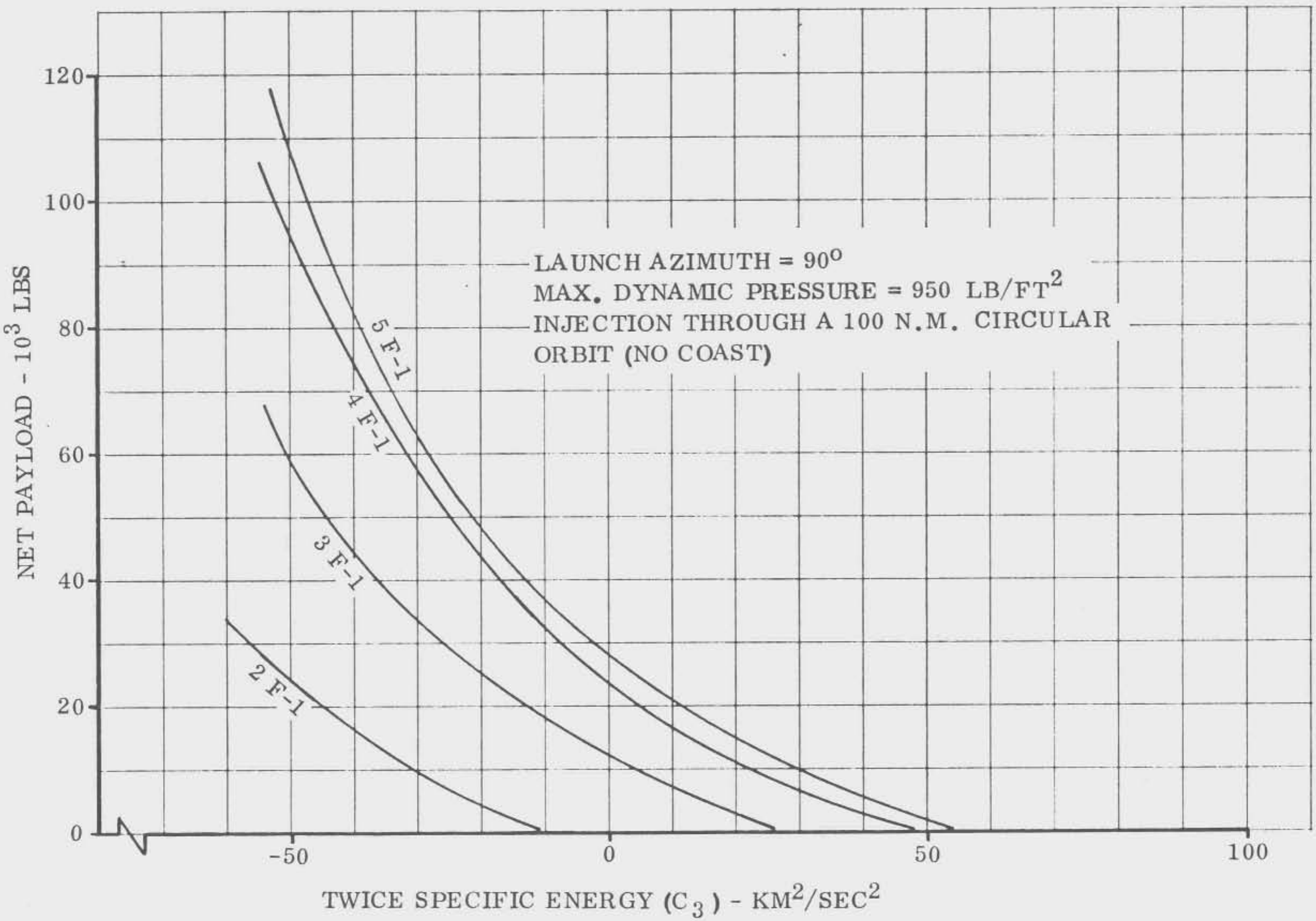


Figure 3.2.1.1-1 INT-20 PAYLOADS FOR LONGITUDINAL ACCELERATION LIMIT = 4.68 g

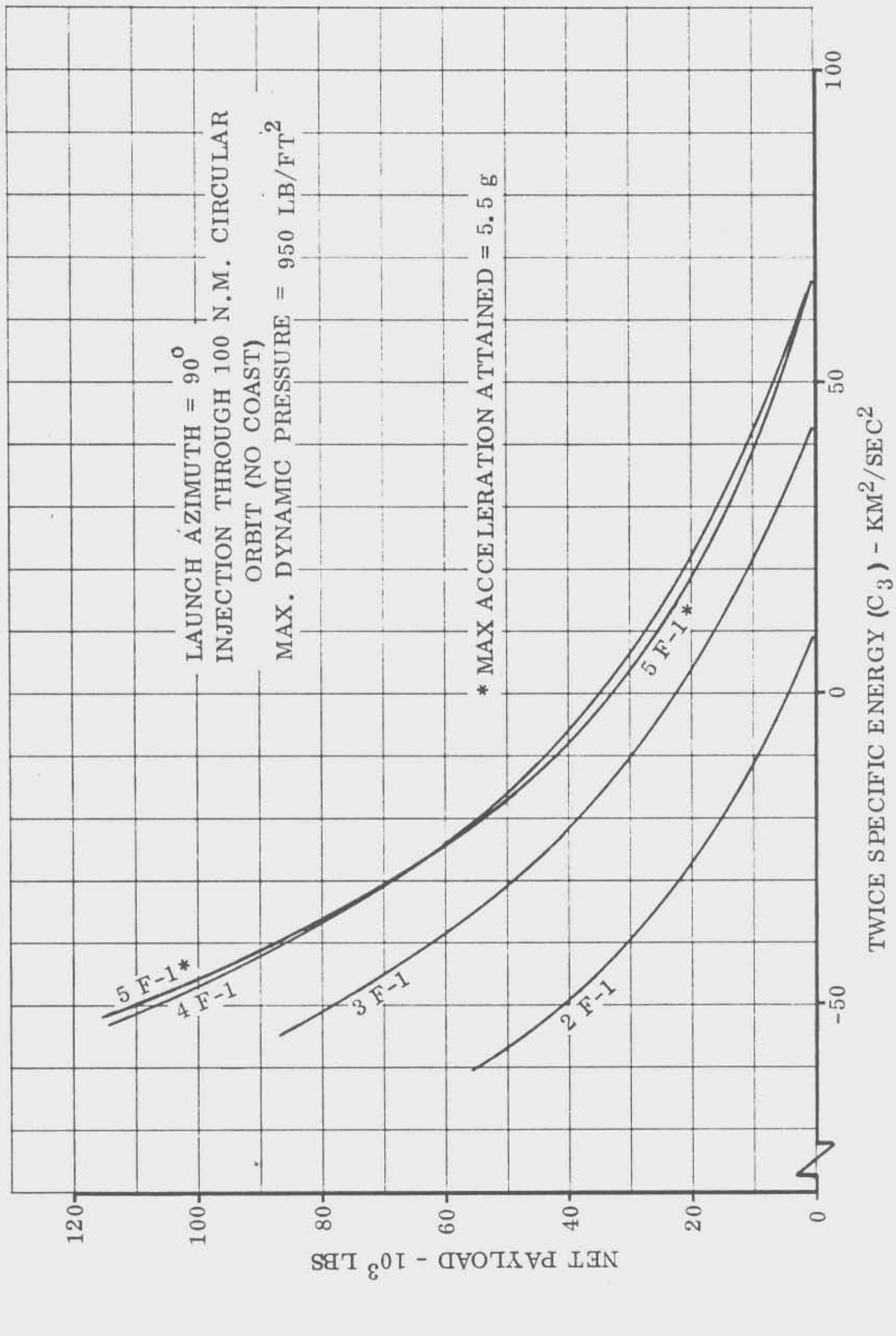


Figure 3.2.1.1-2 INT-20 PAYLOADS FOR LONGITUDINAL ACCELERATION LIMIT = 6.0 g

3.2.1.1 (Continued)

have about the same capability. In both cases, the 5 F-1 vehicle is penalized (flown on a non-optimum trajectory) to limit maximum dynamic pressure (q) to 950 lb/ft^2. The high q of this vehicle occurs because first stage propellant capacity is insufficient to provide the required ballast. The result is a lift-off thrust-to-weight ratio (T/W_o) higher than 1.25 (see Figure 3.2.1.1-3 for variation of T/W_o with C_3). A higher T/W_o causes the vehicle to have higher velocities at corresponding times of flight (thus, higher q's) than vehicles flown at T/W_o =1.25.

b. Adding Engines Versus Increasing Acceleration

The step difference in performance available by adding an engine to a 3 F-1 vehicle is greater than the difference obtained by increasing the 3 F-1 vehicles' acceleration to 6-g's. This is demonstrated in Figure 3.2.1.1-4. This comparison holds true for a 2 F-1 vehicle. However, the performance differential between a 4.68-g, 4 F-1 vehicle and the 6.0-g, 4 F-1 vehicle is more than the differential between the former and the 4.68-g, 5 F-1 vehicle. In this case the addition of an engine does not buy the performance that an increase in acceleration does. Figure 3.2.1.1-5 shows the difference in performance between the 4.68 and 6.0-g, 4 and 5 F-1 vehicles. Note that the 5 F-1 data are for a q-limited vehicle (per NASA ground rules). This q limit is established because spacecraft like Gemini and Apollo were designed for this limit. Allowing the 5 F-1 vehicle to fly maximum performance trajectories would give increases in payload at the expense of increased dynamic pressures (1100 - 1200 lb/ft^2).

3.2.1.2 Low-Earth Orbit Capability

The low-earth orbit capability of each configuration was determined for orbit altitudes of 100 through 300 nautical miles. The direct ascent, due-east launch capability for each configuration is shown in Figure 3.2.1.2-1. For the low-earth orbital missions, addition of engines provides greater capability than increasing acceleration. In the case of the 5 F-1 vehicle, which is restricted to 950 PSF max. dynamic pressure, increase in acceleration provides very little increase in payload. The maximum acceleration reached by the 5 F-1 vehicle is about 5 g's (300 N. M. orbit) because S-IC propellant depletion occurs before the maximum acceleration is reached. Two discontinuities are noted on the figure. These are on the 6-g, 2 F-1 and 4.68-g, 5 F-1 curves. In the first case, ballast propellant is required when the payload decreases below the point noted. The ballast prevents the vehicle from exceeding the desired maximum acceleration. For the 4.68 acceleration 5 F-1 INT-20, a break in the curve occurs at an altitude of approximately 250 miles. For altitudes higher than 250 nautical miles, the flight path of this configuration is such that limiting the longitudinal

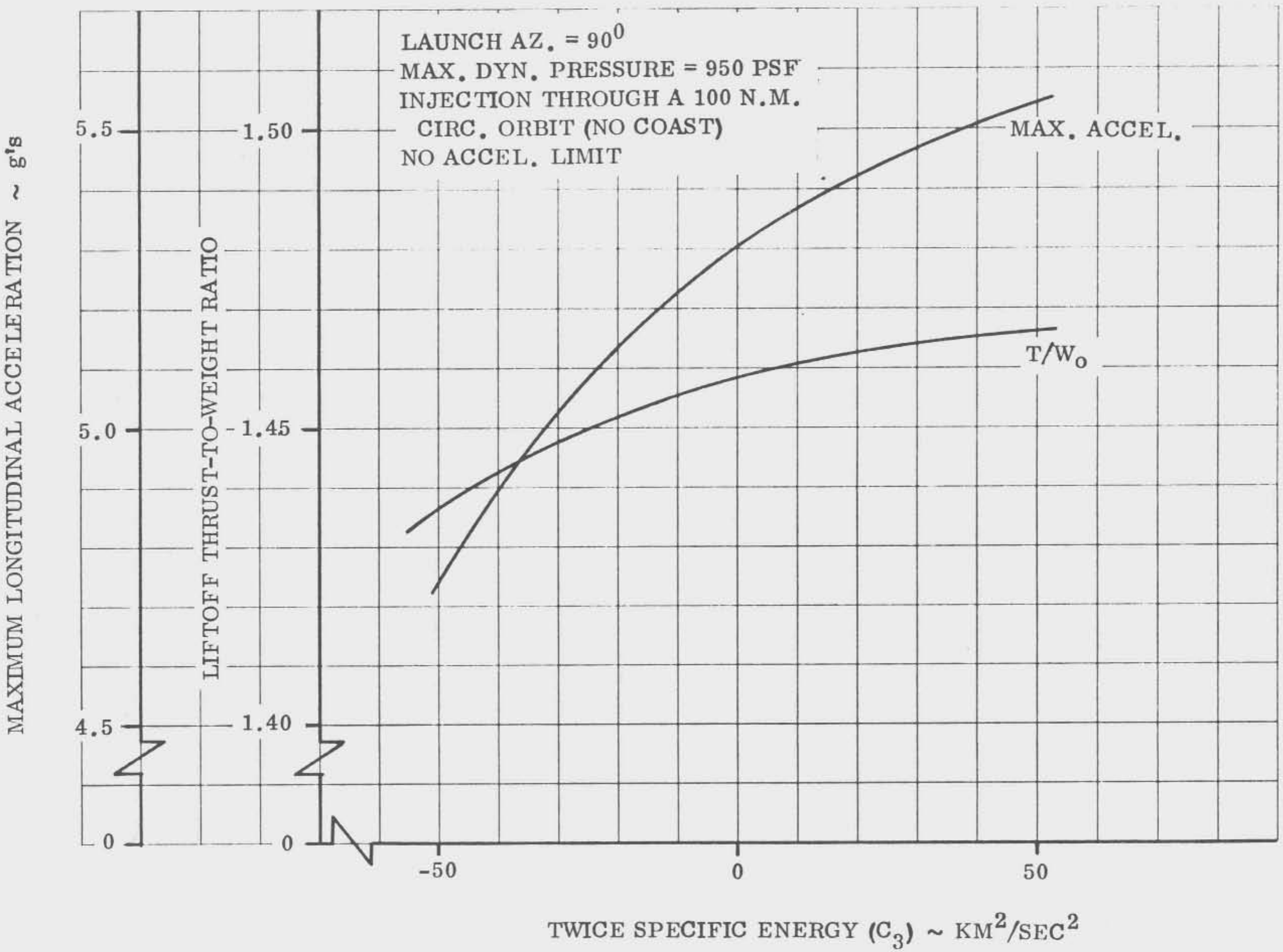


FIGURE 3.2.1.1 -3 5 F-1 INT-20 CHARACTERISTICS

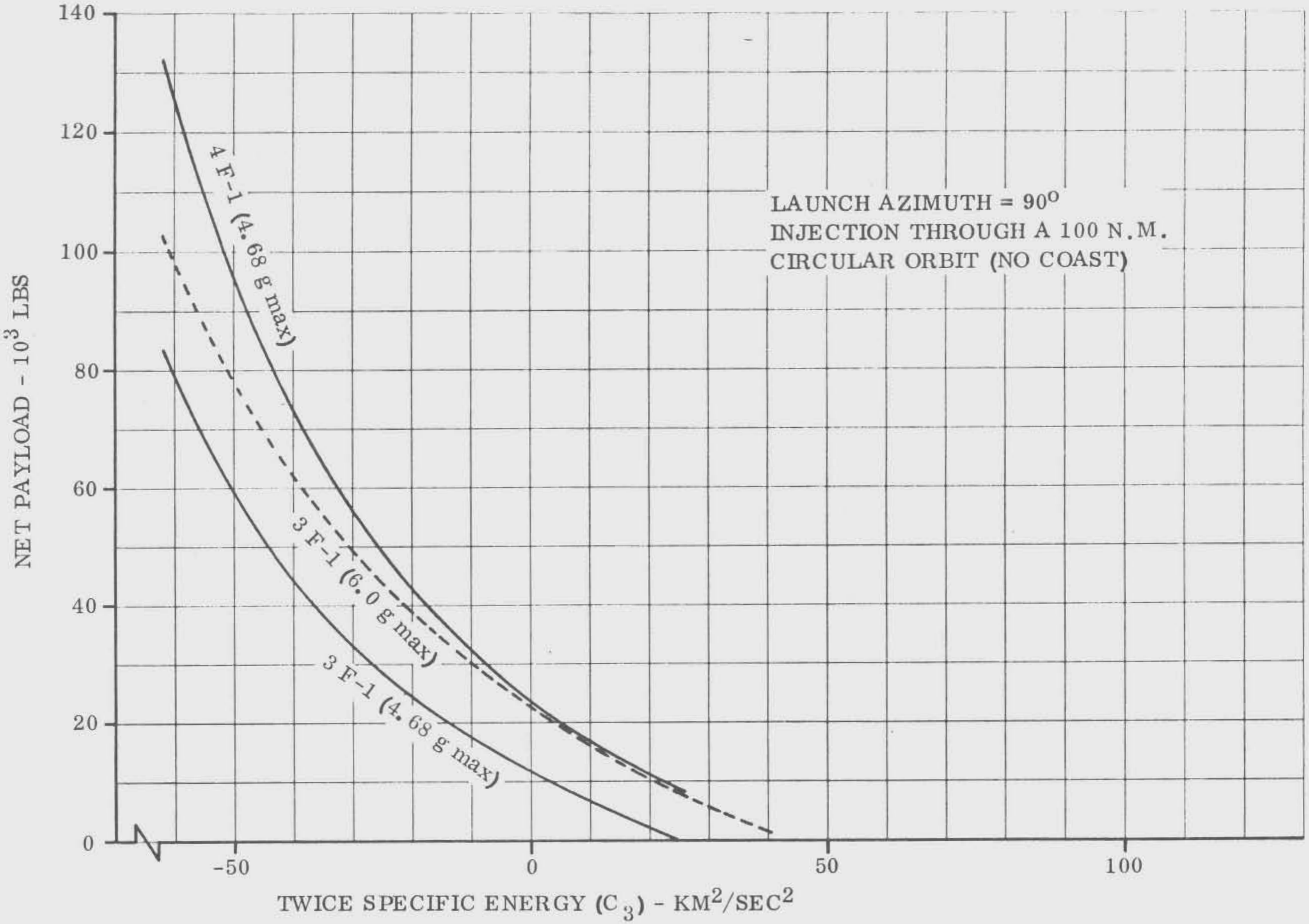


Figure 3.2.1.1-4 ADDING ENGINE VS. INCREASING ACCELERATION

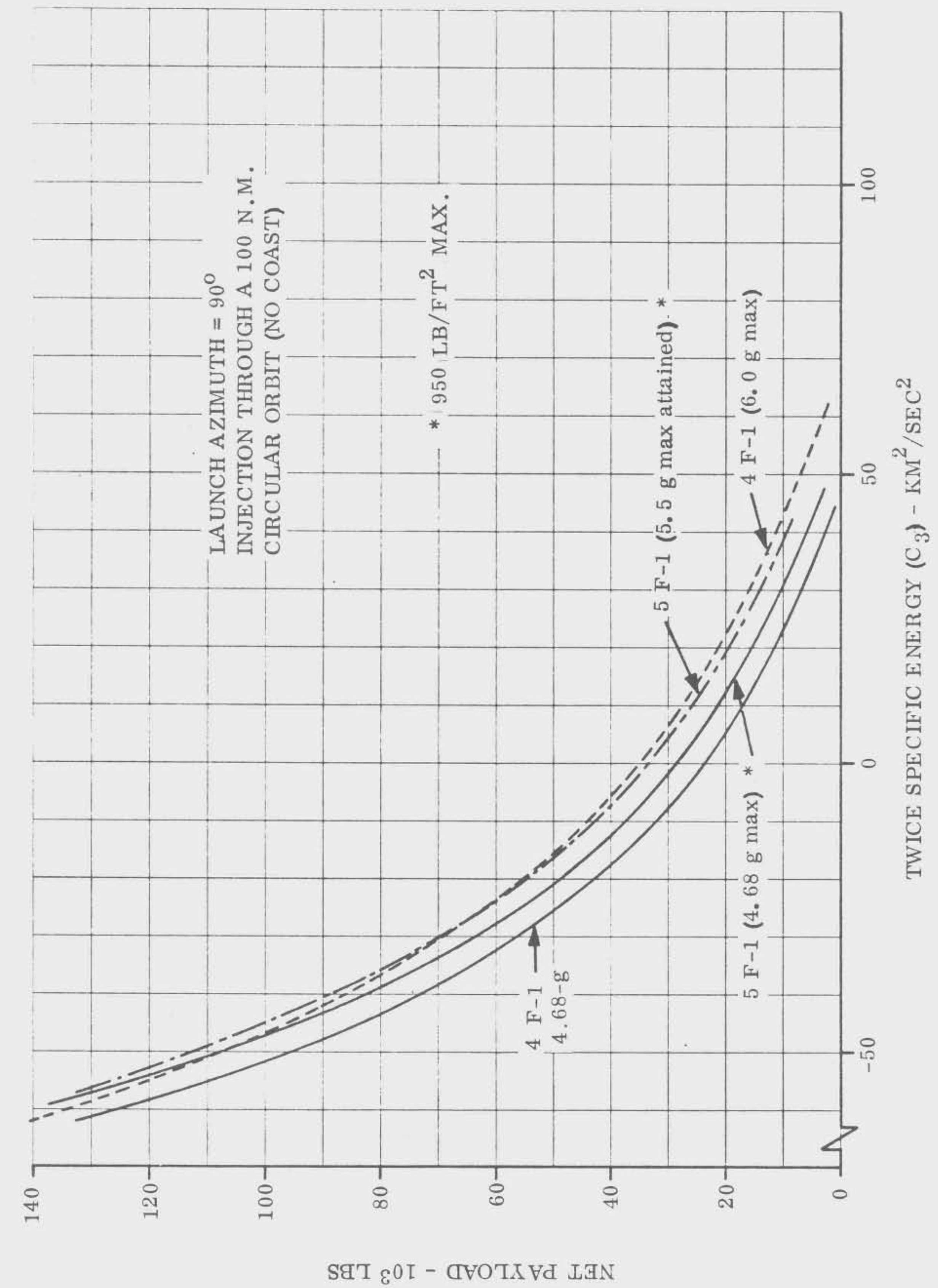


Figure 3.2.1.1-5
4 F-1 INT-20 COMPARISON WITH 5 F-1 VERSIONS

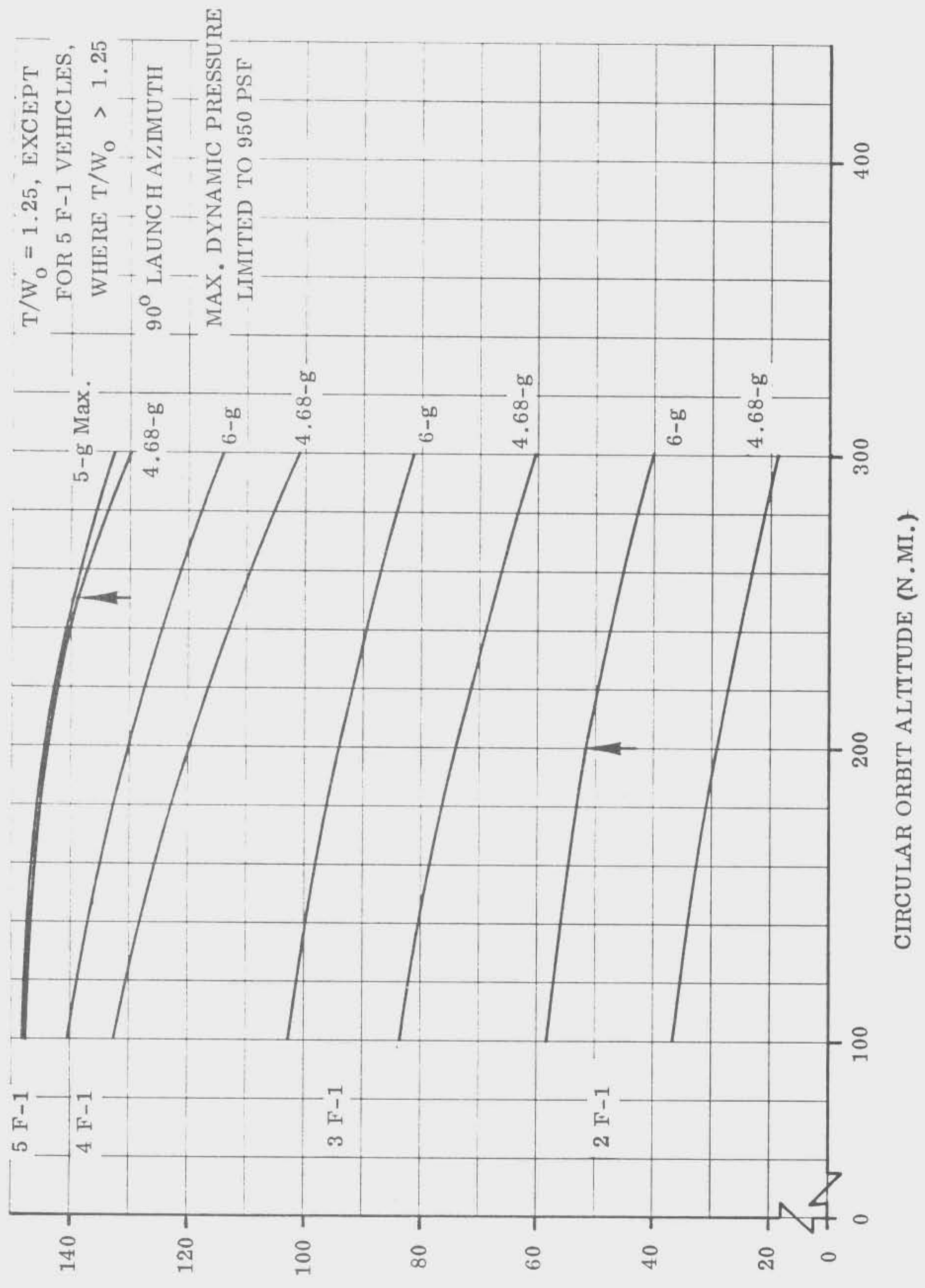


FIGURE 3.2.1.2-1 INT-20 CIRCULAR ORBIT CAPABILITY COMPARISONS

3.2.1.2 (Continued)

acceleration to 4.68 g's results in the maximum dynamic pressure being less than the 950 psf limit. For circular orbit altitudes less than 250 N.M., the maximum dynamic pressure is limited by a premature engine shutdown that occurs prior to when it would occur for acceleration control.

3.2.1.3 Synchronous Orbit Capability

The synchronous orbit performance capabilities of each INT-20 configuration were determined. Data are shown in Figures 3.2.1.3-1, -2, and -3 for the 3 F-1, 4 F-1 and 5 F-1 vehicles, respectively. Data are not shown for the 2 F-1 vehicle because it does not have a synchronous orbit mission capability. INT-20 synchronous mission capabilities are summarized below.

Configuration	Payload (lb.) at Orbit Inclination	
	28.5°	0°
3 F-1		
4.68-g max.	1,500	-
6.0-g	11,000	7,200
4 F-1		
4.68-g	11,900	7,700
6.0-g	21,100	16,400
5 F-1		
4.68-g	15,800	11,300
5.4-g	20,400	15,600

The payload capability values shown in this section are not necessarily the same as those read off the C_3 plots. The variation, if any, is due to the difference in the way both sets of data were calculated. For the C_3 data, the vehicle is flown to a preselected C_3, or energy level. For the synchronous data, the simulation more accurately corresponded to the way the real vehicle would fly the mission.

3.2.1.4 Polar Orbit Capabilities

Investigations of polar orbit capabilities of the INT-20 vehicle were limited in scope. One hundred through 300 N.M. circular orbit capability curves were prepared for the 4.68-g and 6-g, 4 F-1 vehicles. The 260 N.M. circular polar orbit capabilities of the other vehicles were estimated for comparative purposes. The available polar orbit data are shown in Figure 3.2.1.4-1.

The 2 F-1 vehicle polar orbit capability is quite limited. The capabilities of both 5 F-1 configurations are about the same as the 6-g, 4 F-1 vehicle.

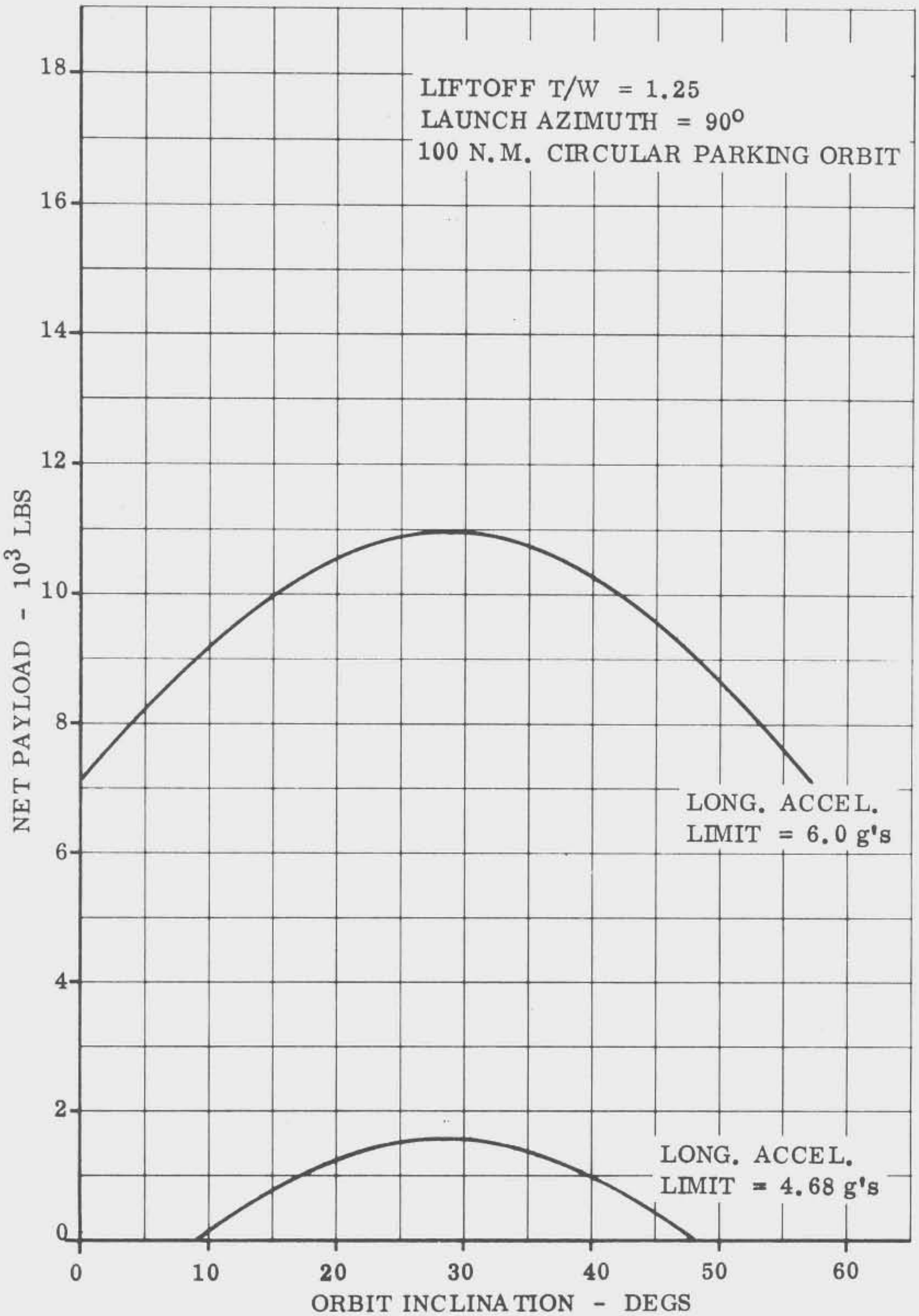


FIGURE 3.2.1.3-1 3 F-1 SYNCHRONOUS ORBIT CAPABILITIES

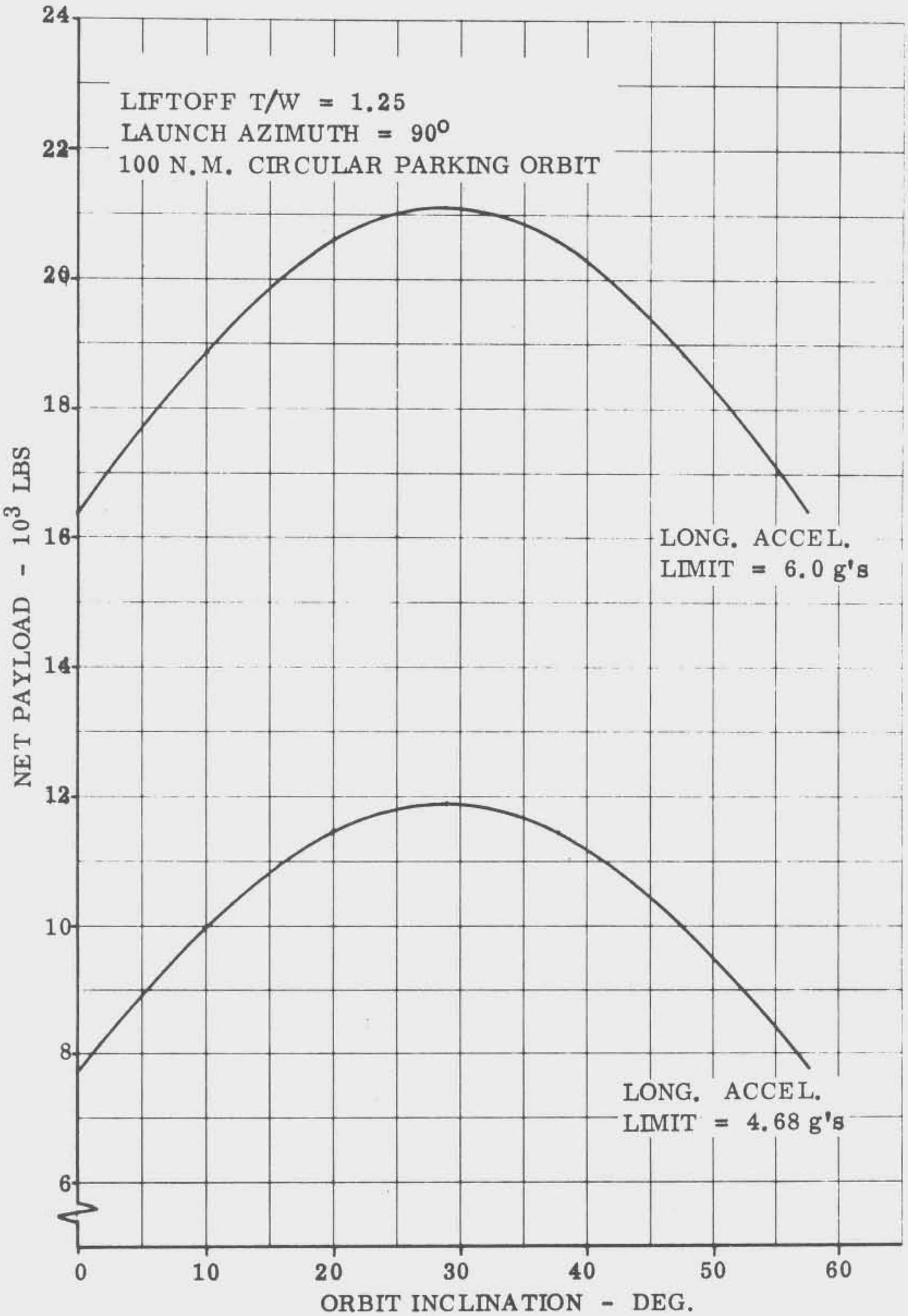


FIGURE 3.2.1.3-2 4 F-1 SYNCHRONOUS ORBIT CAPABILITIES

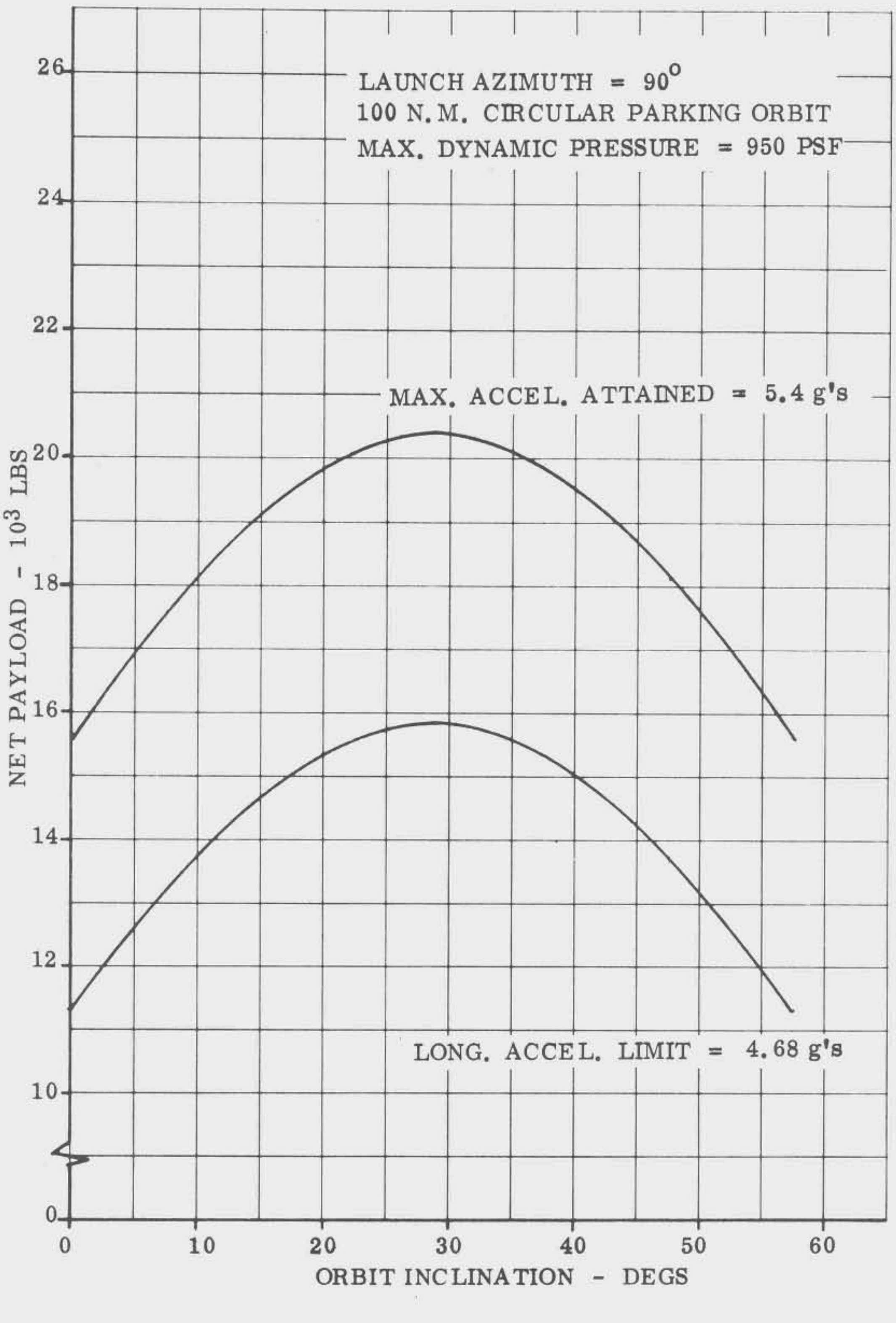


FIGURE 3.2.1.3-3 5 F-1 SYNCHRONOUS ORBIT CAPABILITIES

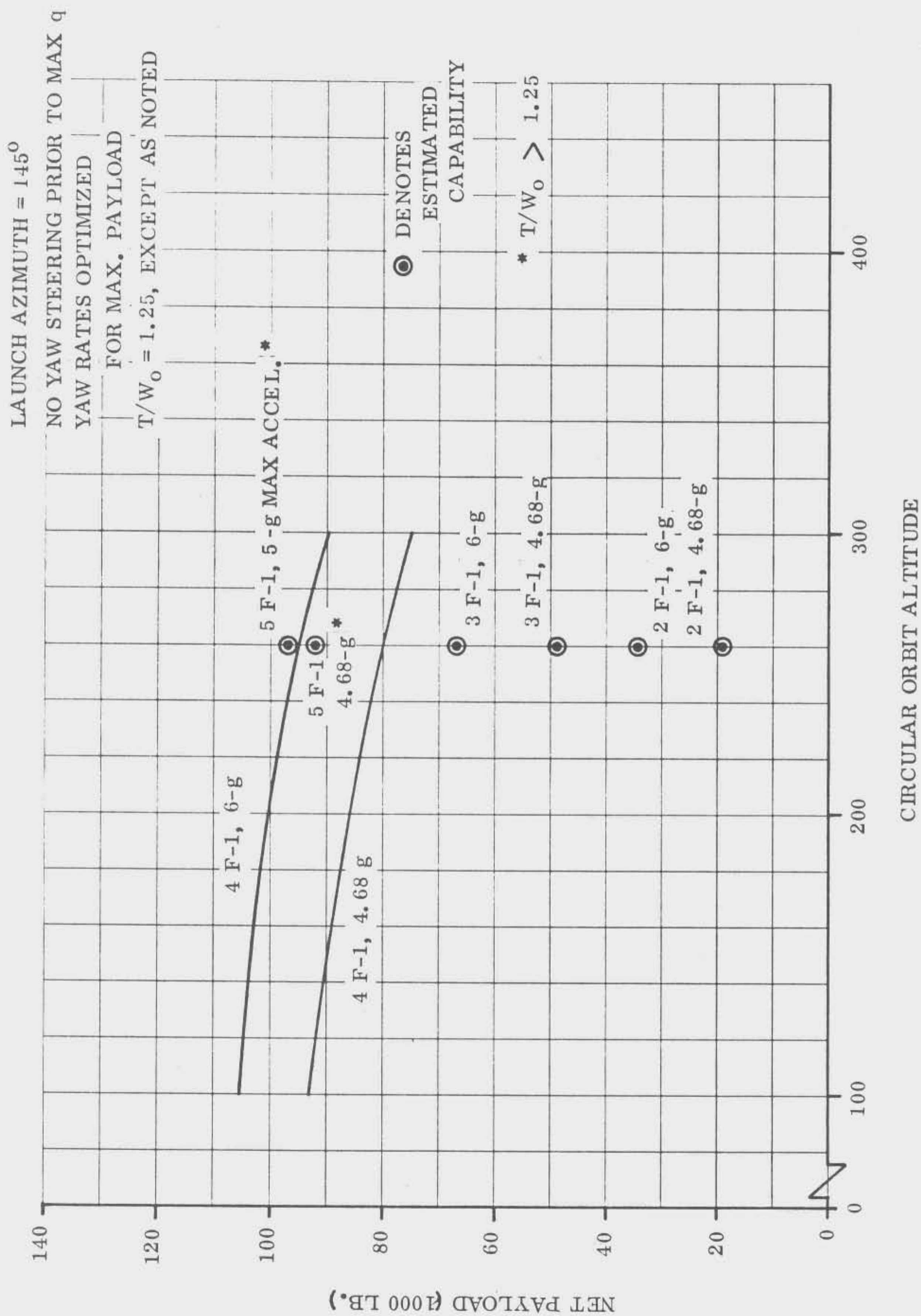


FIGURE 3.2.1.4-1 INT-20 POLAR ORBIT CAPABILITY

3.2.1.4 (Continued)

In all cases, yaw-steering to turn the vehicle from the 145° azimuth flight path into a polar orbit path is initiated after the maximum dynamic pressure time of flight to reduce loads and minimize effects on vehicle controllability. Specific yaw rates are needed for each configuration and each orbit altitude desired to insure minimum land mass overfly.

3.2.1.5 Performance with Injection Stages

The use of INT-20/injection stage performance was ruled out as a factor in baseline vehicle selection by NASA/MSFC. However, a limited analysis was made of the use of injection stages to enhance INT-20 performance. Data were prepared to demonstrate the increases available through the use of both the Centaur and Service Module Injection Stages (SMIS). Centaur data was provided by General Dynamics/Convair Division (see Ref. 3.2.1.5-1). The Centaur is enclosed in a shroud (based on the SLA) to minimize in-flight loads on the stage. The SMIS is the 4-tank, shrouded, independent version. Data on the SMIS was provided by the Space Division of North American Rockwell Corporation (References 3.2.1.5-2 and 3.2.1.5-3).

The general arrangement of each injection stage is depicted in Figure 3.2.1.5-1. The performance enhancement obtained through the use of injection stages is shown in Figure 3.2.1.5-2, for the 4.68, 4 F-1 vehicle, and in Figure 3.2.1.5-3, for the 6-g, 4 F-1 vehicle. Synchronous orbit payload increases due to use of the third stages is shown in Figure 3.2.1.5-4.

3.2.1.6 Unmanned Payloads

Baseline selection data has been generated under the guideline that the vehicle would be designed for manned application. A vehicle structurally-designed for the manned factor of safety (1.4) and a corresponding peak acceleration of 4.68 g's could be advantageously used for unmanned payloads. The maximum acceleration associated with the unmanned factor of safety (1.25) is about 5.25 g's. A vehicle flown at this acceleration level would have increased capability throughout its performance range, without the requirement for structural beef-up. Figure 3.2.1.6-1 shows the differential in capability between the 4.68-g, 5.25-g, and 6-g 4 F-1 INT-20 vehicles. Such increase in capability applies to any 4.68-g vehicle configuration designed for manned application and used for unmanned payloads. (Note - the data presented in figure 3.2.1.6-1 is for a 70° launch azimuth, rather than the 90° launch azimuth.)

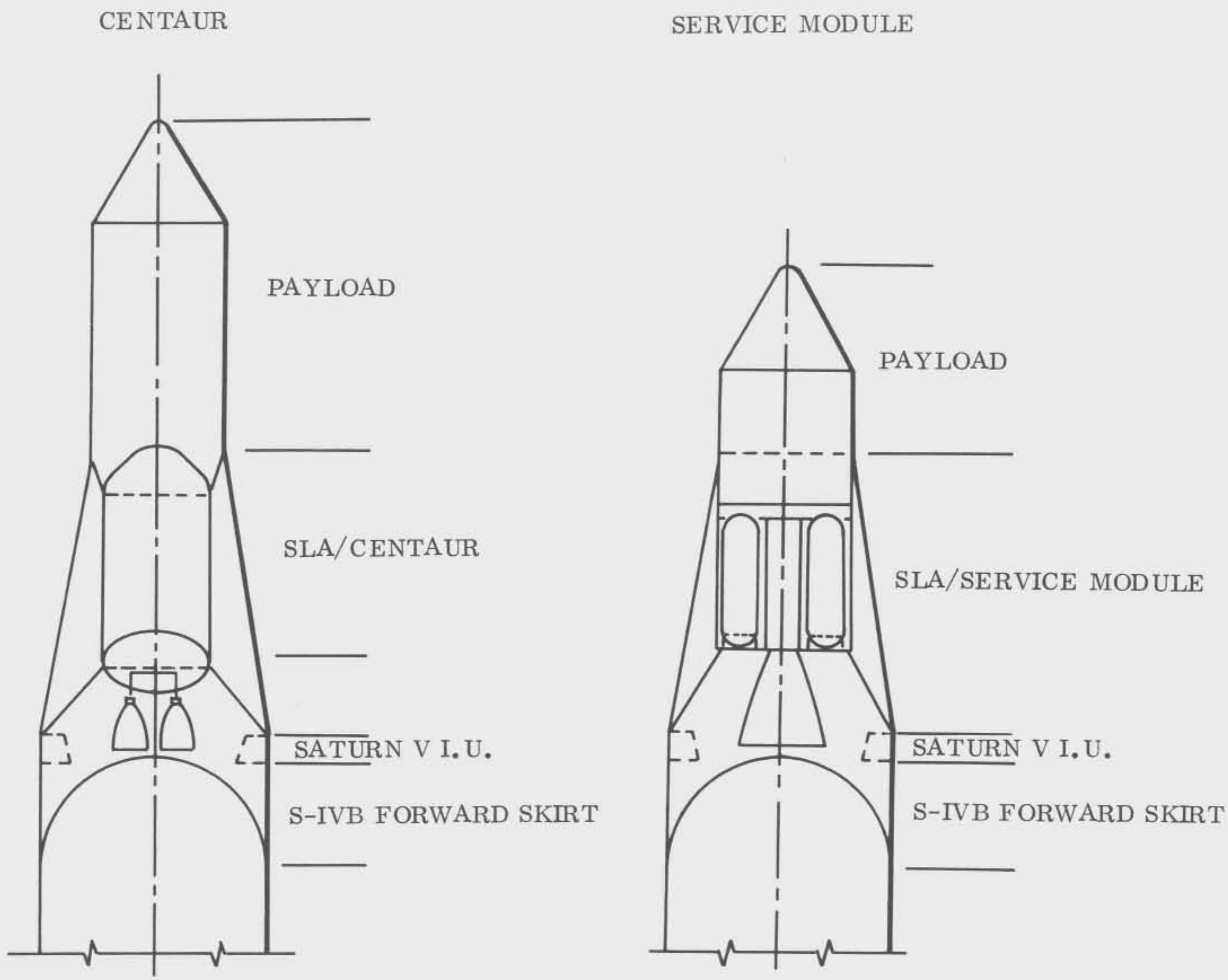


FIGURE 3.2.1.5-1 INJECTION STAGE SHROUD ARRANGEMENTS

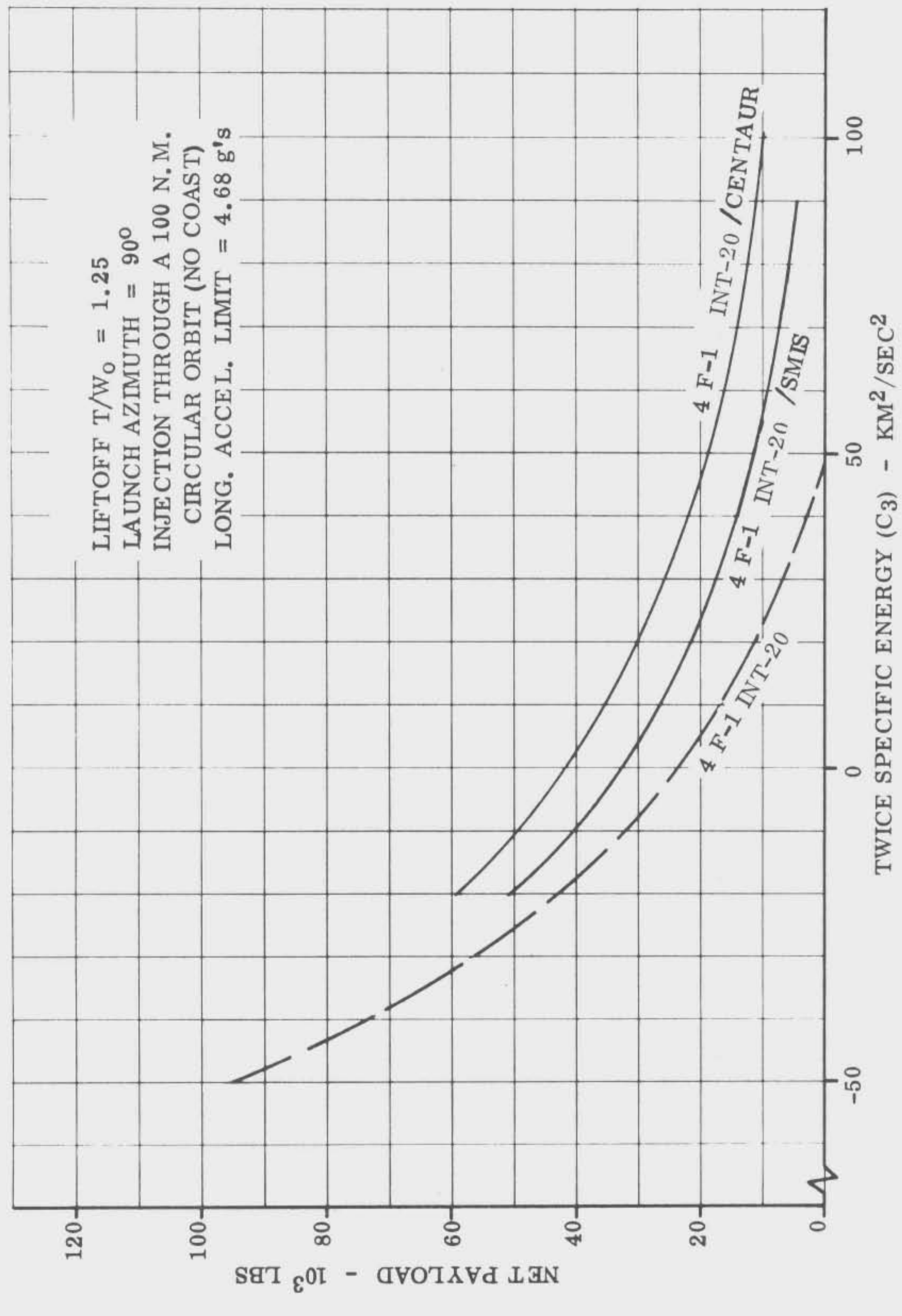


FIGURE 3.2.1.5-2 INT-20/INJECTION STAGE 4.68-g PERFORMANCE

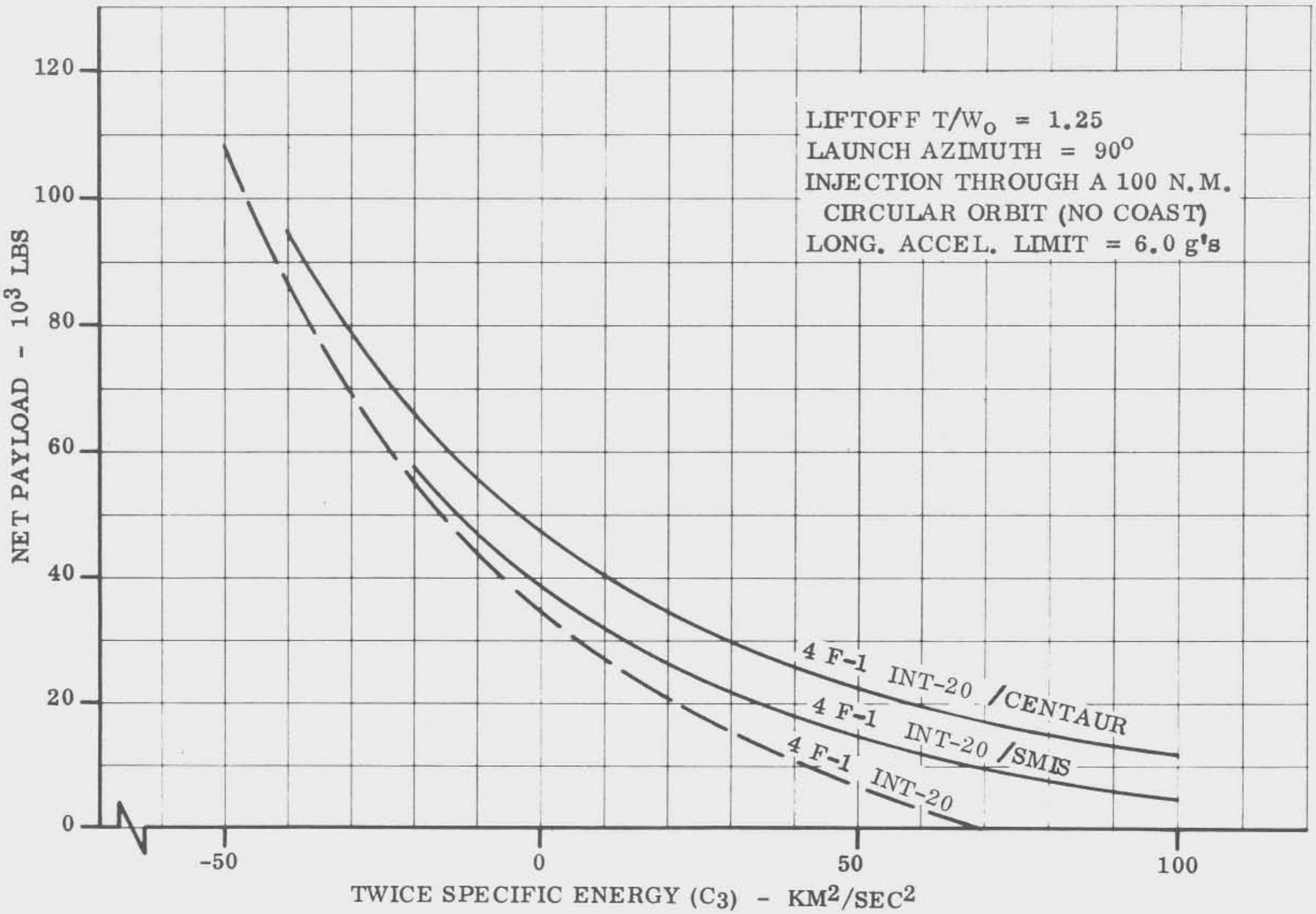


FIGURE 3.2.1.5-3 INT-20 INJECTION STAGE 6-g PERFORMANCE

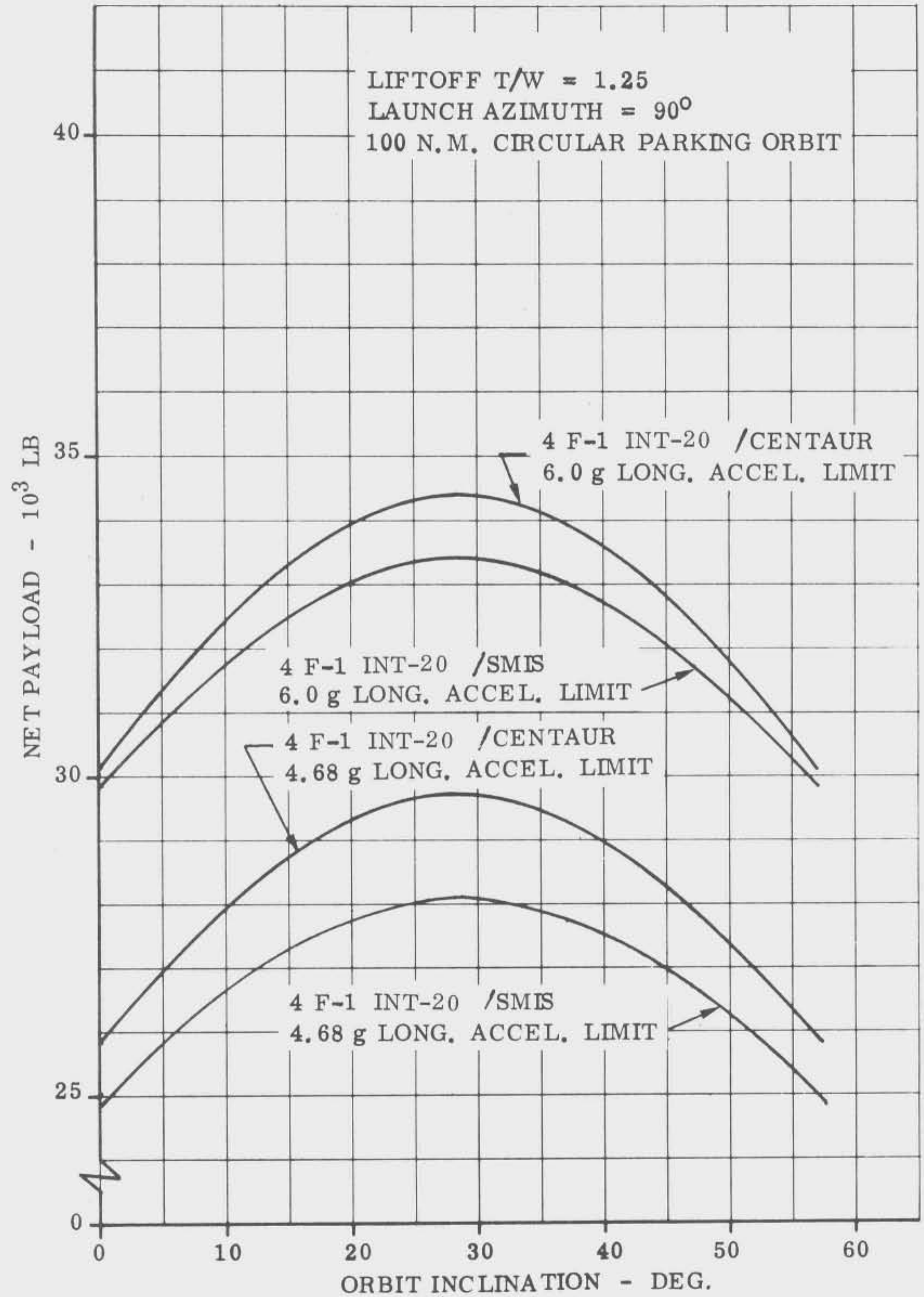


FIGURE 3.2.1.5-4 INT-20/INJECTION STAGE SYNCHRONOUS ORBIT PERFORMANCE

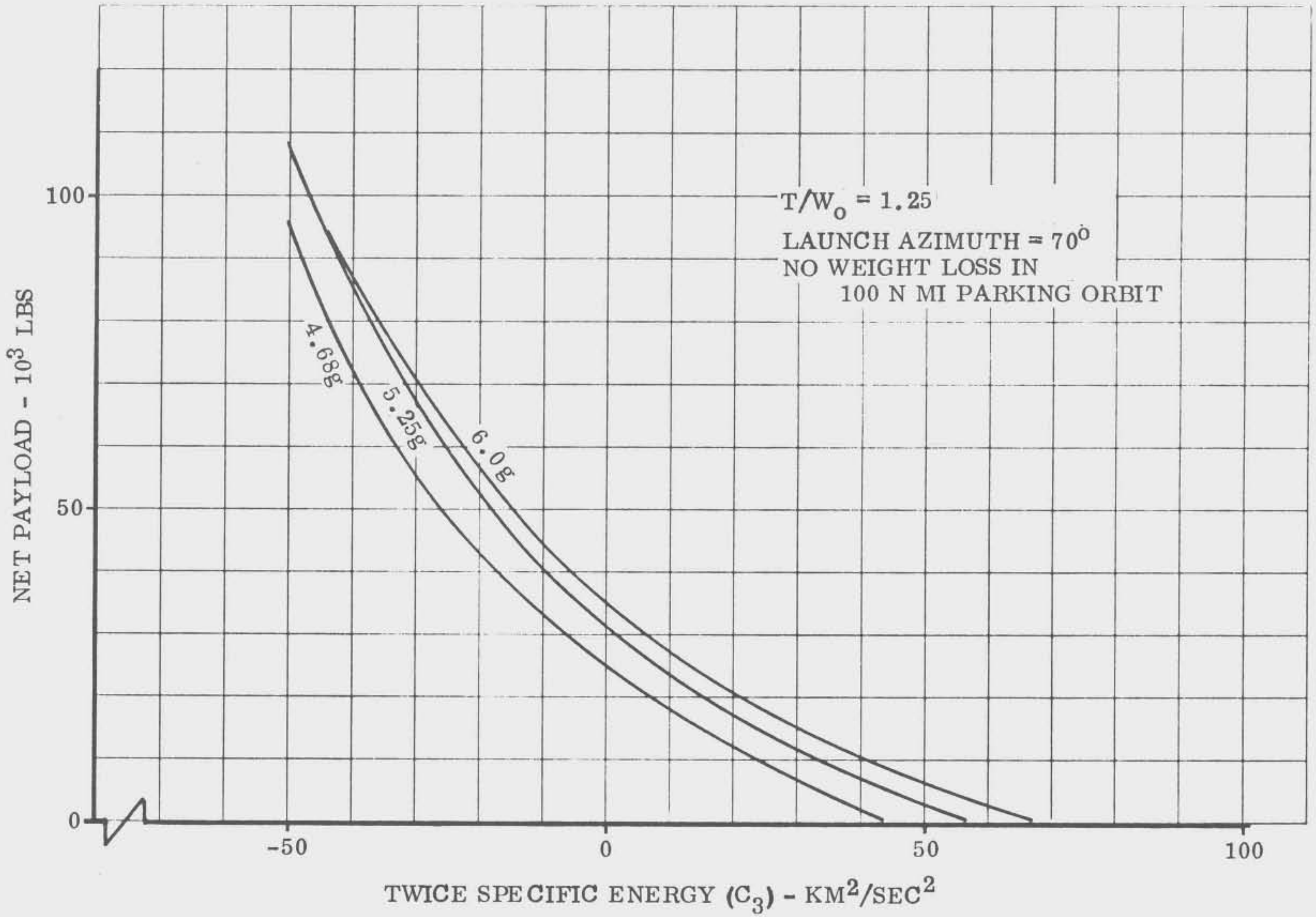


FIGURE 3.2.1.6-1 FOUR F-1 INT-20 FLIGHT PERFORMANCE

3.2.2 Technical Studies

Analyses were made to ascertain each INT-20 configuration's lift-off characteristics. Acoustic analyses were made to determine if the acoustics and vibration levels during flight were within specifications for each configuration. Payload/Wind sensitivity data were developed to show the range of allowable INT-20 payload lengths for the 4 F-1 vehicle.

3.2.2.1 INT-20 Tower Clearance

The two, three and four F-1 vehicles were analyzed to determined the amount of drift that each vehicle experienced at launch when under the influence of design ground winds and expected design/construction parameter tolerances (scatter). These preliminary studies used AS-516 weights and mass characteristics and an Apollo payload shape. The results showed that each of the configurations studied required use of a preprogrammed attitude command (like Saturn V) to avoid possible collision with the tower.

The study used a 95 percentile design surface wind speed envelope with an imbedded gust. The wind profile as a function of altitude is shown in Figure 3.2.2.1-1. Note that this wind is generally exceeded only during heavy rain showers, thunderstorms in the area or over the site, squall lines, some frontal passages, strong pressure gradients, and hurricanes. The peak wind, at point of maximum gust, is about 60 ft/sec.

Standard Saturn V scatter parameters were used. The scatter was combined additively and applied in conjunction with the wind to induce maximum vehicle drift towards the launch tower.

The attitude-attitude rate control mode was used for this study. This control mode actually contributes to drift. The combined effects of the wind, the scatter parameters, and the control mode made it necessary to use a programmed yaw maneuver to reduce vehicle drift to avoid collision with the tower.
The programmed yaw maneuver was initiated one second after liftoff and was increased at a rate of one degree per second until maximum commanded yaw (ψ_c) was reached. Then, beginning at eight seconds after liftoff, the command was reduced one degree per second until it again became zero. The effect was to tilt the vehicle away from the tower and into the disturbances. thus reducing drift toward the tower.

The results for the two, three, and four F-1 vehicle analyses are shown in Figures 3.2.2.1-2, 3.2.2.1-3, and 3.2.2.1-4, respectively. The figures show the trajectories of the standard vehicle reference point, the tip of Fin number 1. Figure 3.2.2.1-2 shows traces for the two F-1 vehicle under no-wind conditions (scatter only), scatter plus wind (commanded yaw of zero), and for maximum commanded yaws of 1.5^{o} and 2.0^{o}. The two F-1 vehicle, being the lightest of those studied, requires the largest commanded yaw to reduce the

3.2.2.1 (Continued)

induced drift. The three F-1 vehicle (Figure 3.2.2.1-3) requires only one degree of commanded yaw to clear the tower, while the four F-1 vehicle would require slightly less. The data shown for the four F-1 vehicle (Figure 3.2.2.1-4) is for a maximum commanded yaw of 1.5°. This data was taken from a previous study (Reference 3.2.2.1-1).

No data were prepared for the five F-1 vehicle because it was felt that this vehicle would be able to clear the tower without a preprogrammed yaw maneuver. The five F-1 vehicle lifts off at thrust-to-weight ratios greater than 1.25 (see Figure 3.2.1.1-3), which reduces the amount of drift experienced.

These analyses were made using an Apollo shape, which has a specific sail area. Use of a payload with a larger sail area would require somewhat larger maximum commanded yaw values to enable the vehicle to avoid impacting the tower under the assumed conditions.

3.2.2.2 Acoustics Analysis

An acoustics analysis was performed to establish the maximum overall sound pressure envelope expected for the INT-20 vehicle and to determine the maximum expected vibration environment. Since the maximum launch acoustics environment is experienced by the five F-1 vehicle, the analysis was limited to this configuration. The conclusions drawn for this case will hold for vehicles with fewer engines. The best estimate of the maximum acoustic environment to be expected at launch can be made by using the upper limit of external measurements taken along the Saturn V/Apollo during the SA-501 through SA-505 flights. This limit is plotted as a function of vehicle station, for two payload lengths, in Figure 3.2.2.2-1.

Configuration A uses an MLV cone only, and Configuration D an MLV cone plus a 70-foot long cylindrical section. As expected, the overall sound pressure levels were within the design specifications.

The near-field launch (lift-off) environment is the same as that for the Saturn V. The in-flight acoustic environment was determined by extrapolating data from the Saturn V AS-501 through AS-505 flights. The measured inflight data from these flights do not exceed the launch environment at critical vehicle areas such as the S-IVB/IU and S-II Forward Skirt/ S-IVB aft interstage regions. The flight time histories for the data show that the inflight environment may equal but doesn't exceed the launch environment at these locations.

Component vibration is basically proportional to the acoustic excitation. The excitation is a function of sound pressure level, frequency and wavelength matching with structural frequencies, and time of duration of the incident field. The maximum acoustic environment of the INT-20 launch vehicle is slightly greater for the S-IVB in the INT-20 configuration than for the Saturn V/Apollo configuration. Some requalification of components may be expected. A thorough statistical analysis of flight and launch data will be necessary to specify these in detail.

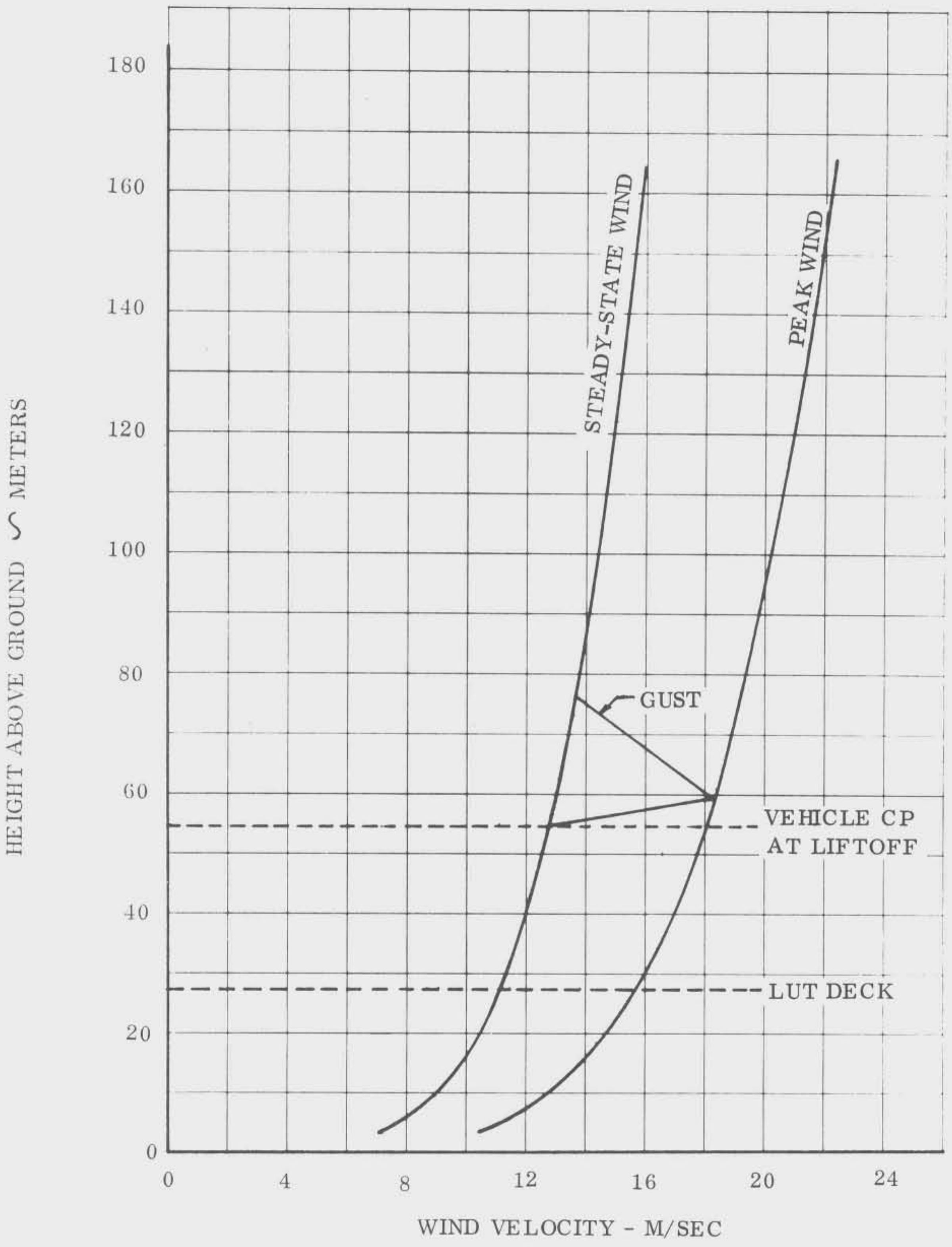


FIGURE 3.2.2.1-1 95 PERCENTILE GROUND DESIGN WIND WITH GUST

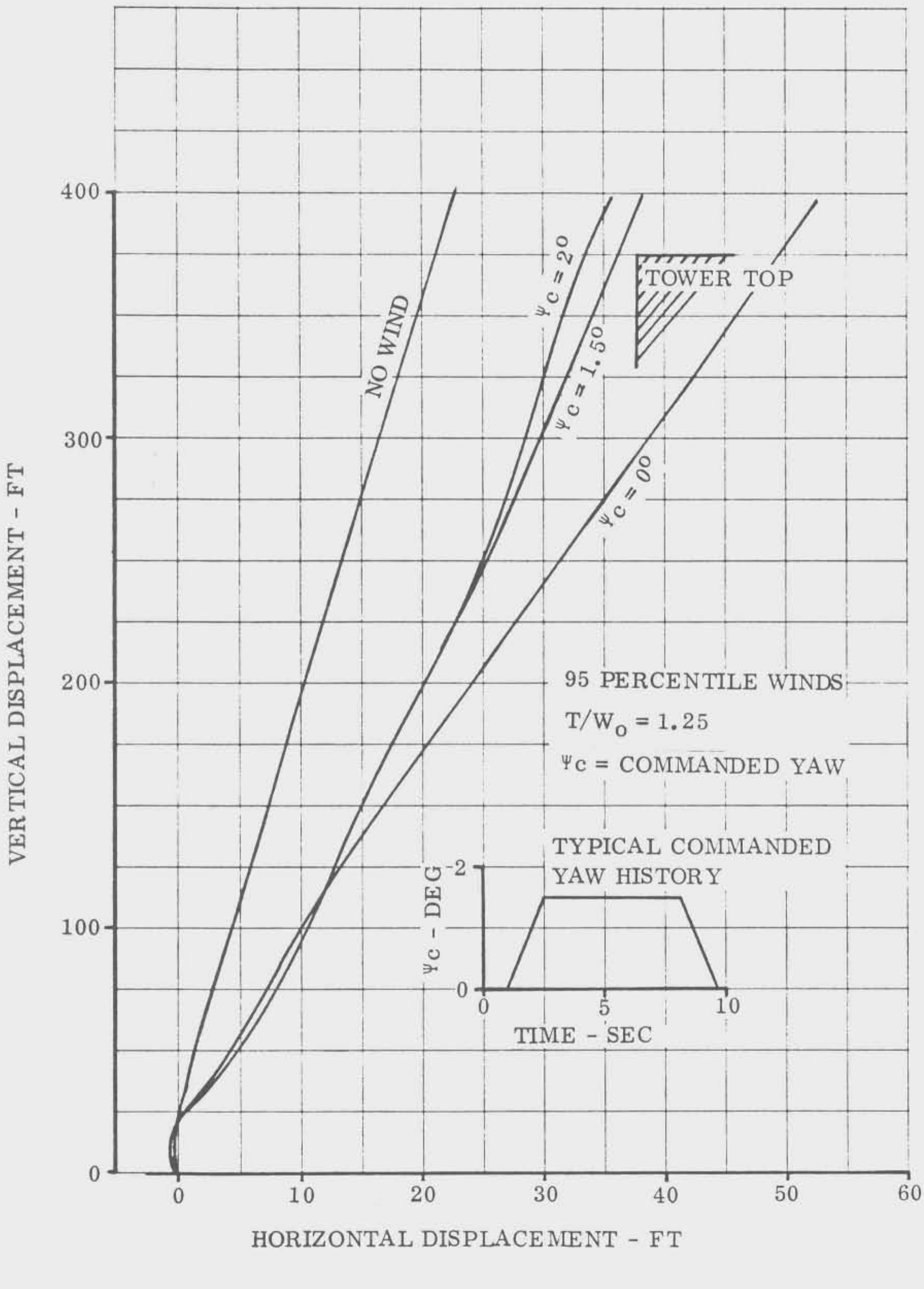


FIGURE 3.2.2.1-2 INT-20, 2 ENGINE VEHICLE FIN TIP TRAJECTORY

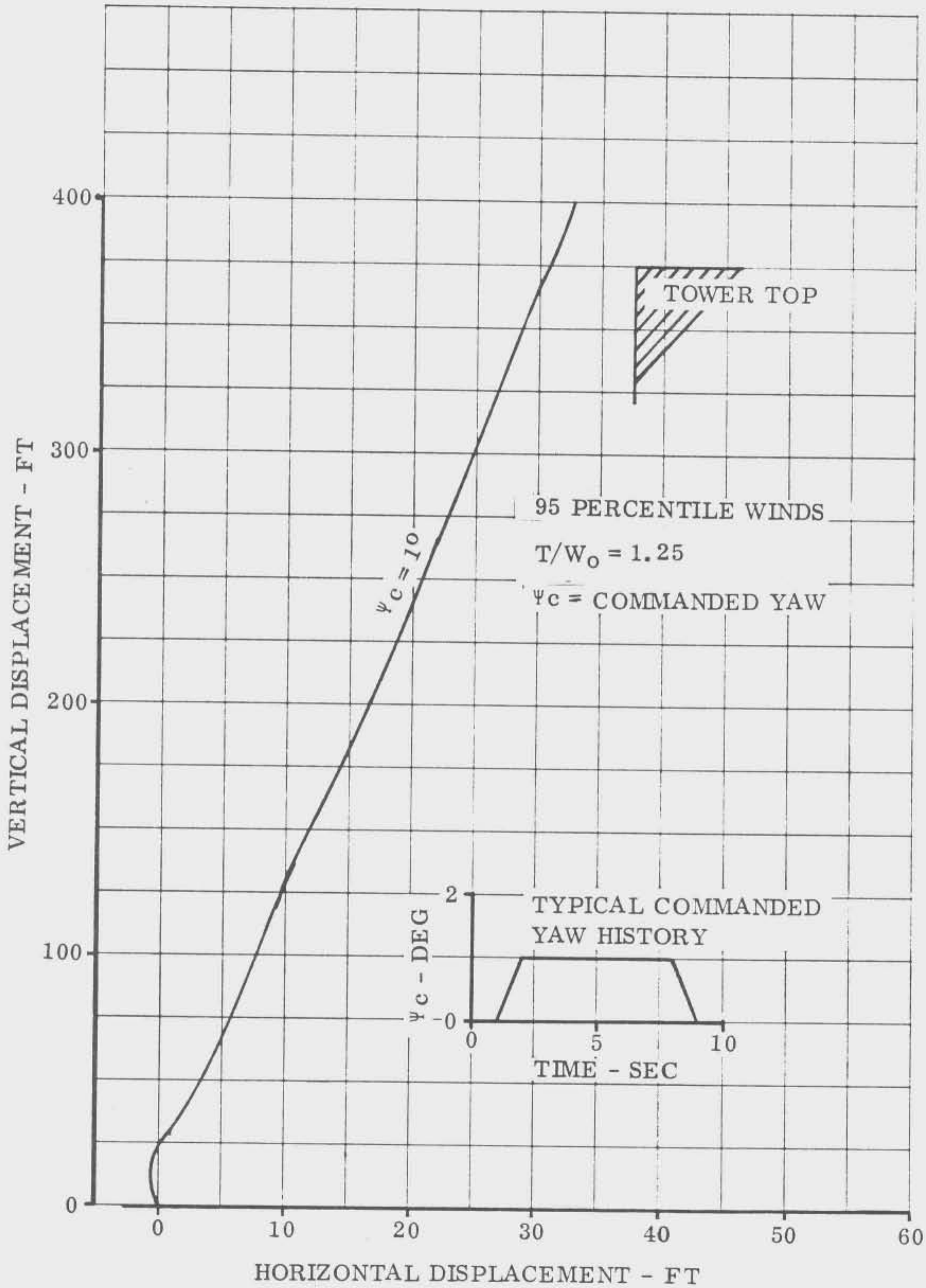


FIGURE 3.2.2.1-3 INT-20, 3 ENGINE VEHICLE FIN TIP TRAJECTORY

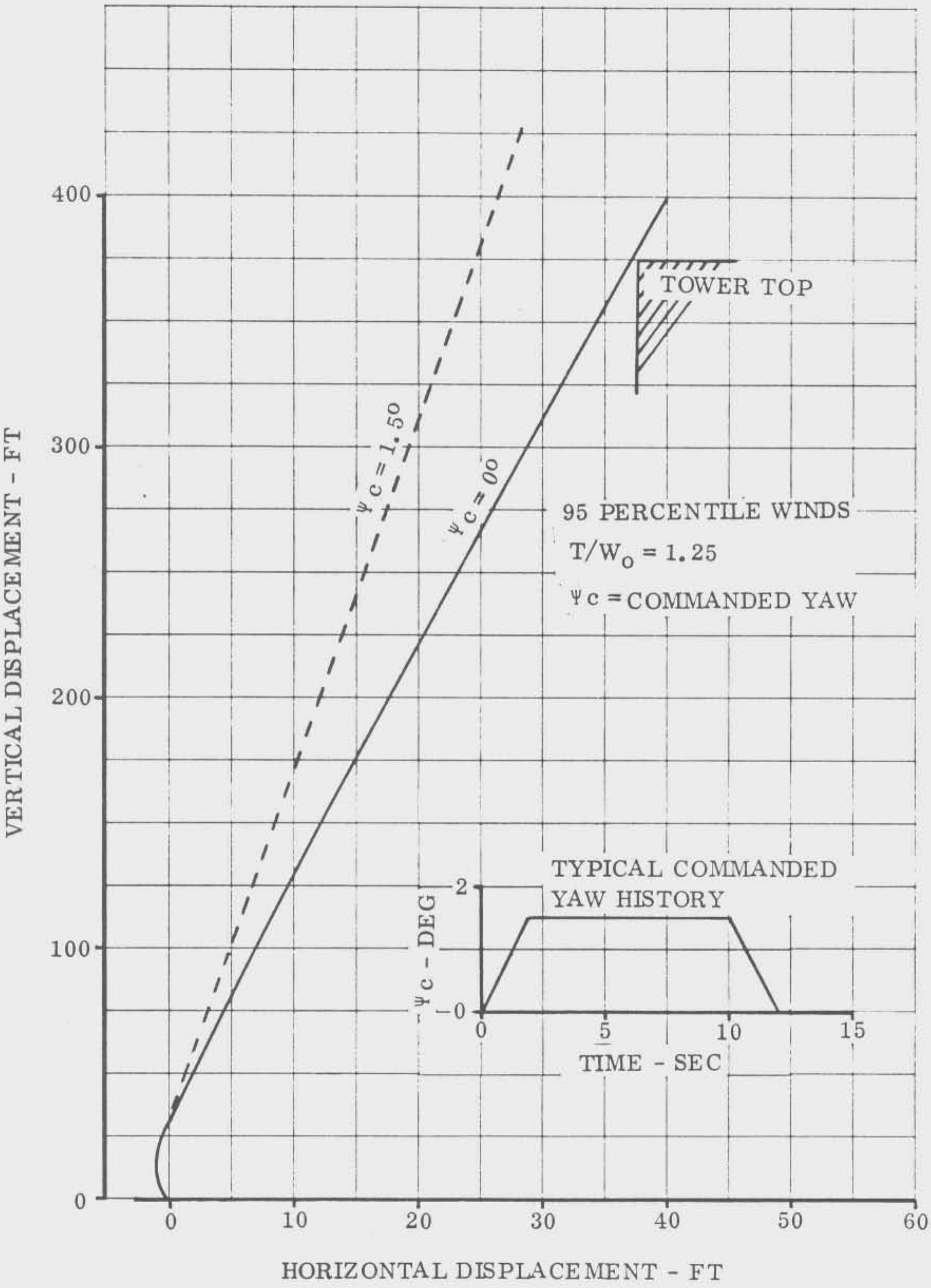


FIGURE 3.2.2.1-4 INT-20, 4 ENGINE VEHICLE FIN TIP TRAJECTORY

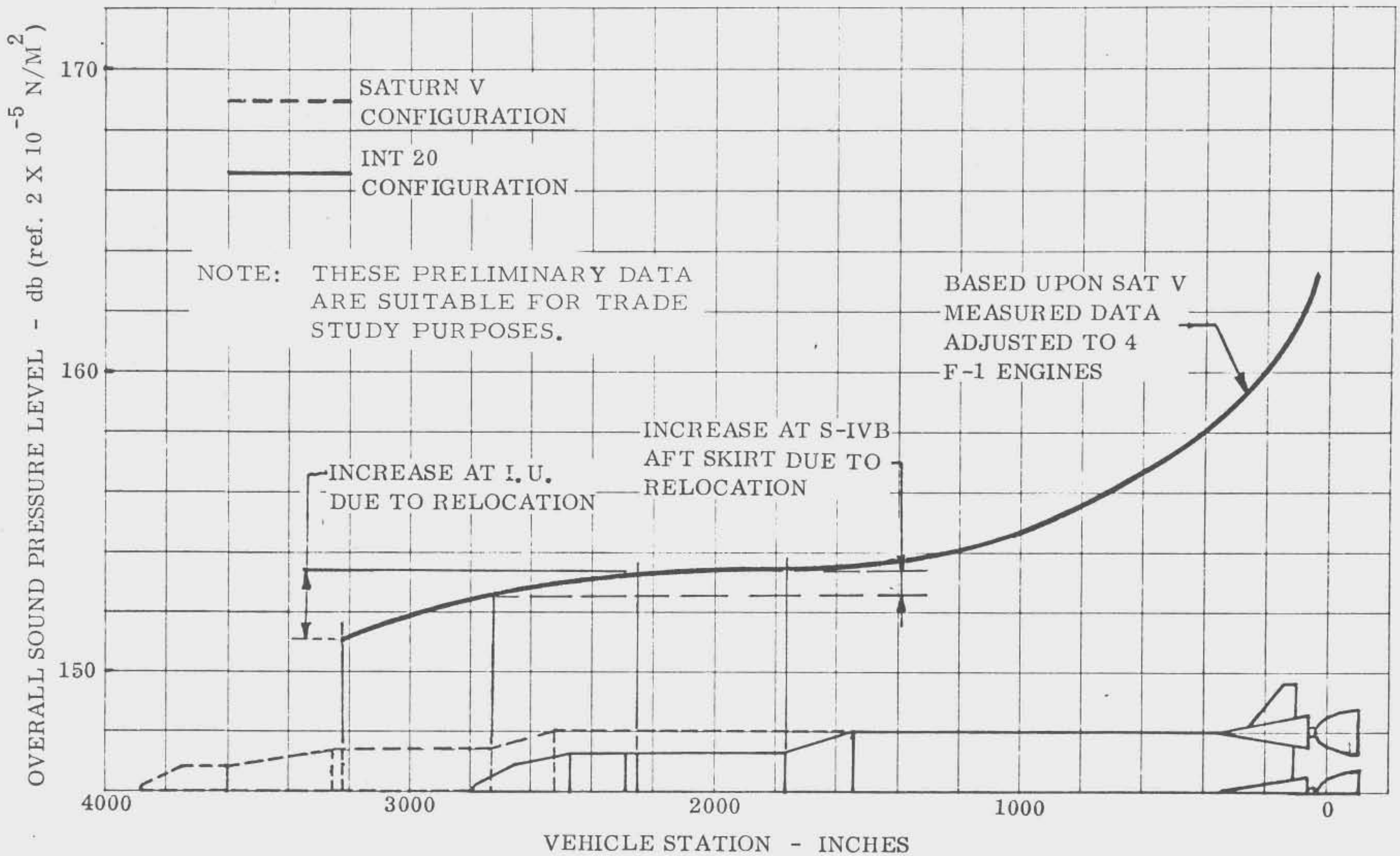


FIGURE 3.2.2.2-1 ACOUSTIC ENVIRONMENT - SAT-INT/20 WITH 4 F-1 ENGINES AND MLV NOSE

3.2.3 Development Requirements

Phase I analyses showed that implementation of any of the proposed INT-20 launch vehicle configurations would not require any major development programs (and associated risks.) Systems Engineering and Integration tasks will be required, as they are for Saturn V, and minimal development is associated with these. The Phase I assessment of the development requirements for implementing an INT-20 configuration is discussed below.

3.2.3.1 Development Tests

No major qualification or verification test program requirement was identified during Phase I. Tests applicable to the INT-20 were reviewed with respect to current test philosophy. The conclusions reached for the major tests are discussed below.

a. Man-Rating Flights

In view of the current success of both the Saturn IB and Saturn V flight programs, and considering that the INT-20 uses basic, flight-proven Saturn hardware, it was deemed that special man-rating flights for any of the INT-20 configurations were unnecessary.

b. Dynamic Tests

It was shown in the Saturn V dynamic test program that accurate engineering predictions could be made of Saturn stage/vehicle dynamic responses. It was concluded that this capability also existed for INT-20 and that dynamic tests would not be necessary if payloads had the same general dynamic characteristics as the existing Apollo payloads. If the new payload significantly affected overall vehicle structural dynamic responses, tests may be required of at least the S-IVB/IU/payload stack. Since payload characteristics are unknown at this time, it was assumed for trades analysis purposes that dynamic tests would not be required.

c. F-1 Engine Tests

The Rocketdyne Division of North American Rockwell Corporation has recommended that a verification test be performed to demonstrate the full-duration operation capability of the F-1 engine. This requirement was discussed in Section 3.1.1.4.

3.2.3.2 Systems Engineering and Integration (SE&I) Tasks

There are four SE&I activities: System Integration; Systems Engineering; Technology; and Launch Vehicle and Mechanical Ground Support Equipment (LVMGSE). Each area is affected by implementation of an INT-20 launch vehicle configuration. The maximum amount of development effort is concentrated in only a few tasks.

a. System Development Facility

The Systems Development Facility (SDF, or "Breadboard") is located at MSFC. It is a functional replica of the launch vehicle systems and associated GSE. It simulates electrically the functions of the vehicle systems and subsystems, from ground checkout through to orbit insertion (or whenever the IU completes its function.) The SDF must be modified to eliminate the S-II stage from the set-up. These changes are mainly electrical patch-work to ESE panels, wiring changes, and some plumbing changes to reflect removal of F-1 engines.

b. Interface Engineering

The interface engineering task is to maintain control of stage-to-stage, stage-to-engine, vehicle-to-GSE, and vehicle-to-facility interfaces. Development effort is required for researching, defining, and documenting the new INT-20 interfaces.

c. Flight Evaluation

In flight evaluation, pre-flight trajectory data are prepared, comparisons made between the flight and preflight trajectories, and anomalities in flight trajectory parameters are identified. For the INT-20, changes must be made to both digital and hybrid simulator computer programs.

d. Propulsion System Analysis

This task covers propulsion systems performance predictions, flight evaluation, environmental control studies, and structural heating studies. Changes will be required to software to reflect the INT-20 configuration.

e. Structural System Analysis

This task covers flight loads and mass analysis, ground winds, structural dynamics, vibration and acoustics data, structural design accuracy, and flight evaluation. Changes will be required to existing computer programs.

3.2.3.2 (Continued)

f. Instrumentation System Analysis

This task covers telemetry and RF systems analysis. Some effort will be required for researching and defining INT-20 instrumentation requirements.

3.2.4 Cost Data

The total development (non-recurring) costs for each vehicle configuration were developed during the trades effort by combining stage costs and adding estimated SE&I costs. It was found that each configuration required about the same amount of development dollars.

Variations between the recurring costs of each configuration were found to be essentially related to the number of F-1 engines required (about $3M per engine).

It should be noted that the total development costs estimated for each configuration during the Phase I analysis were too high. This was determined during the Phase III resources analysis (see Section 5.0). To avoid possible confusion between the Phase I and Phase III estimates, the Phase I total vehicle development costs have not been shown.

Stage cost data developed during Phase I are contained in the respective stage analysis sections.

3.3 VEHICLE COMPARISONS

The individual stage analyses and the vehicle analyses showed that it was feasible to implement all INT-20 variations. The desirability of a particular configuration (or configurations) was discovered to be highly dependent upon specified mission requirements (payload weights, flight environment restrictions), with development requirements and costs being only small factors.

3.3.1 Mission Requirements

The manned logistics support of Earth-orbit space stations was identified by NASA as the most probable application of the INT-20 intermediate vehicle. Available information on postulated space stations showed that station orbits were generally circular, at altitudes of 200 to 300 NM, and with orbit inclinations of from 28.5° to 90° (polar orbit). The logistics package weights for space station support have been generally quoted to be in the 60K to 120K lb range. Spacecraft proposed for use on space station logistics flights included Command Module and Gemini

3.3.1 (continued)

(and their derivatives). These spacecraft have been designed for maximum dynamic pressures of up to 950 lb/ft^2.

It was assumed that the INT-20 would also be required to provide lunar base unmanned logistics support. For these missions, the required lunar-landed payload would be at least on the order of 3,000 to 5,000 lb. Using modifications of existing spacecraft and injection stages, the landing of a few thousand pounds of payload on the lunar surface would require that a launch vehicle be able to inject at least 30,000 lb. into a translunar trajectory.

3.3.2 Performance Comparisons

The first criterion for comparing the performance of each INT-20 launch vehicle configuration is the Earth-orbit space station logistics mission requirement. Since the exact space station support requirements are not known, the 260 NM polar orbit space station was selected as the basis of comparison. Figure 3.3.2-1 shows the 260 NM circular polar orbit mission capabilities of each configuration. As noted in Section 3.3.1, above, logistics packages are expected to weigh between 60K and 120K lb. The 4.68-g 2 F-1 vehicle, the 6-g 2 F-1 vehicle, and 4.68-g 3 F-1 vehicle do not meet the performance requirements of this mission.

A second criterion that was established for performance comparisons is the lunar resupply mission. Figures 3.3.2-2 and 3.3.2-3 show the translunar trajectory mission capabilities of 4.68-g and 6-g INT-20 configurations, respectively. These data show that only the 6-g 4 F-1, 4.68-g 5 F-1, and 5.5-g 5 F-1 vehicles meet this criterion. As was shown in Section 3.2.1.6, however, the 4 F-1 vehicle can operate unmanned at an acceleration level of 5.25-g's to increase its capability without additional structural modification. Referring to Figure 3.3.2-4, it is evident that the 5.25-g 4 F-1 vehicle meets the criterion for unmanned lunar logistics support missions.

A third criterion evolving from the mission support requirements was the necessity to limit dynamic pressures to less than 950 lb/ft^2 for manned missions. The 5 F-1 vehicles must be flown on specially-tailored trajectories to limit their max q to 950 lb/ft^2. On this basis, the 5 F-1 vehicles become less desirable than any of the 4 F-1 vehicles (4.68, 5.25, or 6.0-g configurations) for the manned mission support task.

3.3.3 Development Requirements

There is no significant difference between the development cost requirements of any of the candidate configurations. Recurring (or production) costs would vary in relation to the number of F-1 engines on the S-IC stage (about $2.3M per engine plus $.7M for engine related hardware).

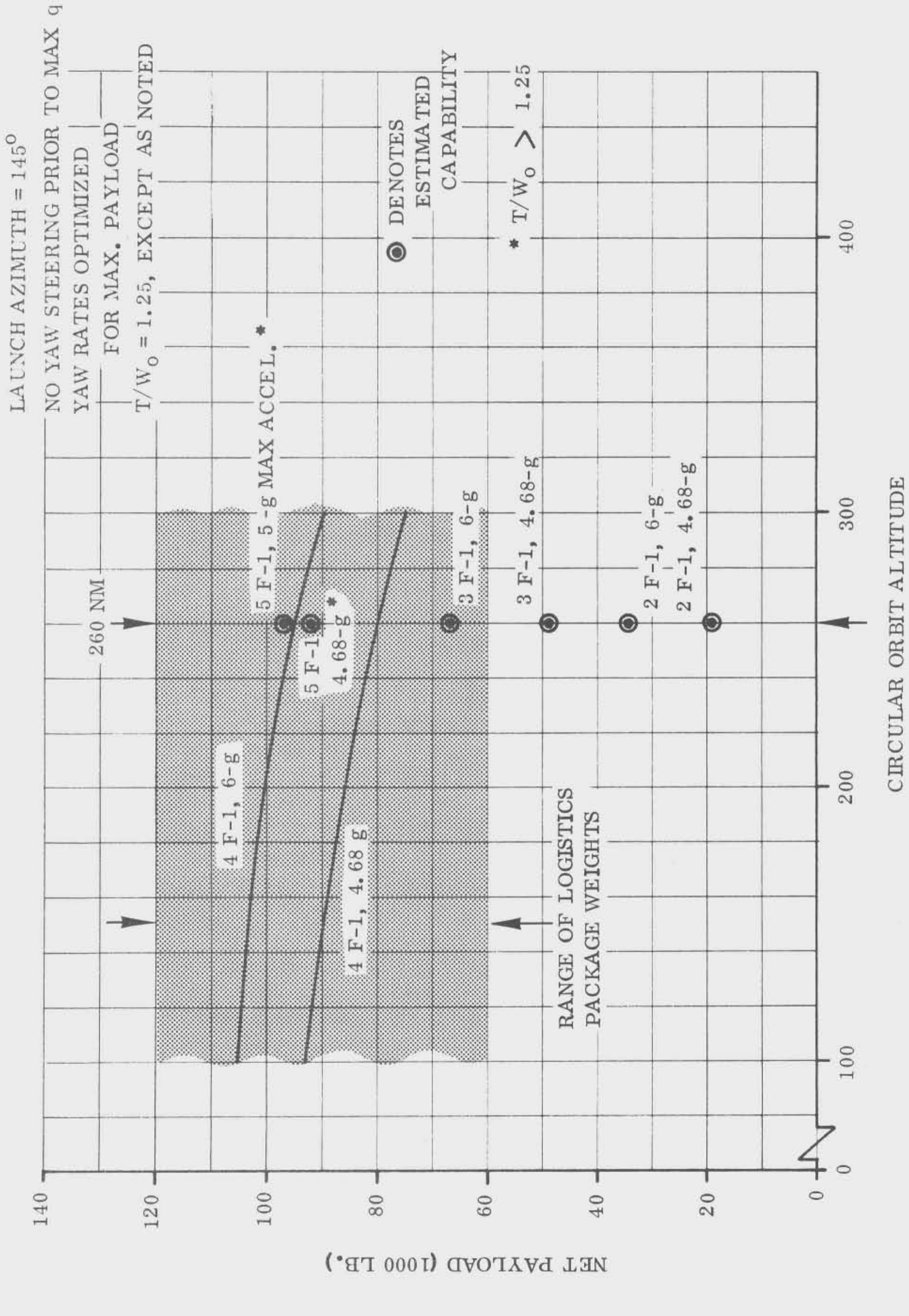


FIGURE 3.3.2-1 INT-20 POLAR ORBIT CAPABILITY COMPARISONS

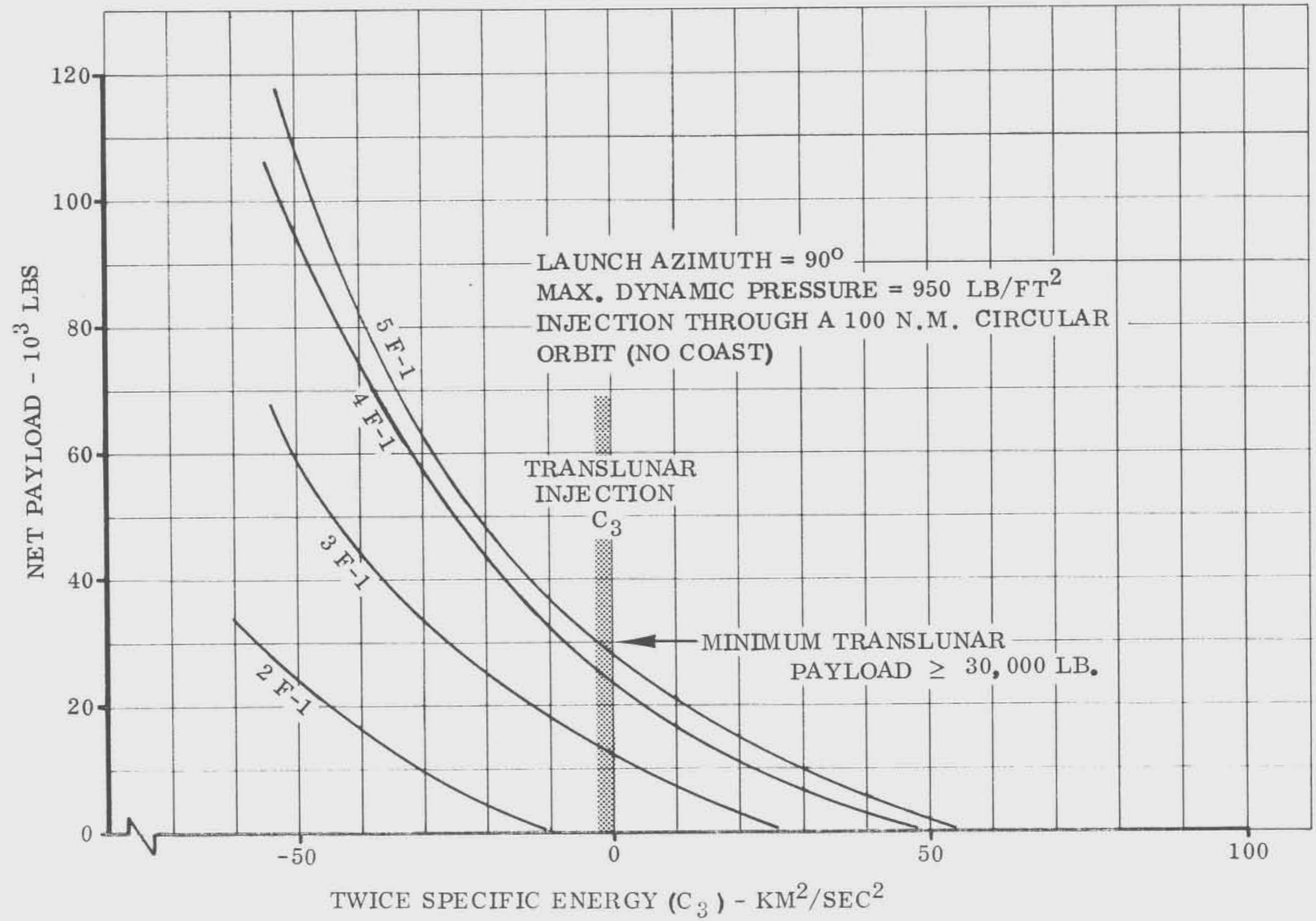


FIGURE 3.3.2-2 COMPARISON OF 4.68-g VEHICLE LUNAR CAPABILITIES

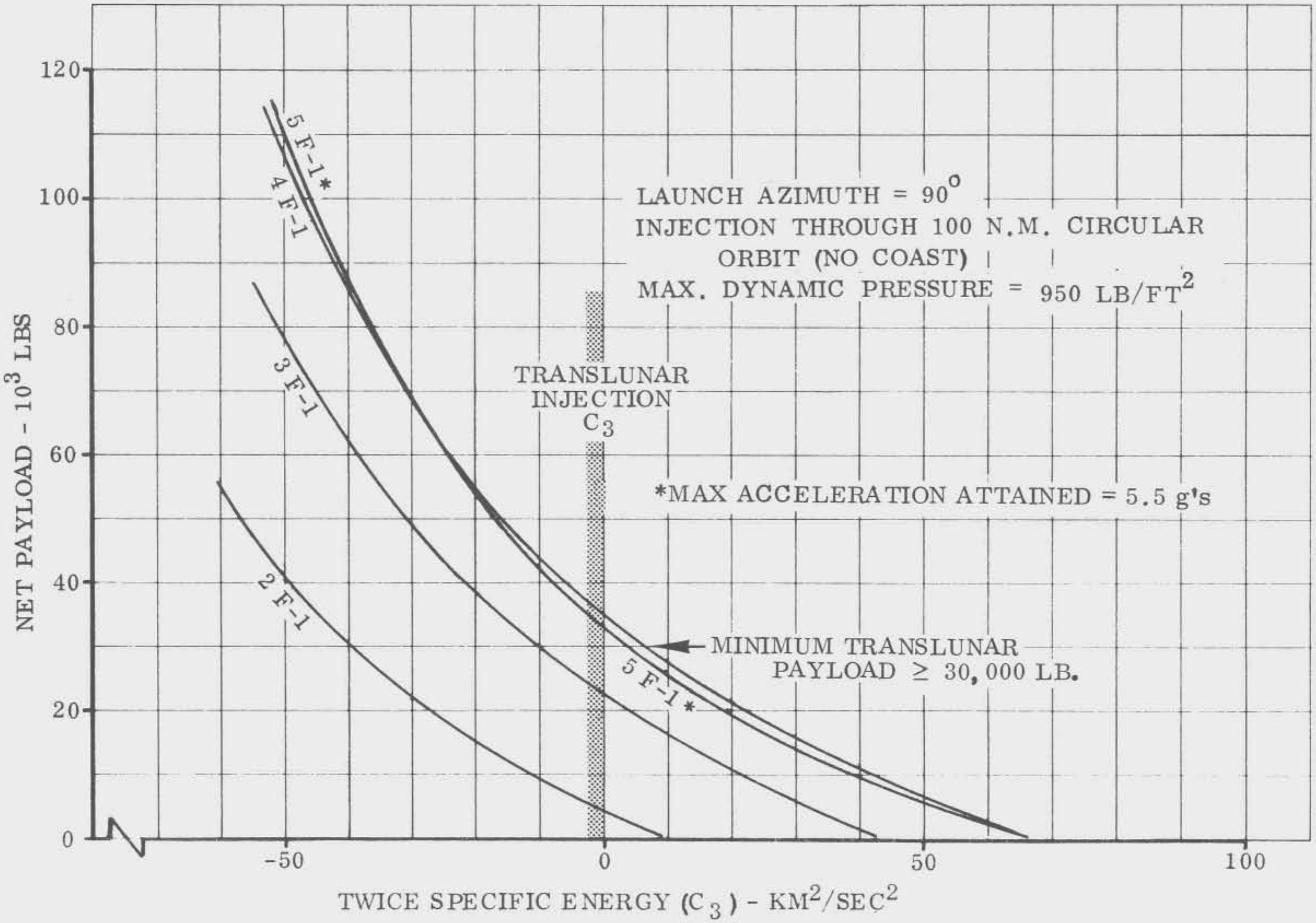


FIGURE 3.3.2-3 COMPARISON OF 6-g VEHICLE LUNAR CAPABILITIES

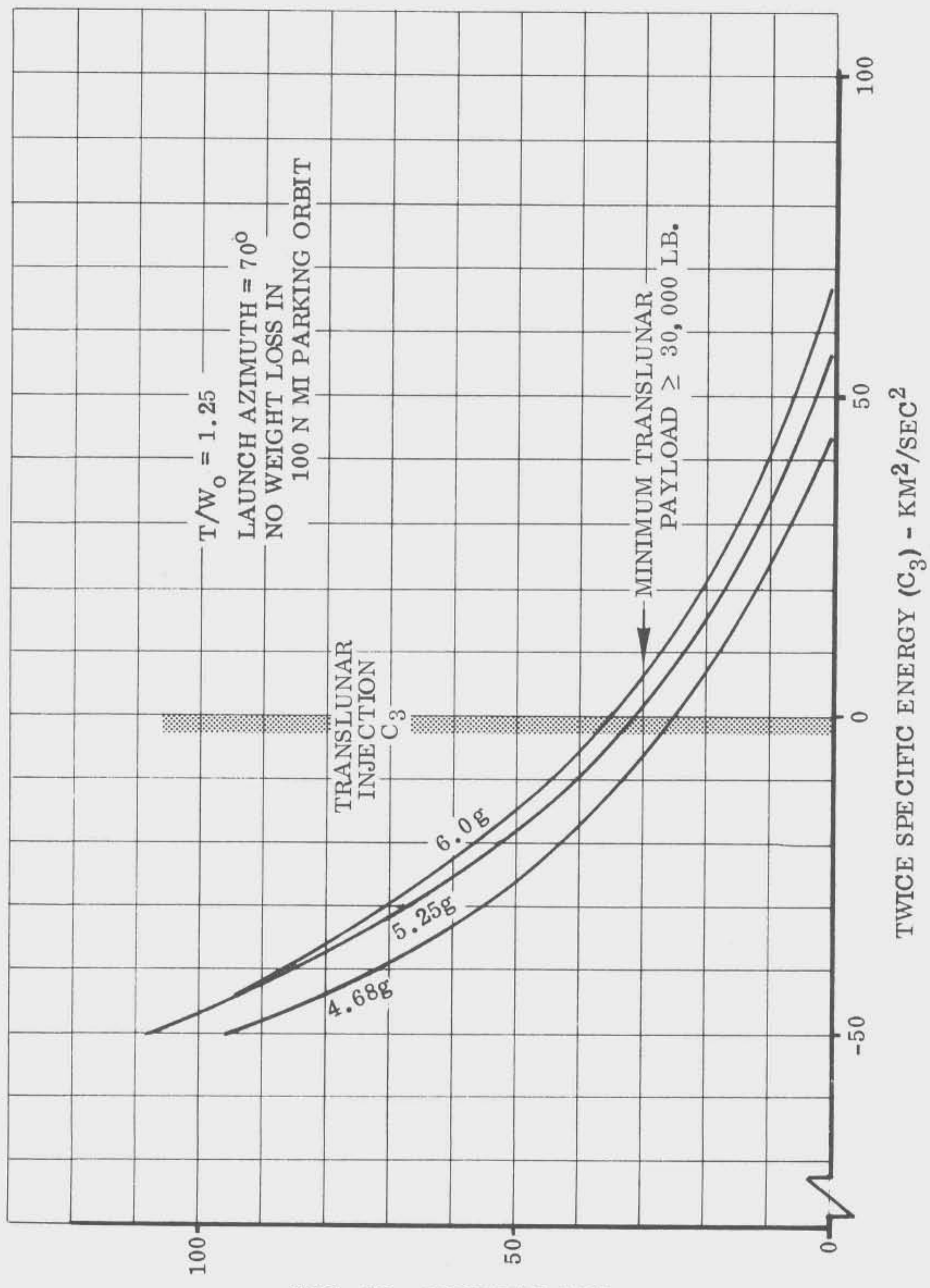


FIGURE 3.3.2-4 4 F-1 VEHICLE LUNAR CAPABILITIES

3.4 Conclusions

The principal conclusions reached during the Phase I analyses were:

a. It is technically feasible to implement all configurations studied.

b. The 500 series (Saturn V) S-IVB and Instrument Unit are most suitable for INT-20 applications.

c. No major stage modifications are required to achieve any configuration.

d. The cost to develop each INT-20 configuration is low. Individual costs do not vary significantly.

e. The recurring cost variation between each configuration is low. The savings in recurring cost is about $3M for each F-1 engine not installed.

f. A wide range of payload capabilities is possible by adding F-1 engines to the S-IC stage and/or increasing the peak longitudinal acceleration allowed during first stage operation from 4.68 g's to 6.0 g's.

g. The 5 F-1 INT-20 vehicle configurations are least desirable for manned mission support using Apollo or Gemini-type spacecraft because of their inherently high (950 lb/ft^3) max q. These vehicles, however, are not ruled out for unmanned applications, particularly where a large payload capability is needed.

h. The 2 F-1 and 3 F-1 vehicles did not meet the payload performance criterion established for vehicle selection.

i. The 4 F-1 vehicle is the most versatile of the INT-20 configurations studied. Its possible capabilities cover the performance range from just higher than the 6-g 3 F-1 vehicle to as great as the 5.5-g 5 F-1 vehicle.

j. The 4.68-g 4 F-1 vehicle is most suitable of the configurations for manned mission support.

k. The 6-g 4 F-1 vehicle is suitable for support of smaller-payload (50,000 lb or less), unmanned missions. It is particularly useful for supporting the very high-energy (C_3 >100) missions when used in conjunction with a third stage.

3.5 RECOMMENDATIONS

The baseline INT-20 recommended to NASA/MSFC for Phase II and Phase III definition was a 4.68-g, 4 F-1 INT-20. The specific recommendations were:

3.5 (Continued)

a. Use 4 F-1 S-IC

b. Use 500 series (Saturn V S-IVB stage and Instrument Unit)

c. Design for manned application (structural factor of safety = 1.4)

d. Design for a 100 NM circular orbit baseline mission

1. Maximum longitudinal acceleration of 4.68 g's at F.S. = 1.4

2. Basic payload of 132,000 lbs.

3. Overall payload length of 43 feet when designed using a 95 percentile March wind (75 m/sec). This choice of payload length was based on previous Boeing-sponsored research data (see Figure 3.5-1). The 43 ft. length matches the "minimum change in structure" criterion desired by NASA/MSFC. Payload length could increase to about 68 ft. for August launches (95% wind = 22 m/sec.)

3.6 BASELINE VEHICLE

The baseline INT-20 launch vehicle approved by NASA/MSFC for study during Phase II and Phase III is depicted in Figure 3.6-1. It consists of:

a. An S-IC with four F-1 engines (center engine removed);

b. An S-II/S-IVB interstage (retromotors deleted), with aft interface adapted to conform to S-IC forward interface;

c. A 500-series S-IVB stage;

d. A 500-series Instrument Unit; and,

e. A 43-foot long payload, comprised of an MSFC double-angle nose cone (MLV shape) plus a 15-foot, 260-inch diameter cylinder.

The vehicle baseline mission is a 100 NM circular orbit. The vehicle capability for this mission is approximately 132,000 lb. Maximum axial acceleration reached is 4.68 g's.

The payload envelope defined was established by two requirements. The first was the desire to minimize structural changes. The second was to maintain a structural factor of safety of 1.4 (manned configuration) when launching during a March 95 percentile design wind.

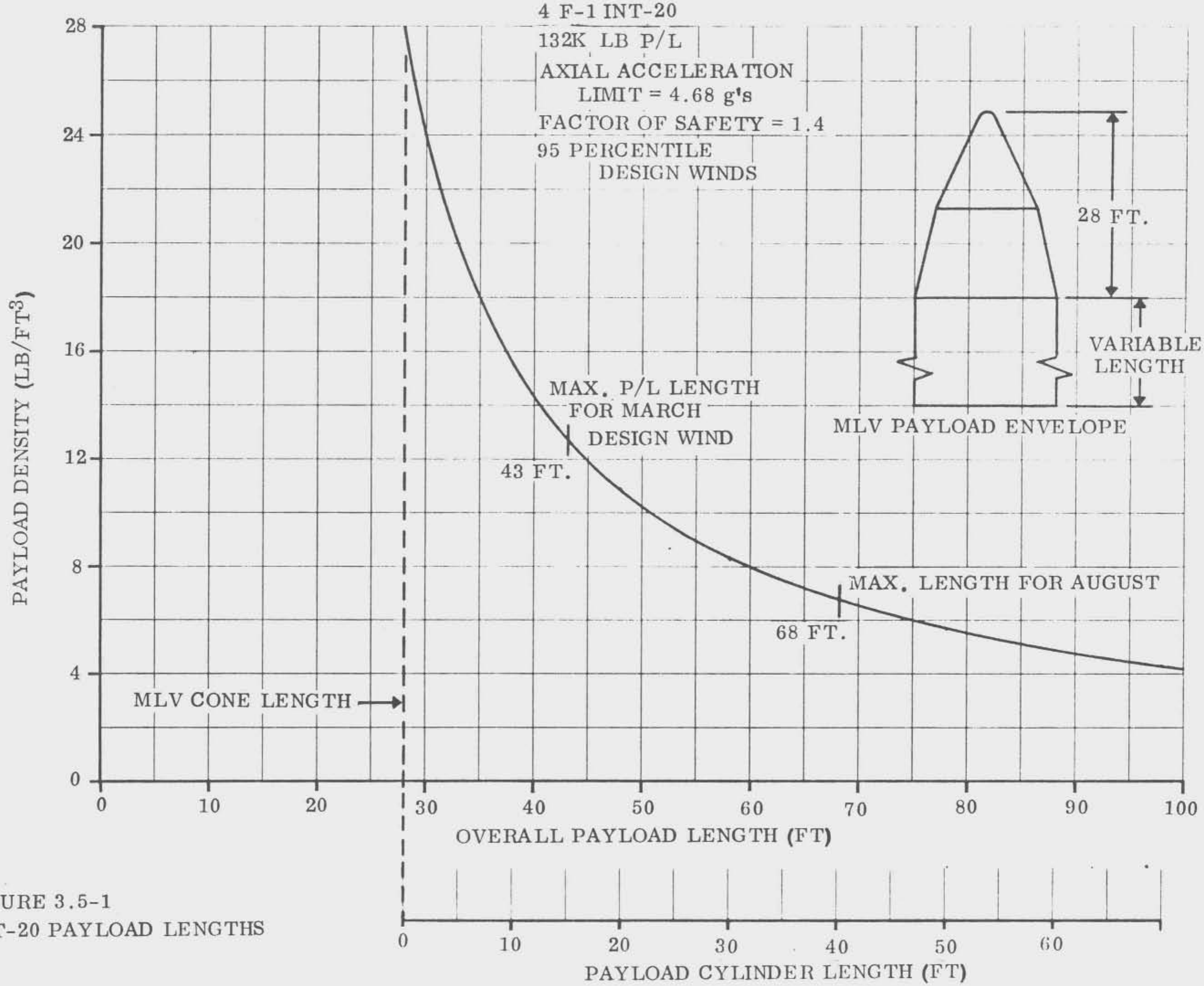


FIGURE 3.5-1
INT-20 PAYLOAD LENGTHS

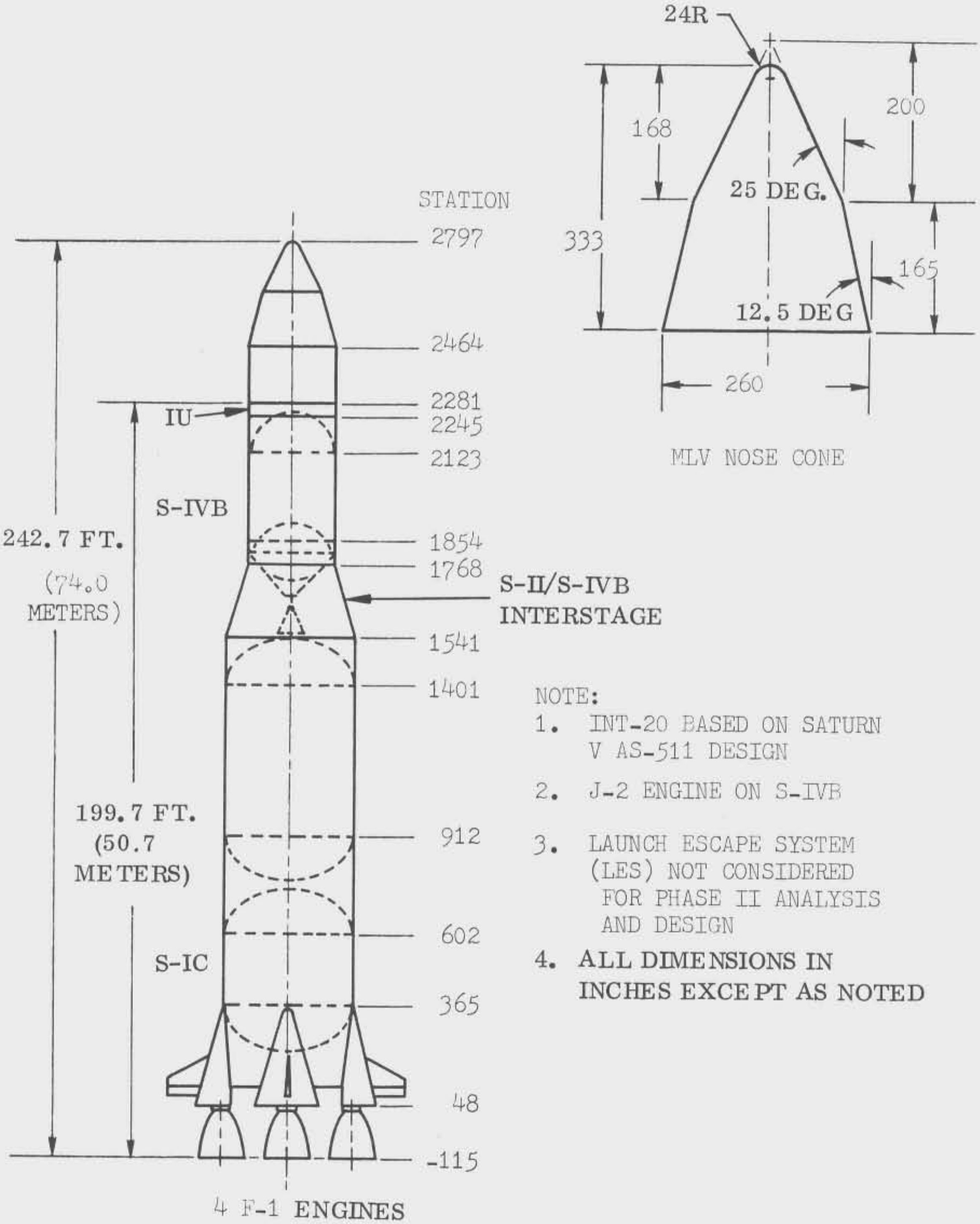


FIGURE 3.6-1 INT-20 BASELINE VEHICLE CONFIGURATION

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SECTION 4
PHASE II DESIGN AND ANALYSIS

4.0 GENERAL

After the baseline vehicle was selected, analyses were continued to provide definition of vehicle performance capabilities and design criteria. Specific stage design studies were provided by the stage contractors where appropriate.

4.1 BASELINE VEHICLE TECHNICAL ANALYSIS

Baseline vehicle technical analysis included vehicle performance and trajectories, aerodynamics and heating, vehicle control, design loads, structural dynamics and staging. Associated investigations were made for payload sensitivity, the Big Gemini payload configuration, removal of S-IVB restart capability and an improved flight control system.

4.1.1 Vehicle Performance

a. Study Baseline

The baseline mission for the INT-20 vehicle consists of a direct ascent boost to a 100 nautical mile (185.2 km) circular orbit. Maximum longitudinal acceleration is restricted to 4.68 g's by premature shutdown of two F-1 engines. Payload design weight is 132,781 pounds (60228 kg) with launch from the Eastern Test Range at an azimuth of 90 degrees. Baseline vehicle arrangement is shown in Section 4.2. The nominal payload length is 43 feet (13.1 m) with a density of 15 pounds per cubic foot (240 kg/m^3), designed for the March wind conditions (see Section 4.3.1). The basic payload configuration consists of a NASA double-angle nose cone (MLV cone) with a cylindrical portion extending the length to 43 feet (13.1 m). No launch escape system (LES) is provided on the baseline configuration.

b. Retrofit

Performance for the retrofit trajectory (see paragraph 4.1.1.4) was generated using the same ground rules as the study baseline performance except that axial acceleration was restricted to 3.68 g's at first two F-1 engine cutoff. Payload weight is 125,250 pounds (56810 kg). Figure 4.1.1-1 shows a comparison of INT-20 payload capability with factors of safety and axial acceleration shown below:

Factor of Safety	1.25	1.40
First F-1 Engine Shutdown	4.05 g's	3.68 g's
Final F-1 Engine Shutdown	4.8 - 5.6 g's	4.68 g's

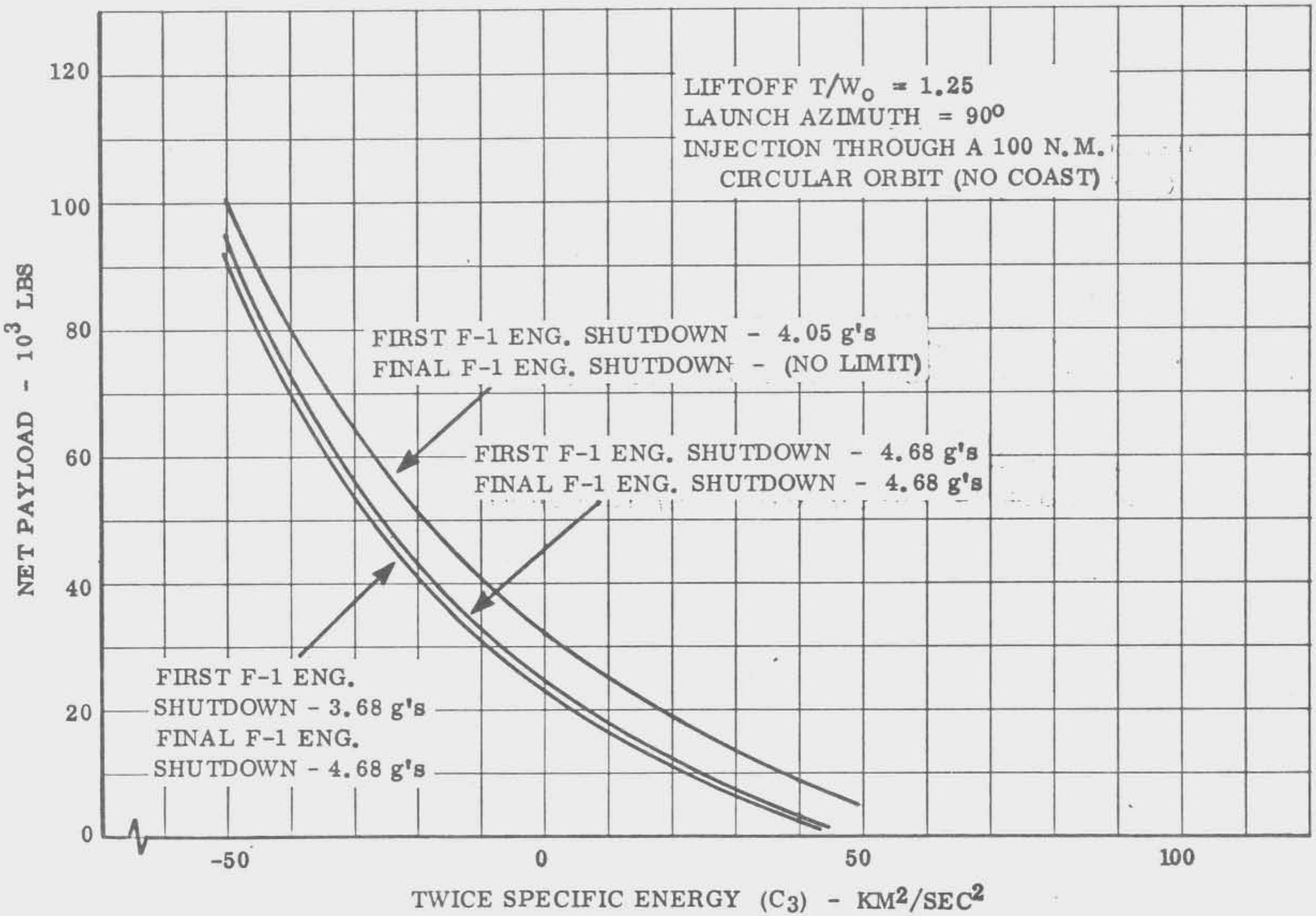


FIGURE 4.1.1-1 INT-20 RETROFIT PAYLOAD CAPABILITY

4.1.1.1 Orbital Missions

In addition to the baseline mission, payload capability for the INT-20 was established for other low Earth orbits, and for polar and synchronous orbits.

a. Low Earth Orbits

Low Earth orbit capability is presented in Figure 4.1.1.1-1 for circular orbit altitudes ranging from 100 to 300 nautical miles (185.2 to 555.6 km) with varied launch azimuths, coplanar with the desired orbit. Payload varies from the baseline 132,000 pounds (59,870 kg) to 101,000 pounds (45,813 kg) as orbital altitude is increased from 100 to 300 nautical miles (185.2 to 555.6 km) at a launch azimuth of 90 degrees.

b. Polar Orbits

Polar orbit payload capability is shown in Figure 4.1.1.1-2 for circular orbit altitudes of 100 to 300 nautical miles (185.2 to 555.6 km). Launch azimuth is assumed to be 145 degrees and yaw steering in both first and second stages is used to minimize land mass overfly. A rate of turn is selected so that, although Cuba and Panama are necessarily overflown, the western coast of South America is cleared. Payload injected into a 100 nautical mile (185.2 km) polar orbit is 91,800 pounds (41,640 kg) with an S-IC rate of turn of 0.45 degrees/second. Smaller rates of turn are required for higher orbital altitudes. In order to clear South America, 0.40 degrees/second and 0.35 degrees/second are required for orbital altitudes of 200 and 300 nautical miles (185.2 and 555.6 km), respectively. The effects upon payload of varying S-IC yaw angle and the resulting ground tracks are shown in Paragraph 4.1.1.4.

c. Synchronous Orbits

Net payloads for the INT-20 vehicle are shown in Figure 4.1.1.1-3 for synchronous orbit inclinations ranging from zero to 55 degrees. Launch azimuth was 90 degrees in all cases and orbit inclination was effected by a plane change at synchronous orbit altitude. The INT-20 has a payload capability of approximately 11,900 pounds (5397 kg) for an orbit inclination coplanar with the 90 degree launch azimuth, and approximately 7,800 pounds (3,538 kg) for an orbital inclination of zero degrees.

4.1.1.2 High Energy Missions

Performance capability of the INT-20 vehicle is enhanced by addition of a third, or injection, stage which increases available energy at injection into the desired mission trajectory. Applications of the INT-20 with an injection stage include missions to

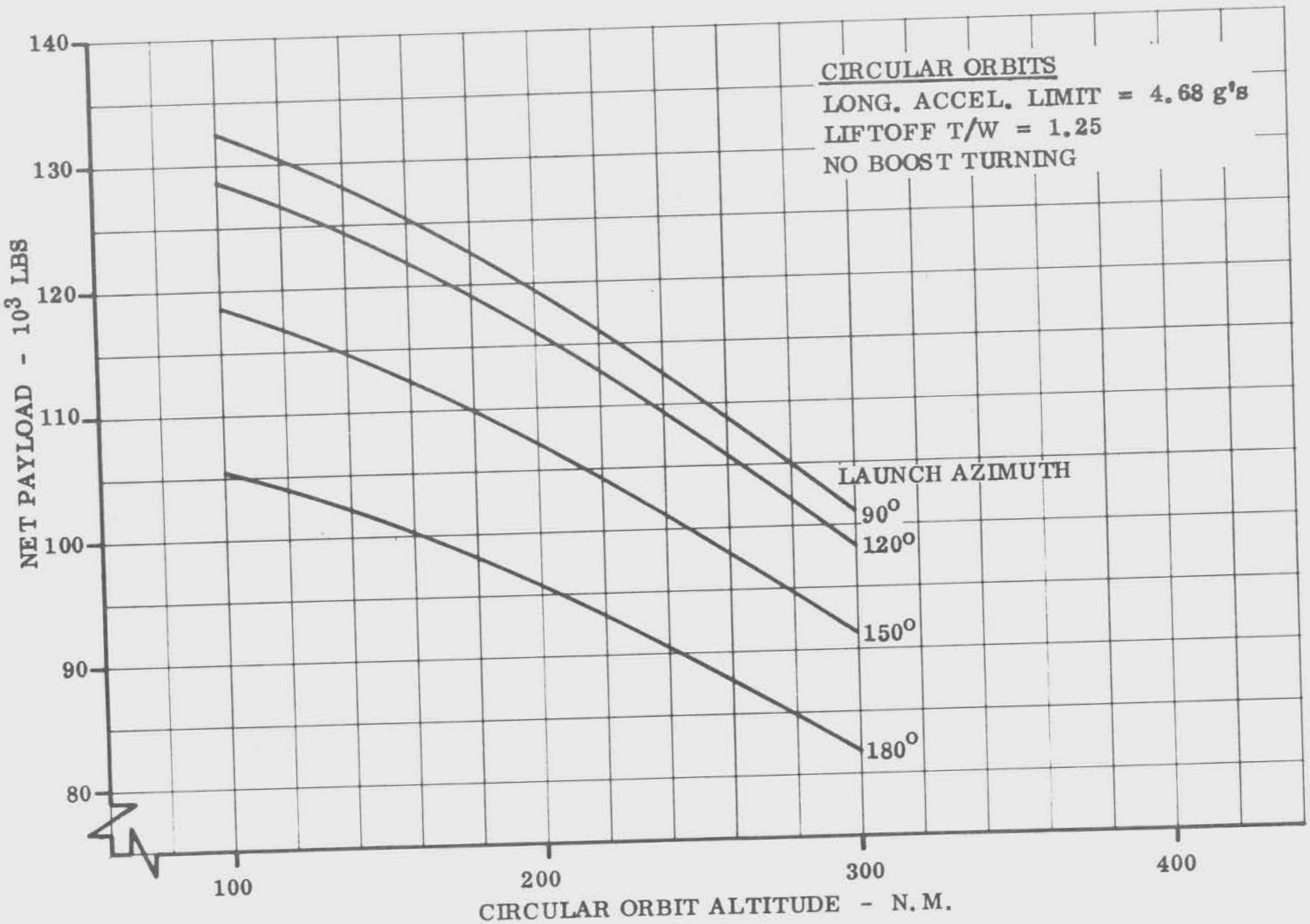


FIGURE 4.1.1.1-1 INT-20 LOW EARTH ORBIT PAYLOADS

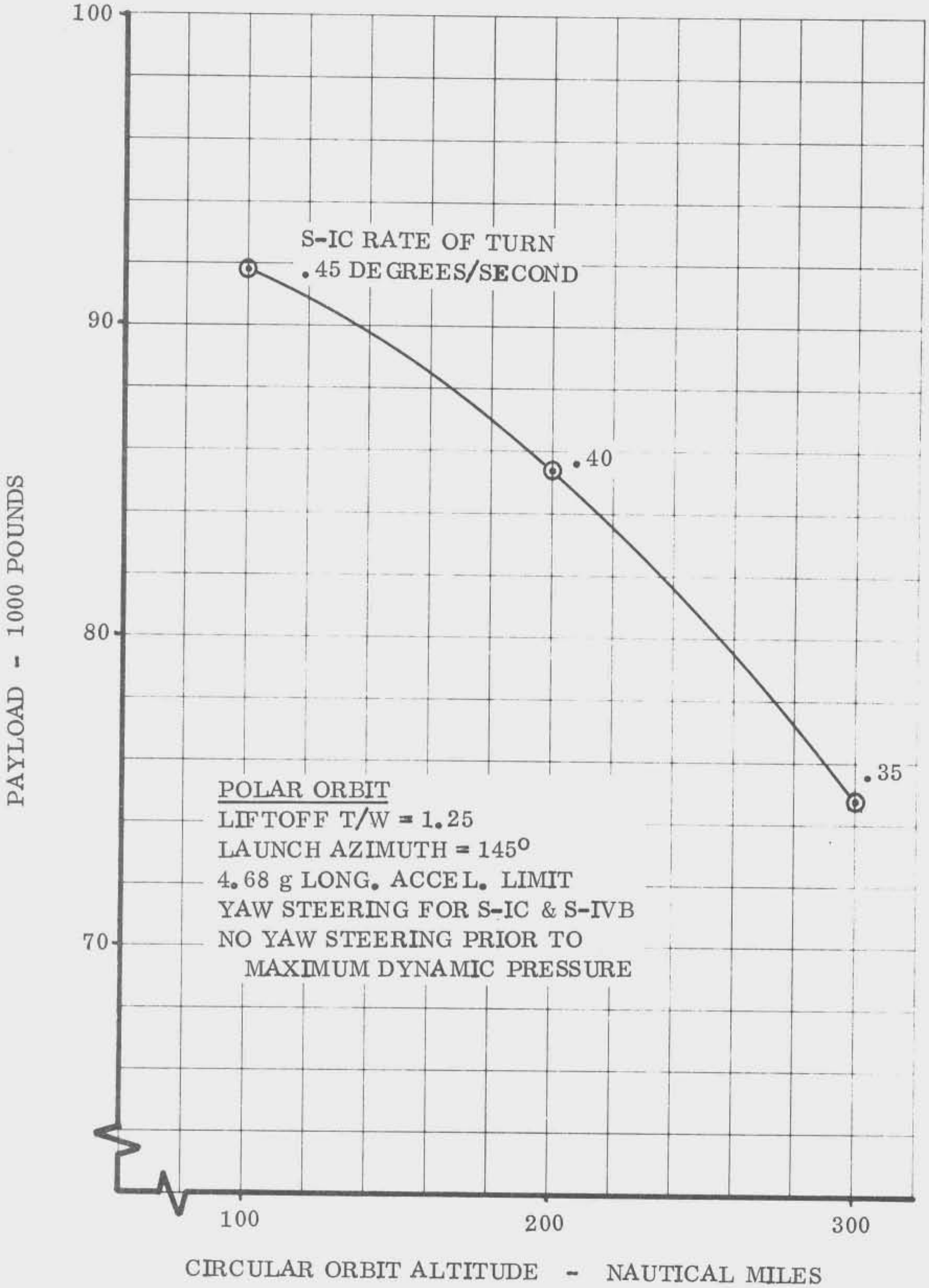


FIGURE 4.1.1.1-2 INT-20 POLAR ORBIT PAYLOADS

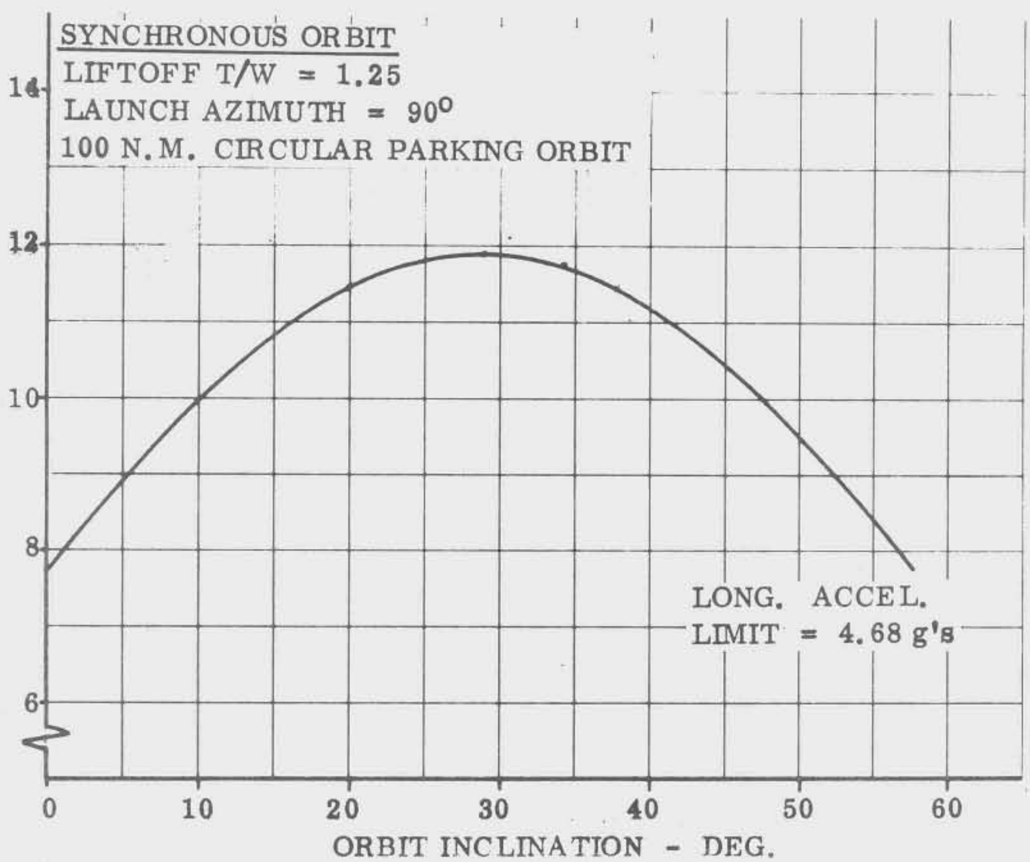


FIGURE 4.1.1.1-3 INT-20 SYNCHRONOUS ORBIT PAYLOADS

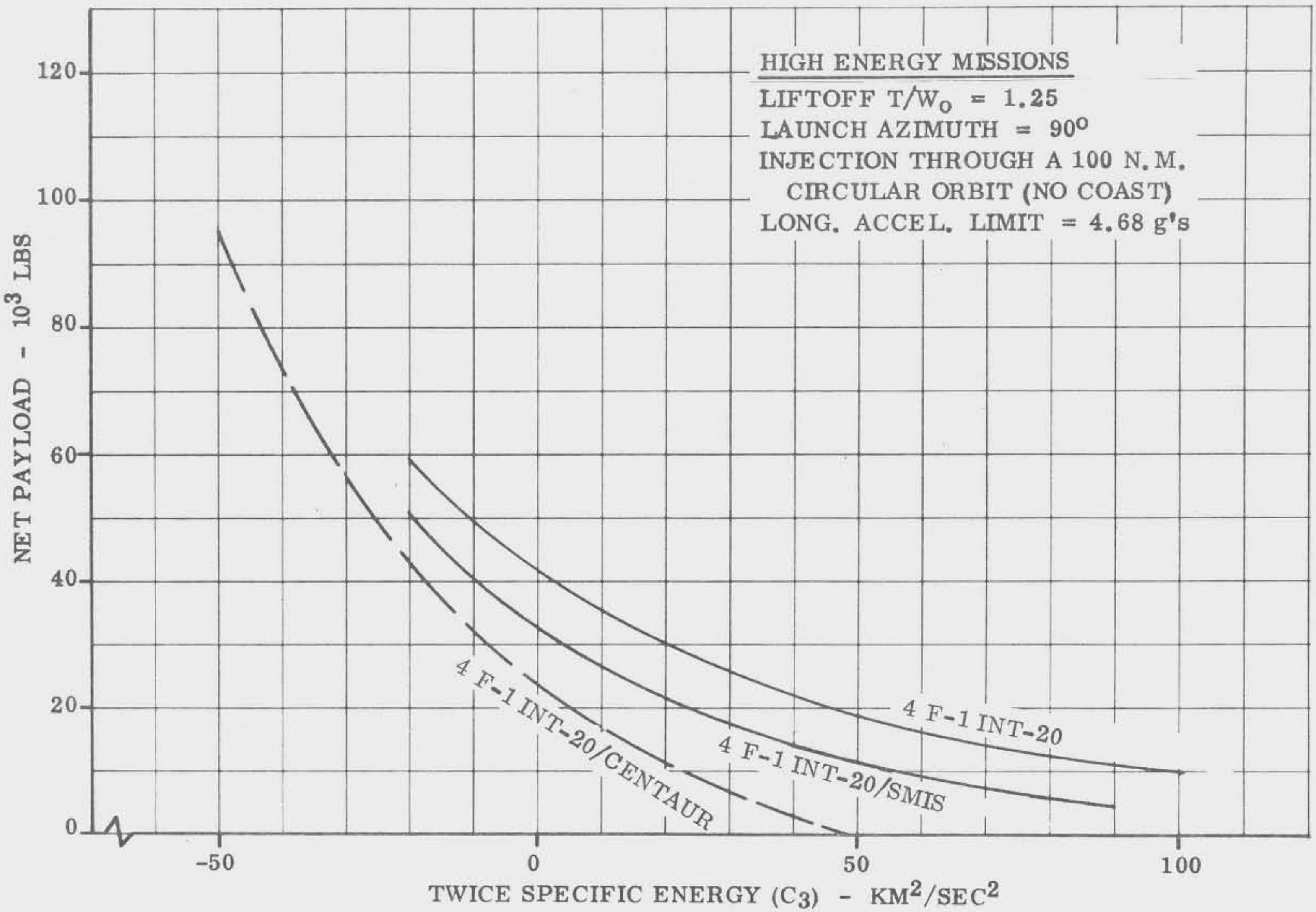


FIGURE 4.1.1.2-1 INT-20 HIGH ENERGY MISSIONS

4.1.1.2 (Continued)

the moon and planets. The Centaur and the Service Module Injection Stage (SMIS) have been considered as injection stages and the payload capability of the INT-20 with these stages is shown in Figure 4.1.1.2-1. Maximum longitudinal acceleration is limited to 4.68 g's. Vehicle payload capabilities are presented for a range of energy levels, using the energy parameter $C_3, km^2/sec^2$, which is equal to twice the specific energy through the relationship:

$$2E = C_3 = V_i^2 - \frac{2g_o \; r_o^2}{r}$$

A C_3 of zero is equivalent to earth escape velocity. Baseline INT-20 vehicle payload capability is also shown for comparison. The INT-20/Centaur has a payload capability of 10,000 pounds (4536 kg) at a C_3 of 100, with a limit of 4.68 g's maximum longitudinal acceleration.

A C_3 of zero is equivalent to earth escape velocity. Baseline INT-20 vehicle payload capability is also shown for comparison. The INT-20/Centaur has a payload capability of 10,000 pounds (4536 kg) at a C_3 of 100, with a limit of 4.68 g's maximum longitudinal acceleration.

4.1.1.3 Exchange Ratios

Payload exchange ratios which relate the effect of perturbations in vehicle parameters on net payload were generated for the baseline INT-20. These exchange ratios (or trade factors) are presented in Figures 4.1.1.3-1 through 4.1.1.3-7 for both the S-IC and S-IVB stages for Δ payload versus Δ thrust, Δ specific impulse and Δ propellant. The slopes of these curves at the nominal value of the perturbed parameter are tabulated in Table 4.1.1.3-I. With the exception of the ΔPLD/ΔWP, exchange ratio, all of the exchange ratios were developed using the following ground rules:

a. Liftoff thrust to weight ratio was constant at 1.25.

b. Maximum longitudinal acceleration limit was 4.68 g's.

The acceleration limit of 4.68 g's was met by premature shutdown of two F-1 engines at approximately 146 seconds after liftoff (see Figure 4.1.1.4-2 for acceleration-time history.) The other two engines continued to burn until the acceleration limit was again reached, at which time final cutoff occurred. Although total stage burn time was longer than if cutoff of the first two engines were not early, excess propellants remained in the S-IC stage at final cutoff. This propellant weight was staged as ballast. An increase in net payload weight due to thrust or specific impulse perturbations results in an equal decrease in ballast (and vice-versa) due to the requirement for maintaining a constant liftoff thrust/weight ratio.

4.1.1.3 (Continued)

Total propellant weight was constant in these cases. S-IVB ignition weight varies directly with S-IC ballast, producing a one to one trade between net payload and S-IC ballast. This is not true for propellant weight perturbations since total propellant weight changes.

The exchange ratio for WP_1 was obtained for two different ground rules, as follows:

a. Liftoff thrust/weight ratio was held constant and S-IC burnout acceleration was not limited to 4.68 g's. Variation of acceleration with S-IC propellant loading is shown in Figure 4.1.1.3-1 along with change in payload. Any reduction or change in S-IC propellant must be traded with ballast on an equal basis in order to maintain the thrust/weight ratio at 1.25. The reverse is also true (change in ballast to determine a ballast exchange ratio must be matched by a change in S-IC propellant). Therefore, for the INT-20 vehicle, the negative of this S-IC propellant exchange ratio is the ballast exchange ratio.

b. S-IC burnout acceleration was held constant at 4.68 g's and liftoff thrust/weight ratio varied. Change in thrust/weight ratio with S-IC propellant loading is shown in Figure 4.1.1.3-2 along with change in payload.

The exchange ratio for S-IC thrust is shown in Figure 4.1.1.3-3. This parameter was also affected by the study ground rules in that, as the thrust was reduced, vehicle liftoff weight and in turn, S-IC burnout weight, was reduced in order to maintain the liftoff thrust/weight ratio at 1.25. Consequently, the S-IC propellant weight was not held constant and ballast loading was reduced which resulted in a payload gain that negated payload loss due to lower thrust. Therefore, until the ballast loading became zero (at which point the S-IC propellant rather than ballast was varied to maintain the thrust/weight ratio), reducing the S-IC thrust resulted in a payload gain. The break in the curve occurs at the point where ballast became zero.

The exchange ratio for S-IVB propellant is shown in Figure 4.1.1.3-5. There is a break in this curve, similar to that of the S-IC exchange ratio curve, at the point at which S-IC ballast becomes zero. In order to maintain the thrust/weight ratio at 1.25, S-IC ballast was decreased as S-IVB propellant was increased until the required S-IC ballast became zero and S-IC propellant itself was decreased. The effect on net payload was greater when S-IVB propellant was traded for ballast than for S-IC propellant, thus the break in the curve.

These exchange ratios are unique to the baseline INT-20 configuration, with ground rules applied to thrust/weight ratio, and maximum longitudinal acceleration limit with S-IC stage ballast. They should not be used with other configurations.

TABLE 4.1.1.3-I INT-20 PAYLOAD EXCHANGE RATIOS

T/W = 1.25, 4.68 g Acceleration
Limit, except as noted

	S-IC STAGE	S-IVB STAGE
$\frac{\Delta WPL}{\Delta WP} \sim \frac{LBS}{LB} \sim \frac{KG}{KG}$	0.3* 0.025**	0.549
$\frac{\Delta WPL}{\Delta F} \sim \frac{LBS}{LB} \sim \frac{KG}{KG}$	-.00185	0.190
$\frac{\Delta WPL}{\Delta Isp} \sim \frac{LBS}{SEC} \sim \left(\frac{KG}{SEC}\right)$	1500 (680)	500 (227)

* ACCELERATION = f (WP_1)

** T/W = f (WP_1)

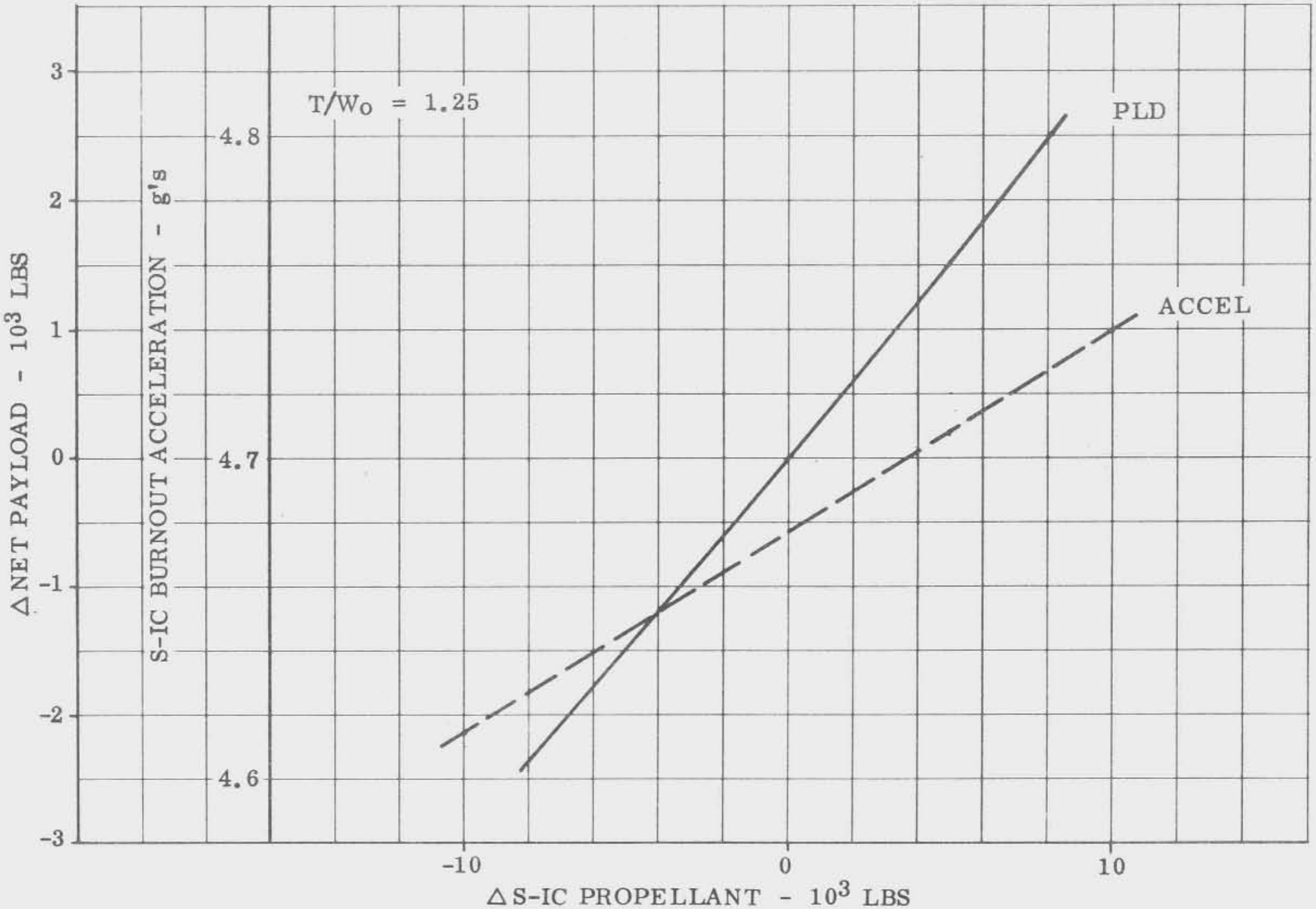


FIGURE 4.1.1.3-1 S-IC PROPELLANT EXCHANGE RATIO - T/W_o CONSTANT

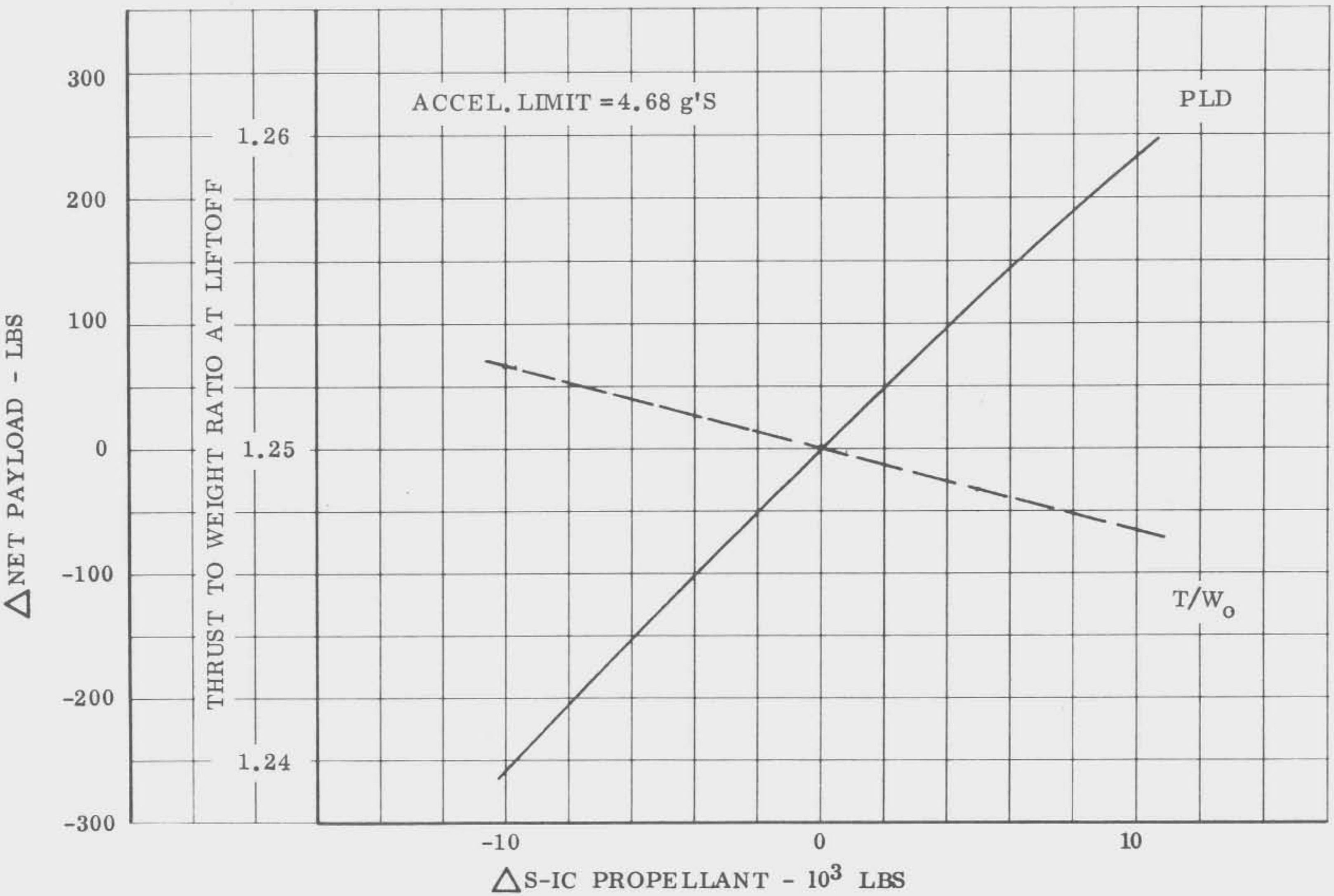


FIGURE 4.1.1.3-2 S-IC PROPELLANT EXCHANGE RATIO - ACCELERATION CONSTANT

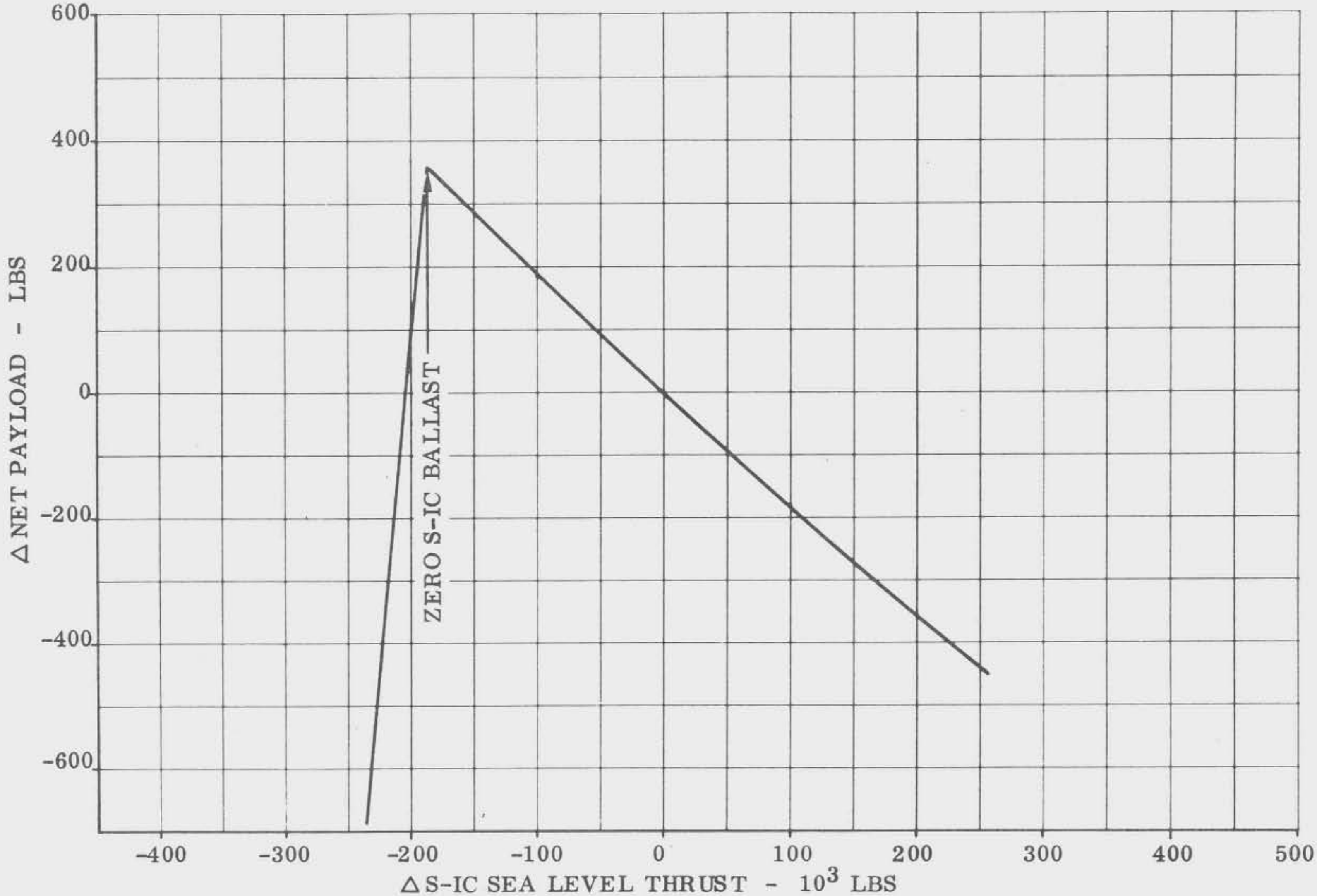


FIGURE 4.1.1.3-3 S-IC THRUST EXCHANGE RATIO

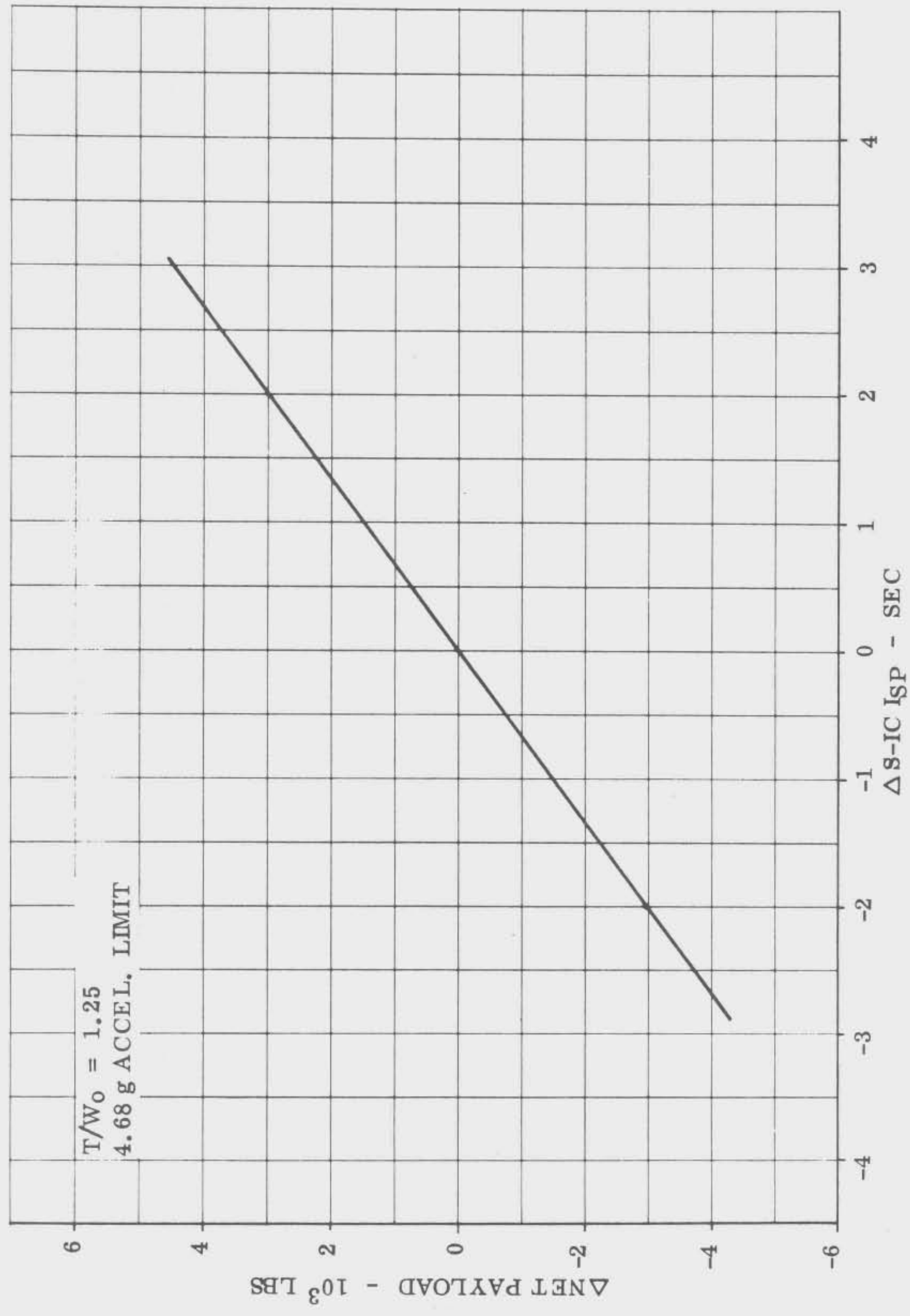


FIGURE 4.1.1.3-4 S-IC SPECIFIC IMPULSE EXCHANGE RATIO

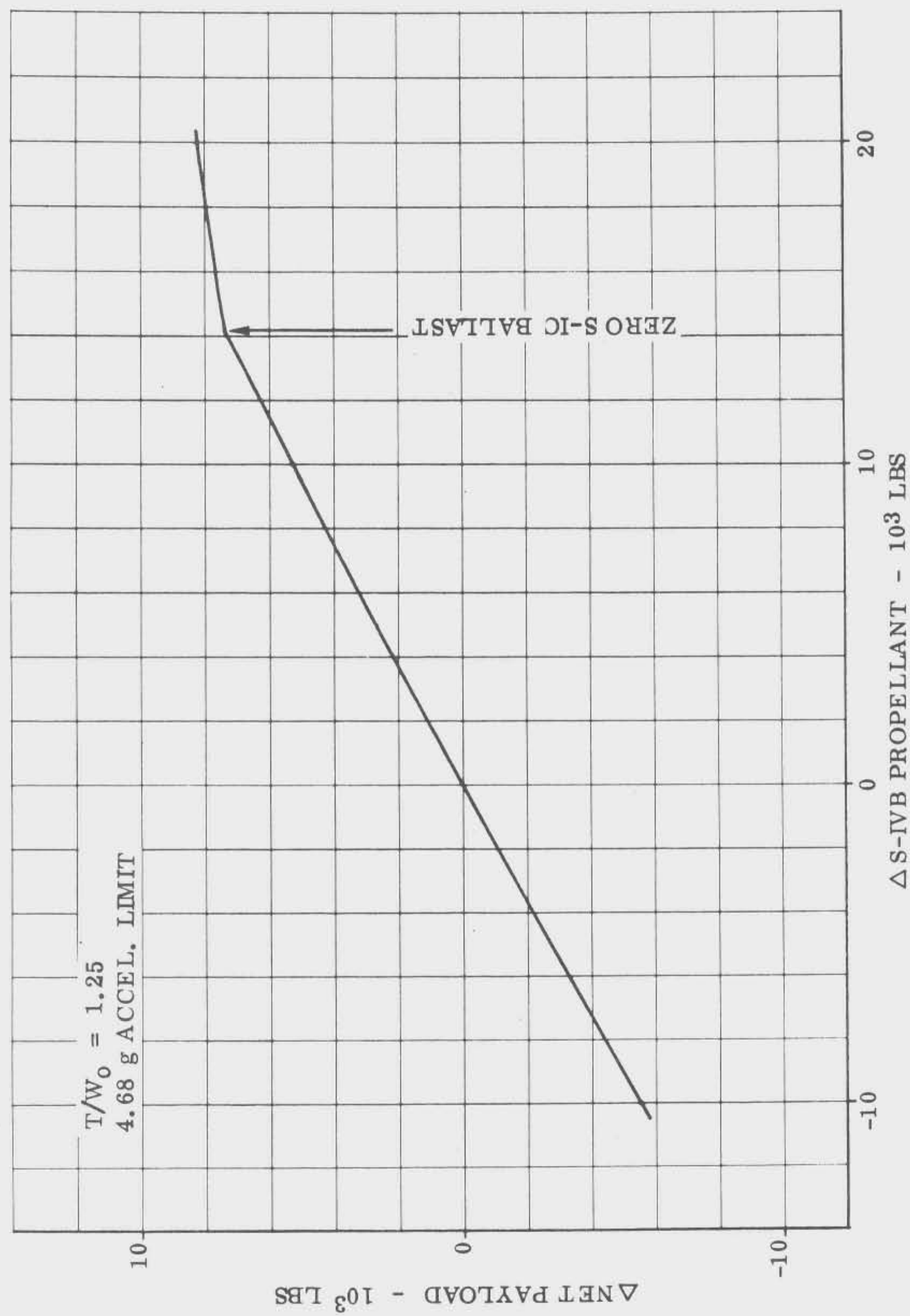


FIGURE 4.1.1.3-5 S-IVB PROPELLANT EXCHANGE RATIO

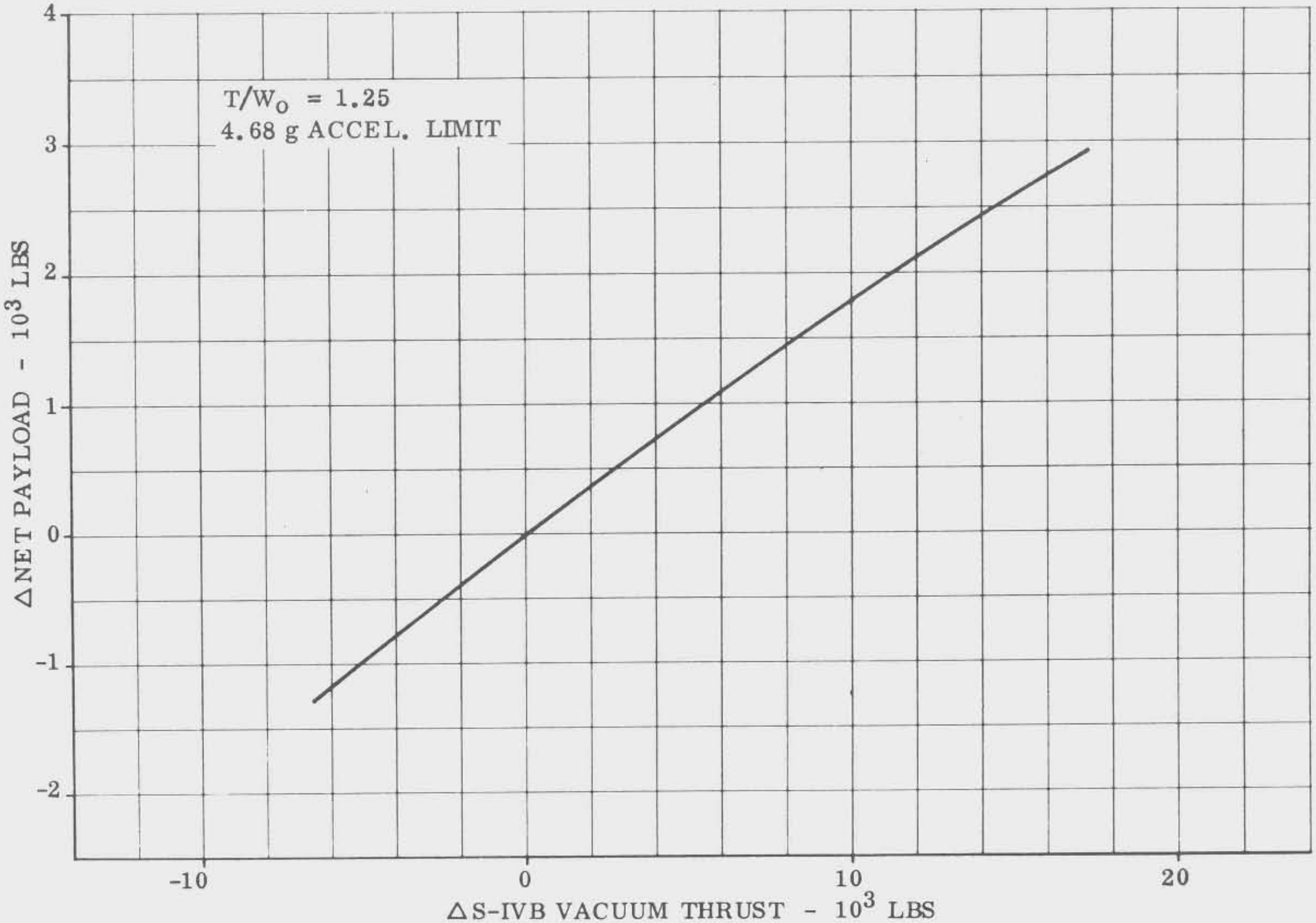


FIGURE 4.1.1.3-6 S-IVB THRUST EXCHANGE RATIO

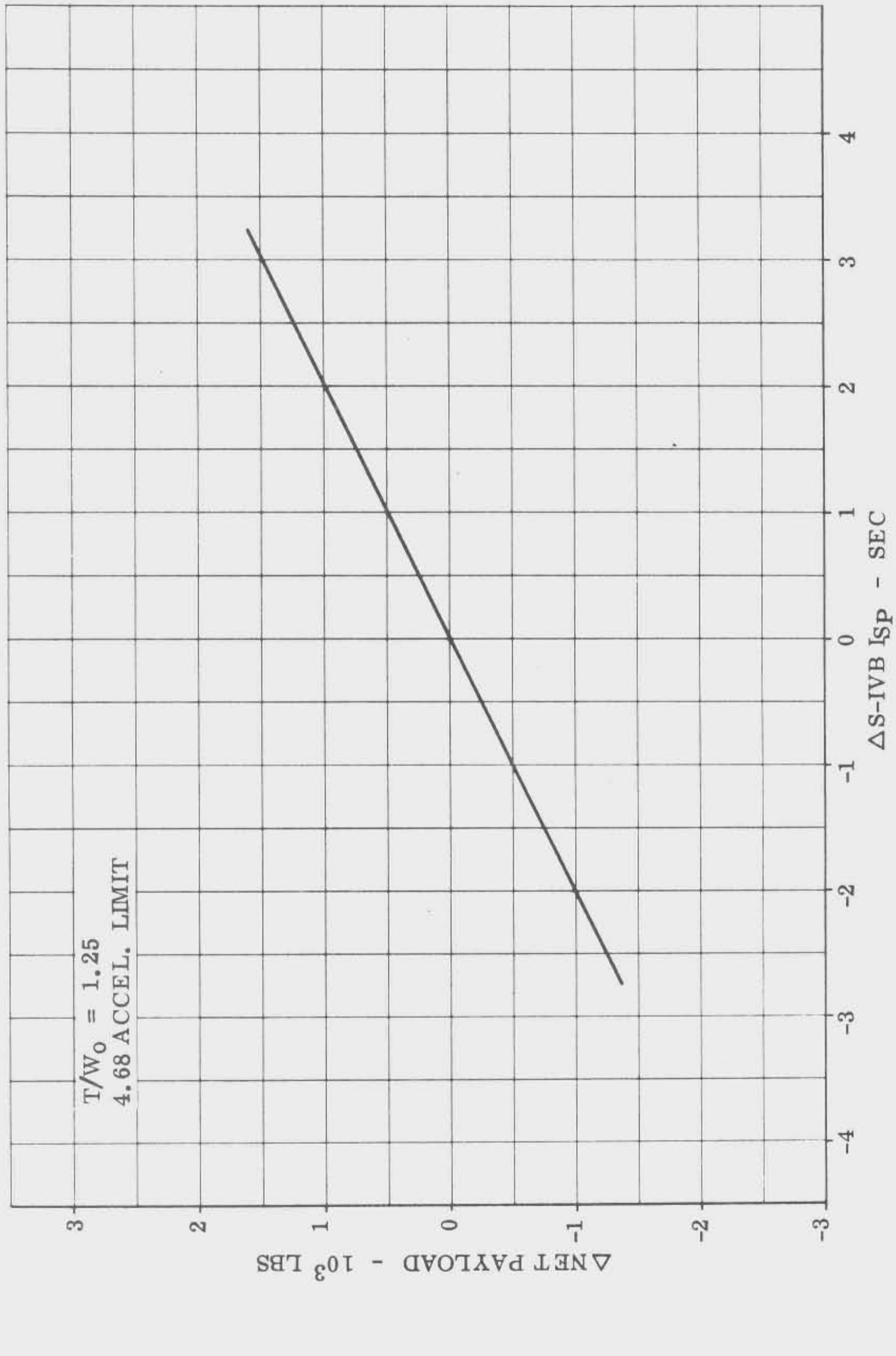


FIGURE 4.1.1.3-7 S-IVB SPECIFIC IMPULSE EXCHANGE RATIO

4.1.1.4 Trajectories

a. Baseline Trajectory

1. Study Baseline

The baseline trajectory printout is shown in Table 4.1.1.4-II. The trajectory profile is coplanar with the desired orbit and is characterized by vertical liftoff, a roll and tilt into the desired flight azimuth, and a gravity turn trajectory which is continued until the dynamic pressure decreases to 50 kilograms per square meter or second stage ignition, whichever occurs first. From this point to orbit, the trajectory is optimized for maximum weight in orbit. Liftoff thrust-weight ratio is 1.25 and shut-down of engines 2 and 4 is used to restrict maximum longitudinal acceleration to 4.68 g's. Engines 1 and 3 continue to thrust until the acceleration limit again is reached, at which time final stage cutoff is effected. Any usable propellants remaining are treated as ballast and are staged with the S-IC. A 3.8 second coast is assumed between S-IC shutdown and S-IVB ignition. Baseline trajectory simulation was made with a computer program which consideres the vehicle a point mass and optimizes the thrust vector angle in the pitch plane through use of the calculus of variations (COV). Vehicle mission weight history is shown in Table 4.1.1.4-I. Summarized baseline trajectory ground rules are as follows:

(a) The mission flown was direct injection with two stages into a 100 NM (185.2 km) circular orbit, launch being from the AMR at an azimuth of 90 degrees.

(b) A lift off thrust/weight ratio of 1.25 was obtained by off-loading S-IC propellant.

(c) Vehicle weights are those shown in Paragraph 4.1.7.

(d) No mixture ratio shift was used in either stage.

(e) A 3.8 second coast was flown between S-IC final engine cutoff and S-IVB ignition.

(f) Maximum longitudinal acceleration was limited to 4.68 g's. This was accomplished by shutting down two F-1 engines at t = 146 seconds and then staging ballast with the S-IC stage at final engine cutoff.

4.1.1.4 (Continued)

2. Retrofit Trajectory

In order to retrofit the current S-IC for use on the INT-20, structural studies have indicated that the acceleration loading on the S-IC bulkheads will have to be reduced. (See section 4.1.6.4). Presented in this section are the performance and trajectory characteristics of the INT-20 with an acceleration schedule which allows use of the current S-IC on the INT-20. A vehicle definition of this retrofit INT-20 is presented in Table 4.1.1.4-IV; trajectory data are presented in Table 4.1.1.4-VI.

The retrofit baseline trajectory is essentially the same as that presented in paragraph a.1., above, except that preliminary baseline weights (Table 4.1.1.4-V) were used rather than final baseline weights (Table 4.1.1.4-I). Separation weights for the S-IVB stage and the I.U. are different for the final baseline than for the preliminary baseline. Also, there is a difference in the ground rules - this being that the acceleration level of the first F-1 engine shutdown (normally 4.68 g's at a time from liftoff of 146 seconds) is limited to 3.68 g's (which occurs at a time from liftoff of 129 seconds). All other ground rules, propellant capacities, stage weights, and propulsion characteristics remain the same as those employed for the study baseline INT-20 vehicle described above.

As shown in Table 4.1.1.4-IV, employing this adjusted acceleration results in a net payload to a 100 N.M. orbit of 125,250 lbs for the retrofit INT-20. This represents a loss of 6,766 lbs when compared to the preliminary baseline INT-20 presented in Table 4.1.1.4-V.

Variation of axial acceleration and fuel level with time is shown for the INT-20 and the Saturn V in Figure 4.1.1.4-2.

b. Polar Orbit Trajectory

The trajectory for the polar orbit mission is simulated in the same sequence as the baseline, except that the program used has the additional capability of introducing a constant yaw angle into the yaw plane of flight. The constant yaw angle is used to generate a rate of turn from the launch azimuth of 145 degrees so that polar orbit inclination can be achieved. After the vehicle passes through peak dynamic pressure, application of a specified S-IC yaw angle (into the control loop) results in a rate of turn that is maintained for the remainder of the first stage burn time. At S-IVB ignition, a yaw angle produces

4.1.1.4 (Continued)

a rate of turn that results in attainment of the desired orbit inclination of 90 degrees. The first stage rate of turn, or azimuth change, is selected to minimize land mass impact (i.e., overfly Cuba and Panama only). The effect upon payload of varying S-IC yaw is shown in Figure 4.1.1.4-1 for polar orbit altitudes of 100, 200, and 300 nautical miles (185.2, 370.4 and 555.6 km). The payload for each altitude increases with yaw rate, or azimuth change, to a maximum and then decreases. However, the maximums are obtained with yaw rates which produce ground tracks that clear South America. Ground tracks and stage impact points for a range of S-IC yaw rates are plotted in Figures D.2-1 through D.2-6 of the appendix, D5-17009-2, for each of the polar orbit altitudes.

c. Synchronous Orbit Trajectory

Launch azimuth is 90 degrees for all cases and orbital inclination is attained by plane change after reaching orbital altitude. The trajectory is simulated by direct injection into a 100 nautical mile (185.2 km) circular parking orbit followed by a coast, reignition of the S-IVB stage, and boost to transfer velocity. Synchronous orbit altitude, approximately 19,230 nautical miles (35,614 km), is reached after a 5.25 hour coast through an elliptical transfer and the S-IVB is then reignited for circularization and plane change. Orbit and transfer propellant boil-off and start-up losses assumed for the S-IVB stage are: 2,517 pounds (1,132 kg) for 4.5 hours in parking orbit, and 1,692 pounds (767 kg) for the 5.25 hour coast from parking orbit to synchronous orbit altitude.

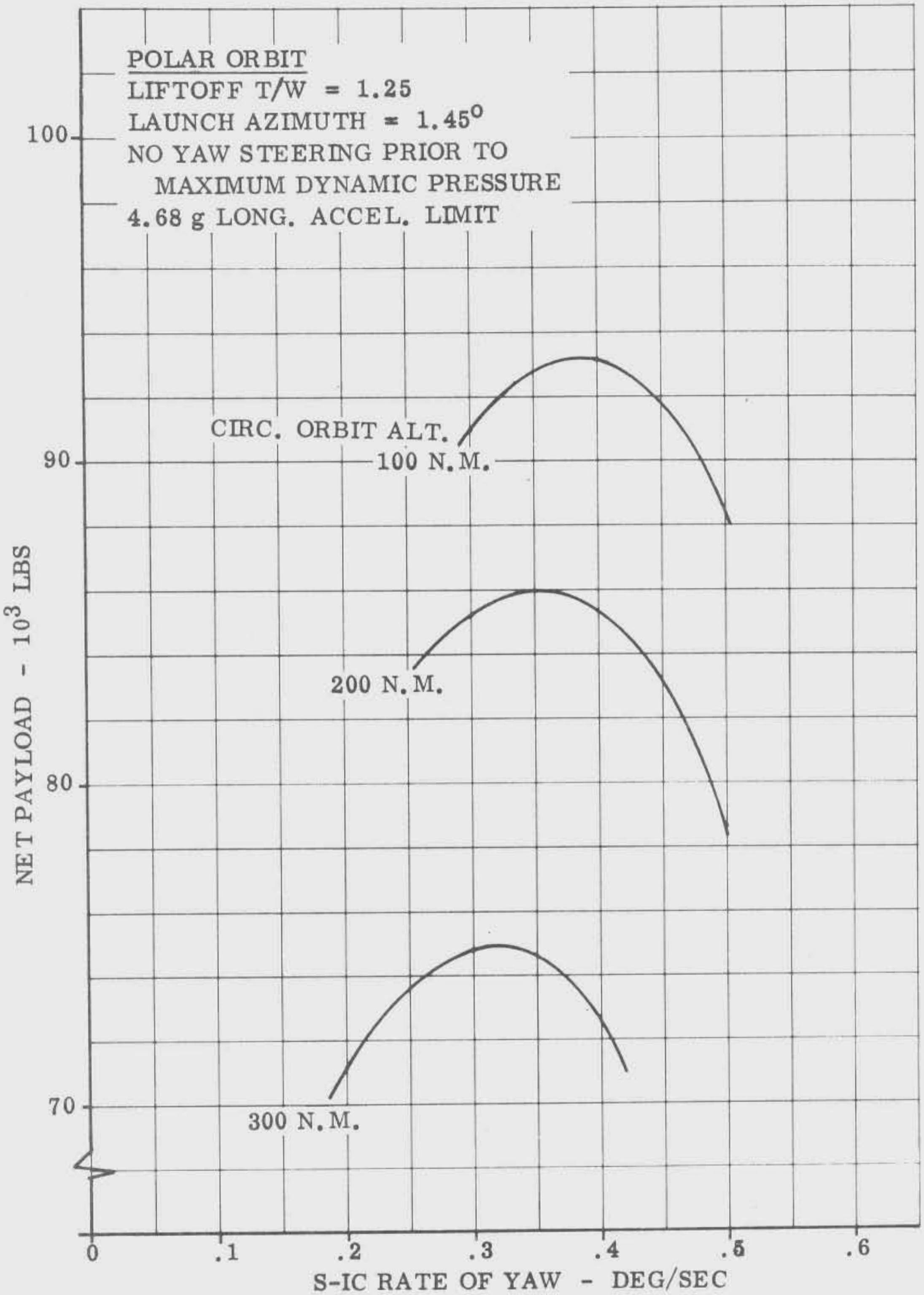
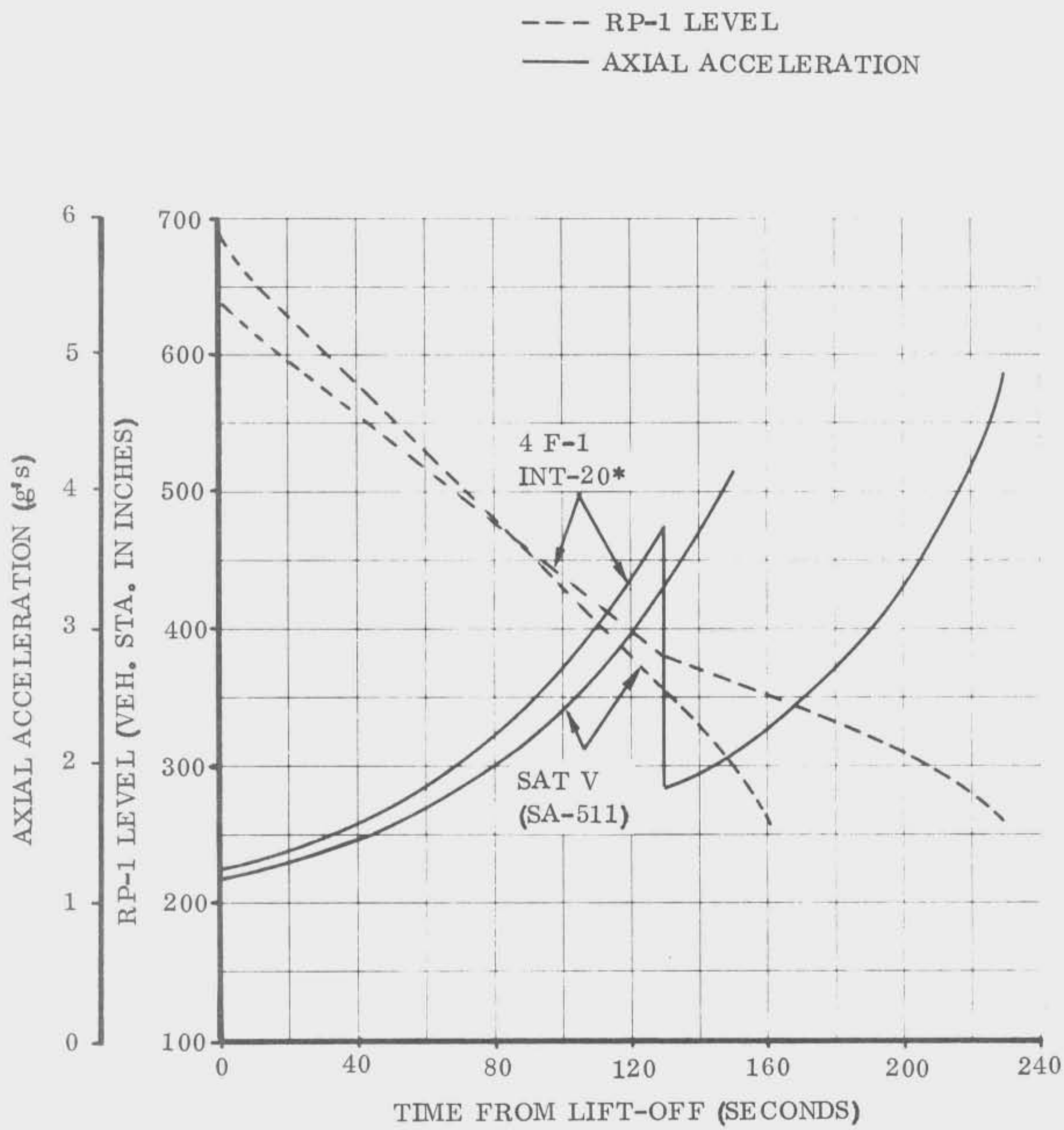


FIGURE 4.1.1.4-1 EFFECT OF S-IC YAW UPON INT-20 PAYLOAD



*NAS8-30502 BASELINE WITH 2-ENG. CUT-OFF @ 3.68 g's

FIGURE 4.1.1.4-2 AXIAL ACCELERATION AND RP-1 LEVEL - INT - 20 AND SATERN V SATURN V.

TABLE 4.1.1.4-I INT-20 BASELINE MISSION WEIGHT HISTORY

(100 N.M. CIRCULAR ORBIT, LAUNCH AZ. = 90^{o})

Liftoff Weight	lbs	4,870,400
Sea Level Thrust	lbs	6,088,000
Sea Level Specific Impulse	sec	263.58
Propellant Consumed	lbs	4,122,325
Stage Weight at Separation*	lbs	354,362
Thrust-to-Weight Ratio at Liftoff		1.25
Weight at Ignition	lbs	393,713
Vacuum Thrust	lbs	205,000
Vacuum Specific Impulse	sec	426
Propellant Capacity	lbs	230,000
Propellant Consumed	lbs	227,322
Stage Weight at Separation	lbs	26,629
Gross Payload	lbs	139,762
Weight to be Subtracted		
Astrionics Equipment	lbs	4,284
Flight Performance Reserves (3/4%)	lbs	2,678
Net Payload	lbs	132,800

* Includes 14,053 lbs of ballast

TABLE 4.1.1.4-II INT-20 BASELINE TRAJECTORY

	TIME SEC.	MASS KGS.	WEIGHT LBS.	THRUST NEWTONS	THRUST LBS.	LONGIT. ACCEL. MT/S.SQ.	LONGIT. ACCEL. FT/S.SQ.
LIFT OFF	.000	2209176.4	4870400.2	27080974.5	6088045.1	12.242	40.165
	8.000	2125362.1	4685621.3	27119343.5	6096670.9	12.734	41.777
	16.000	2041547.8	4500842.4	27242467.3	6124350.2	13.295	43.617
	24.000	1957733.4	4316063.3	27459273.3	6173090.1	13.940	45.735
	32.000	1873919.0	4131284.3	27772251.3	6243450.4	14.655	48.079
	43.000	1758674.3	3877213.2	28342622.0	6371674.8	15.762	51.712
	51.000	1674860.0	3692434.3	28828465.3	6480896.7	16.641	54.598
	59.000	1591045.7	3507655.4	29335117.0	6594796.6	17.413	57.129
	67.000	1507231.5	3322876.5	29820823.3	6703987.6	17.777	58.324
10 KMS	70.161	1474117.8	3249873.3	29997822.3	6743778.6	18.234	59.823
	75.000	1423417.1	3138097.6	30246673.3	6799722.6	19.242	63.130
Q MAX	79.000	1381510.0	3045708.2	30428154.5	6840521.1	20.129	66.041
14 KMS	81.196	1358500.5	2994980.9	30517364.8	6860576.4	20.666	67.801
	81.196	1358500.5	2994980.9	30517364.8	6860576.4	20.666	67.801
	91.000	1255788.6	2768539.9	30818960.5	6928377.8	23.276	76.363
	99.000	1171974.3	2583751.0	30958472.0	6959741.3	25.528	83.753
	107.000	1088160.0	2398982.2	31036520.8	6977287.3	27.916	91.589
	115.000	1004345.8	2214203.4	31078674.8	6986763.9	30.549	100.227
	123.000	920531.5	2029424.5	31100396.8	6991647.2	33.539	110.035
	131.000	836717.2	1844645.7	31111044.0	6994040.8	37.034	121.502
	139.000	752903.0	1659866.9	31116273.0	6995216.4	41.243	135.312
	146.000	679565.5	1498185.4	31118636.5	6995747.7	45.740	150.066
1 ENG CO	146.000	679565.5	1498185.4	15559318.3	3497873.8	22.844	74.947
	146.000	679565.5	1498185.4	15559318.3	3497873.8	22.844	74.947
	155.000	632419.9	1394247.3	15560077.8	3498044.6	24.565	80.594
	163.000	590512.8	1301357.9	15560379.6	3498112.4	26.329	86.383
	171.000	548605.7	1209468.5	15560509.1	3498141.6	28.353	93.022
	179.000	506698.6	1117579.1	15560558.1	3498152.6	30.705	100.738
	187.000	464791.4	1024689.7	15560573.4	3498156.0	33.477	109.833
	195.000	422884.3	932300.3	15560577.4	3498156.9	36.796	120.721
	203.000	380977.1	839910.8	15560578.4	3498157.1	40.844	134.002
CUTOFF	210.952	339321.0	748074.7	15560578.5	3498157.2	45.358	150.452
SEPARATION	210.952	178585.1	393712.7	.0	.0	-.000	-.000
	210.952	178585.1	393712.7	.0	.0	-.000	-.000
END COAST	214.752	178585.1	393712.7	.0	.0	-.000	-.000

TABLE 4.1.1.4-II (Continued)

	TIME SEC.	MASS KGS.	WEIGHT LBS.	THRUST NEWTONS	THRUST LBS.	LONGIT. ACCEL. MT/S.SQ.	LONGIT. ACCEL. FT/S.SQ.
	214.752	178585.1	393712.7	.0	.0	-.000	-.000
BOOSTER IMP.	575.334	160735.9	354362.0	.0	.0	-.000	-.000
START COV	214.752	178585.1	393712.7	911885.4	205000.0	5.106	16.753
	230.000	175256.8	386375.1	911885.4	205000.0	5.203	17.071
	246.000	171764.3	378675.5	911885.4	205000.0	5.309	17.418
	262.000	168271.9	370976.0	911885.4	205000.0	5.419	17.779
	278.000	164779.4	363276.5	911885.4	205000.0	5.534	18.156
	294.000	161287.0	355576.9	911885.4	205000.0	5.654	18.549
	310.000	157794.5	347877.4	911885.4	205000.0	5.779	18.960
	326.000	154302.1	340177.9	911885.4	205000.0	5.910	19.389
	342.000	150809.6	332478.4	911885.4	205000.0	6.047	19.838
	358.000	147317.2	324778.8	911885.4	205000.0	6.190	20.308
	374.000	143824.3	317079.3	911885.4	205000.0	6.340	20.801
	390.000	140332.3	309379.8	911885.4	205000.0	6.498	21.319
	406.000	136839.9	301680.2	911885.4	205000.0	6.664	21.863
	422.000	133347.4	293980.7	911885.4	205000.0	6.838	22.436
	438.000	129855.0	286281.2	911885.4	205000.0	7.022	23.039
	454.000	126362.5	278581.6	911885.4	205000.0	7.216	23.676
	466.352	123866.4	272637.8	911885.4	205000.0	7.374	24.192
	470.000	122870.1	270882.1	911885.4	205000.0	7.422	24.349
	486.000	119377.6	263182.6	911885.4	205000.0	7.639	25.061
	502.000	115885.2	255483.1	911885.4	205000.0	7.869	25.817
	518.000	112392.7	247783.5	911885.4	205000.0	8.113	26.619
	534.000	108900.3	240084.0	911885.4	205000.0	8.374	27.472
	550.000	105407.8	232384.5	911885.4	205000.0	8.651	28.383
	566.000	101915.4	224684.9	911885.4	205000.0	8.947	29.355
	582.000	98422.9	216985.4	911885.4	205000.0	9.265	30.397
	598.000	94930.5	209285.9	911885.4	205000.0	9.606	31.515
	614.000	91438.0	201586.3	911885.4	205000.0	9.973	32.719
	630.000	87945.6	193886.8	911885.4	205000.0	10.369	34.018
	646.000	84453.1	186187.3	911885.4	205000.0	10.798	35.425
	662.000	80960.7	178487.8	911885.4	205000.0	11.263	36.953
	678.000	77468.2	170788.2	911885.4	205000.0	11.771	38.619
	687.127	75476.0	166396.1	911885.4	205000.0	12.082	39.638
CUTOFF	687.137	75473.7	166391.1	911885.4	205000.0	12.082	39.640

TABLE 4.1.1.4-II (Continued)

	TIME SEC.	INERTIAL VELOCITY MT/SEC.	INERTIAL VELOCITY FT/SEC.	RELATIVE VELOCITY MT/SEC.	RELATIVE VELOCITY FT/SEC.	THETA S DEG.	THETA R DEG.
LIFT OFF	.000	408.641	1340.687	.000	.000	.0000	90.0000
	3.000	409.177	1342.445	21.530	70.637	3.0162	89.8091
	16.000	411.376	1349.664	47.238	154.980	6.5937	89.8357
	24.000	418.862	1374.218	77.823	255.324	10.7001	87.8710
	32.000	435.279	1428.079	114.030	374.115	15.1078	84.2152
	43.000	477.763	1567.464	174.903	573.828	20.9564	77.6804
	51.000	527.000	1729.004	228.988	751.272	24.4000	71.9403
	59.000	592.422	1943.642	292.500	959.645	26.7673	65.8064
	67.000	670.617	2200.188	363.163	1191.480	27.8466	59.6052
10 KMS	70.161	705.136	2313.437	393.746	1291.816	27.9881	57.1845
	75.000	763.363	2504.472	445.492	1461.589	28.0010	53.5598
Q MAX	79.000	816.840	2679.923	493.412	1618.806	27.8559	50.6717
14 KMS	81.196	848.327	2783.224	521.823	1712.019	27.7232	49.1369
	81.196	848.327	2783.224	521.823	1712.019	27.7232	49.1369
	91.000	1008.397	3308.388	668.512	2193.282	26.7723	42.8015
	99.000	1163.049	3815.778	813.153	2667.825	25.6800	38.3023
	107.000	1339.596	4395.001	980.806	3217.868	24.4303	34.3940
	115.000	1539.059	5049.407	1172.566	3847.001	23.1220	31.0260
	123.000	1763.473	5785.673	1390.419	4561.744	21.8287	28.1379
	131.000	2015.916	6613.699	1637.334	5371.831	20.6004	25.6709
	139.000	2300.943	7549.025	1917.732	6291.771	19.4697	23.5726
	146.000	2582.170	8471.687	2195.543	7203.226	18.5767	22.0042
1 ENG CO	146.000	2582.170	8471.687	2195.543	7203.226	18.5767	22.0042
	146.000	2582.170	8471.687	2195.543	7203.226	18.5767	22.0042
	155.000	2766.240	9082.152	2377.462	7800.139	17.2635	20.2149
	163.000	2949.285	9676.132	2555.308	8383.558	16.1756	18.7557
	171.000	3146.881	10324.412	2750.069	9022.534	15.1653	17.4187
	179.000	3363.258	11034.310	2963.954	9724.258	14.2348	16.2019
	187.000	3601.279	11815.220	3199.791	10498.002	13.3861	15.1034
	195.000	3864.669	12679.361	3461.268	11355.867	12.6213	14.1211
	203.000	4153.371	13642.949	3753.297	12313.967	11.9423	13.2535
CUTOFF	210.952	4487.001	14721.131	4080.473	13387.377	11.3549	12.5036
SEPARATION	210.952	4487.001	14721.131	4080.473	13387.377	11.3549	12.5036
	210.952	4487.001	14721.131	4080.473	13387.377	11.3549	12.5036
END COAST	214.752	4479.995	14698.145	4072.791	13362.174	11.0505	12.1715

TABLE 4.1.1.4-II (Continued)

	TIME SEC.	INERTIAL VELOCITY MT/SEC.	INERTIAL VELOCITY FT/SEC.	RELATIVE VELOCITY MT/SEC.	RELATIVE VELOCITY FT/SEC.	THETA S DEG.	THETA R DEG.
	214.752	4477.995	14698.145	4072.791	13362.174	11.0505	12.1715
BOOSTER IMP.	575.334	4711.294	15457.002	4325.720	14191.994	-18.2929	-19.9897
START COV	214.752	4477.995	14698.145	4072.791	13362.174	11.0505	12.1715
	230.000	4530.551	14864.012	4121.090	13520.638	10.0471	11.0573
	246.000	4587.727	15051.598	4176.170	13701.345	9.0380	9.9373
	262.000	4647.055	15252.334	4235.661	13896.527	8.0747	8.8689
	278.000	4714.464	15467.402	4299.477	14105.897	7.1533	7.8533
	294.000	4733.892	15695.134	4367.536	14329.187	6.2893	6.8916
	310.000	4857.283	15935.758	4439.764	14566.156	5.4684	5.9844
	326.000	4934.589	16189.598	4516.096	14816.588	4.6938	5.1321
	342.000	5015.772	16455.944	4596.474	15080.297	3.9719	4.3348
	358.000	5100.801	16734.911	4680.853	15357.131	3.2966	3.5928
	374.000	5187.657	17026.435	4769.197	15646.972	2.6701	2.9057
	390.000	5282.332	17330.486	4861.481	15949.742	2.0923	2.2735
	406.000	5378.829	17647.076	4957.694	16265.401	1.5630	1.6958
	422.000	5479.162	17976.253	5057.838	16593.956	1.0821	1.1722
	438.000	5583.361	18318.113	5161.927	16935.456	.6473	.7023
	454.000	5691.468	18672.795	5269.993	17290.004	.2644	.2855
	466.352	5777.633	18955.470	5356.167	17572.725	-.0000	-.0001
	470.000	5803.542	19040.491	5382.083	17657.754	-.0048	-.0785
	486.000	5919.657	19421.447	5498.262	18038.917	-.3626	-.3904
	502.000	6039.907	19815.969	5618.613	18433.769	-.6051	-.6505
	518.000	6164.405	20224.427	5743.240	18842.652	-.8006	-.8593
	534.000	6293.287	20647.267	5872.272	19265.984	-.9492	-1.0173
	550.000	6426.712	21085.012	6005.860	19704.265	-1.0511	-1.1247
	566.000	6564.567	21538.278	6144.185	20158.087	-1.1062	-1.1819
	582.000	6707.971	22007.780	6287.459	20628.147	-1.1146	-1.1892
	598.000	6855.278	22494.350	6435.931	21115.259	-1.0764	-1.1467
	614.000	7010.080	22998.950	6589.888	21620.367	-.9913	-1.0545
	630.000	7169.718	23522.676	6749.665	22144.572	-.8571	-.9126
	646.000	7335.584	24066.378	6915.353	22687.150	-.6778	-.7211
	662.000	7508.135	24632.789	7088.302	23255.583	-.4529	-.4797
	678.000	7687.899	25222.766	7268.137	23845.595	-.1780	-.1883
	687.127	7793.906	25570.558	7374.173	24193.480	.0000	.0005
CUTOFF	687.137	7794.029	25570.963	7374.296	24193.885	.0000	.0007

TABLE 4.1.1.4-II (Continued)

	TIME SEC.	XXX MT.	XXX FT.	YYY MT.	YYY FT.	ZZZ MT.	ZZZ FT.
LIFT OFF	.000	.0	.0	.0	.0	-.0	-.0
	8.000	3269.1	10725.3	82.7	271.2	-.5	-1.5
	16.000	6537.9	21449.8	352.2	1155.6	-1.8	-6.0
	24.000	9816.6	32206.8	844.7	2771.3	-4.1	-13.5
	32.000	13140.7	43112.6	1600.1	5249.7	-7.3	-24.0
	43.000	17891.1	58697.9	3144.0	10314.8	-13.2	-43.3
	51.000	21590.7	70835.6	4684.6	15369.3	-18.6	-60.9
	59.000	25622.7	84064.1	6607.3	21677.4	-24.8	-81.5
	67.000	30110.8	98788.6	8911.2	29236.2	-32.0	-105.0
10 KMS	70.161	32036.0	105105.1	9919.8	32545.4	-35.1	-115.2
	75.000	35179.3	115417.5	11570.9	37962.2	-40.1	-131.6
Q MAX	77.000	37978.3	124600.7	13034.9	42765.3	-44.5	-146.0
14 KMS	81.196	39600.8	129923.9	13877.5	45529.8	-47.0	-154.2
	81.196	39600.8	129923.9	13877.5	45529.8	-47.0	-154.2
	91.000	47694.9	156479.2	17981.6	58994.7	-59.0	-193.6
	99.000	55503.7	182098.9	21751.3	71362.6	-69.8	-229.0
	107.000	64598.4	211937.0	25898.3	84968.1	-81.5	-267.4
	115.000	75172.0	246627.2	30416.5	99791.6	-94.1	-308.7
	123.000	87426.8	286833.2	35300.8	115816.2	-107.6	-353.0
	131.000	101581.7	333273.2	40550.8	133040.8	-122.0	-400.2
	139.000	117882.3	386753.0	46174.5	151491.0	-137.3	-450.4
	146.000	134129.1	440056.1	51416.7	168689.9	-151.4	-496.6
1 ENG CO	146.000	134129.1	440056.1	51416.7	168689.9	-151.4	-496.6
	146.000	134129.1	440056.1	51416.7	168689.9	-151.4	-496.6
	155.000	157188.3	515709.7	58294.0	191253.4	-170.5	-559.3
	163.000	179246.8	588079.9	64291.4	210929.8	-188.4	-618.2
	171.000	202899.4	665680.5	70170.9	230219.5	-207.3	-680.0
	179.000	228282.1	748957.0	75925.1	249098.1	-227.0	-744.7
	187.000	255550.5	838420.3	81548.2	267540.5	-247.6	-812.2
	195.000	284886.6	934667.2	87036.3	285552.1	-269.0	-882.6
	203.000	316506.4	1038408.8	92388.3	303111.1	-291.3	-955.9
CUTOFF	210.952	350459.5	1149801.5	97575.6	320129.8	-314.4	-1031.5
SEPARATION	210.952	350459.5	1149801.5	97575.6	320129.8	-314.4	-1031.5
	210.952	350459.5	1149801.5	97575.6	320129.8	-314.4	-1031.5
END COAST	214.752	367329.3	1205148.6	99955.2	327936.9	-325.7	-1068.6

TABLE 4.1.1.4-II (Continued)

	TIME	XXX		YYY		ZZZ	
	SEC.	MT.	FT.	MT.	FT.	MT.	FT.
	214.752	367329.3	1205148.6	99955.2	327936.9	-325.7	-1068.6
BOOSTER IMP.	575.334	1884347.6	6182242.8	-284484.7	-933348.9	-2225.6	-7301.9
START COV	214.752	367329.3	1205148.6	99955.2	327936.9	-325.7	-1068.6
	230.000	435506.7	1428827.7	108324.9	355396.5	-371.4	-1218.5
	246.000	508123.9	1667073.0	115159.8	377820.7	-419.2	-1375.4
	262.000	581851.8	1908962.5	119995.3	393685.2	-466.9	-1531.8
	278.000	656696.7	2154523.3	122825.0	402969.1	-514.4	-1687.6
	294.000	732673.6	2403784.7	123641.2	405647.0	-561.7	-1842.8
	310.000	805786.1	2656778.5	122434.8	401689.1	-608.6	-1997.4
	326.000	888040.7	2913539.2	119195.4	391061.2	-655.7	-2151.2
	342.000	967466.8	3174103.8	113911.2	373724.4	-702.3	-2304.2
	358.000	1048056.6	3438512.3	106568.8	349635.3	-748.7	-2456.4
	374.000	1129835.0	3706807.8	97153.7	318745.7	-794.8	-2607.7
	390.000	1212610.3	3979036.4	85649.6	281002.8	-840.7	-2758.1
	406.000	1296999.6	4255248.1	72039.0	236348.5	-886.2	-2907.5
	422.000	1382419.3	4535496.3	56302.6	184720.0	-931.4	-3055.8
	438.000	1469086.9	4819838.9	38419.7	126048.8	-976.3	-3203.1
	454.000	1557021.4	5108337.9	18367.7	60261.4	-1020.8	-3349.2
	466.352	1625783.9	5333936.7	1389.2	4557.9	-1055.0	-3461.1
	470.000	1646243.3	5401060.6	-3877.6	-12721.9	-1065.0	-3494.1
	486.000	1736774.6	5698079.3	-28342.3	-92986.6	-1108.6	-3637.7
	502.000	1828639.3	5999472.6	-55054.4	-180624.6	-1152.2	-3780.1
	518.000	1921863.2	6305325.5	-84044.0	-275735.0	-1195.1	-3921.1
	534.000	2016474.6	6615730.4	-115343.4	-378423.4	-1237.7	-4060.6
	550.000	2112504.1	6930787.8	-148987.4	-488803.6	-1279.8	-4198.8
	566.000	2209985.1	7250607.3	-185013.0	-606998.0	-1321.4	-4335.4
	582.000	2308954.2	7575309.0	-223460.4	-733137.9	-1362.6	-4470.5
	598.000	2409451.3	7905023.9	-264372.8	-867364.7	-1403.3	-4603.9
	614.000	2511520.5	8239890.6	-307796.4	-1009830.8	-1443.4	-4735.7
	630.000	2615210.2	8580086.0	-353781.6	-1160700.9	-1483.1	-4865.8
	646.000	2720574.0	8925767.8	-402382.8	-1320153.5	-1522.2	-4994.1
	662.000	2827671.3	9277136.8	-453658.9	-1488382.3	-1560.6	-5120.7
	678.000	2936568.1	9634409.6	-507674.4	-1665598.5	-1598.8	-5245.3
	687.127	2999520.7	9840947.1	-537740.6	-1770802.6	-1620.2	-5315.6
CUTOFF	687.137	2999593.4	9841185.5	-537778.0	-1770925.2	-1620.2	-5315.7

TABLE 4.1.1.4-II (Continued)

	TIME	ALTITUDE		DOWN RANGE		DYN. PRESSURE	
	SEC.	MT	FT	KM	NAUT. MI.	KG/M.SQ.	LB/F.SQ.
LIFT OFF	.000	-.125	-.410	.000	.000	.00	.00
	8.000	83.313	273.335	.000	.000	27.77	5.69
	16.000	355.438	1166.134	.000	.000	130.42	26.71
	24.000	852.063	2795.481	.000	.000	338.00	69.23
	32.000	1613.438	5293.430	.053	.029	674.32	138.11
	43.000	3168.813	10396.366	.310	.167	1357.21	277.98
	51.000	4720.875	15488.435	.733	.396	1983.54	406.26
	59.000	6658.563	21845.677	1.485	.802	2646.46	542.04
	67.000	8982.000	29468.504	2.689	1.452	3183.51	652.04
10 KMS	70.161	10000.000	32808.398	3.315	1.790	3340.13	684.12
	75.000	11067.563	36279.404	4.467	2.412	3512.62	719.44
Q MAX	77.000	13147.500	43134.842	5.617	3.033	3556.82	728.50
14 KMS	81.196	13999.938	45931.553	6.334	3.420	3530.95	723.20
	81.196	13999.938	45931.553	6.334	3.420	3530.95	723.20
	91.000	18159.188	59577.387	10.373	5.601	2932.70	600.67
	99.000	21991.750	72151.410	14.858	8.023	2234.66	457.70
	107.000	26223.750	86035.925	20.610	11.129	1627.45	333.33
	115.000	30856.938	101236.671	27.818	15.020	1126.37	230.70
	123.000	35896.188	117769.644	36.677	19.804	728.44	149.20
	131.000	41354.000	135675.852	47.399	25.593	457.53	93.71
	139.000	47255.125	155036.498	60.220	32.516	284.90	58.35
	146.000	52814.625	173276.328	73.375	39.619	192.23	39.37
1 ENG CO	146.000	52814.625	173276.328	73.375	39.619	192.23	39.37
	146.000	52814.625	173276.328	73.375	39.619	192.23	39.37
	155.000	60211.688	197544.904	92.393	49.888	100.10	20.50
	163.000	66782.625	219103.100	110.303	59.829	51.88	10.63
	171.000	73359.938	240682.209	130.746	70.597	24.15	4.95
	179.000	79958.000	262329.395	152.354	82.265	9.72	1.99
	187.000	86597.313	284111.914	175.775	94.911	3.00	.61
	195.000	93335.250	306119.586	201.182	108.630	.88	.18
	203.000	100113.938	328474.203	228.782	123.532	.27	.05
CUTOFF	210.952	107144.750	351196.684	258.635	139.652	.09	.02
SEPARATION	210.952	107144.750	351196.684	258.635	139.652	.09	.02
	210.952	107144.750	351196.684	258.635	139.652	.09	.02
END COAST	214.752	110353.375	362051.754	273.514	147.686	.05	.01

TABLE 4.1.1.4-II (Continued)

	TIME SEC.	ALTITUDE M	ALTITUDE FT	DOWN RANGE KM	DOWN RANGE NAUT. MI.	DYN. PRESSURE KG/M.SQ.	DYN. PRESSURE LB/F.SQ.
	214.752	110353.375	362051.754	273.514	147.686	.05	.01
BOOSTER IMP.	575.334	-.000	-.000	1677.776	905.927	1129149.70	231268.94
START COV	214.752	110353.375	362051.754	273.522	147.690	.05	.01
	230.000	122917.625	403273.047	333.606	180.133	.01	.00
	246.000	134995.813	442899.645	397.571	214.671	.00	.00
	262.000	145973.688	478916.297	462.521	249.741	.00	.00
	278.000	155884.313	511431.469	528.506	285.370	.00	.00
	294.000	164761.875	540557.328	595.574	321.584	.00	.00
	310.000	172641.125	566407.883	663.776	358.410	.00	.00
	326.000	179557.875	589100.633	733.160	395.875	.00	.00
	342.000	185549.250	608757.375	803.781	434.007	.00	.00
	358.000	190653.563	625503.813	875.693	472.836	.00	.00
	374.000	194910.813	639471.164	948.947	512.390	.00	.00
	390.000	198362.125	650794.367	1023.603	552.702	.00	.00
	406.000	201051.063	659616.344	1099.716	593.799	.00	.00
	422.000	203022.688	666084.930	1177.346	635.716	.00	.00
	438.000	204324.500	670355.969	1256.553	678.485	.00	.00
	454.000	205006.125	672592.266	1337.401	722.139	.00	.00
	466.352	205141.188	673035.391	1400.975	756.466	.00	.00
	470.000	205120.063	672966.078	1419.952	766.713	.00	.00
	486.000	204721.500	671658.461	1504.274	812.243	.00	.00
	502.000	203868.938	668801.336	1590.434	858.766	.00	.00
	518.000	202624.063	664777.102	1678.503	906.319	.00	.00
	534.000	201052.313	659620.445	1768.553	954.942	.00	.00
	550.000	199223.438	653620.195	1860.659	1004.675	.00	.00
	566.000	197211.313	647018.734	1954.698	1055.561	.00	.00
	582.000	195094.938	640075.250	2051.351	1107.641	.00	.00
	598.000	192958.438	633065.734	2150.100	1160.961	.00	.00
	614.000	190892.000	626286.086	2251.230	1215.567	.00	.00
	630.000	188991.688	620051.461	2354.832	1271.507	.00	.00
	646.000	187361.188	614702.055	2460.996	1328.832	.00	.00
	662.000	186111.750	610602.652	2569.821	1387.592	.00	.00
	678.000	185362.875	608145.914	2681.406	1447.843	.00	.00
	687.127	185209.063	607641.273	2746.334	1482.902	.00	.00
CUTOFF	687.137	185209.063	607641.273	2746.409	1482.942	.00	.00

TABLE 4.1.1.4-II (Continued)

	TIME SEC.	MACH	AZ. S DEG.	LATITUDE DEG.	LONGITUDE DEG.	ALPHA DEG.	CHI DEG.
LIFT OFF	.000	.000	90.0000	28.6080	.0000	.0000	90.0000
	8.000	.062	89.9914	28.6080	.0000	.0000	90.0000
	16.000	.137	89.9811	28.6080	.0000	.5087	89.4418
	24.000	.227	89.9695	28.6080	-.0001	.6359	87.1532
	32.000	.335	89.9580	28.6080	-.0006	.2121	83.8874
	43.000	.521	89.9463	28.6081	-.0032	.0000	77.5206
	51.000	.694	89.9449	28.6081	-.0075	.0000	71.7470
	59.000	.908	89.9521	28.6081	-.0152	.0000	65.5767
	67.000	1.167	89.9690	28.6082	-.0275	.0000	59.3352
10 KMS	70.161	1.286	89.9782	28.6082	-.0339	.0000	56.8972
	75.000	1.495	89.9947	28.6082	-.0457	.0000	53.2444
Q MAX	79.000	1.690	90.0104	28.6082	-.0574	.0000	50.3312
14 KMS	81.196	1.804	90.0199	28.6082	-.0648	.0000	48.7819
	81.196	1.804	90.0199	28.6082	-.0648	.0000	48.7819
	91.000	2.325	90.0686	28.6081	-.1061	.0000	42.3740
	99.000	2.764	90.1160	28.6080	-.1519	.0000	37.8051
	107.000	3.271	90.1701	28.6078	-.2107	.0000	33.8156
	115.000	3.846	90.2312	28.6075	-.2844	.0000	30.3534
	123.000	4.415	90.3000	28.6070	-.3750	.0000	27.3563
	131.000	5.020	90.3772	28.6062	-.4846	.0000	24.7635
	139.000	5.697	90.4641	28.6051	-.6157	.0000	22.5205
	146.000	6.514	90.5490	28.6039	-.7502	.0000	20.8080
1 ENG CO	146.000	6.514	90.5490	28.6039	-.7502	.0000	20.8080
	146.000	6.514	90.5490	28.6039	-.7502	.0000	20.8080
	155.000	7.457	90.6651	28.6017	-.9446	.0000	18.8147
	163.000	8.517	90.7755	28.5992	-1.1328	.0887	17.1606
	171.000	9.622	90.8932	28.5962	-1.3367	.0974	15.6149
	179.000	11.472	91.0190	28.5924	-1.5575	.1051	14.1744
	187.000	12.402	91.1534	28.5879	-1.7969	.1126	12.8359
	195.000	13.415	91.2973	28.5823	-2.0565	.1195	11.5958
	203.000	14.547	91.45[illegible]7	28.5757	-2.3385	.1253	10.4507
CUTOFF	210.952	15.815	91.6168	28.5677	-2.6435	.1305	9.4032
SEPARATION	210.952	15.815	91.6168	28.5677	-2.6435	.1305	9.4032
	210.952	15.815	91.6168	28.5677	-2.6435	.1305	9.4032
END COAST	214.752	15.785	91.6973	28.5634	-2.7955	.1387	8.9233

TABLE 4.1.1.4-II (Continued)

	TIME SEC.	MACH	AZ. S DEG.	LATITUDE DEG.	LONGITUDE DEG.	ALPHA DEG.	CHI DEG.
	214.752	15.785	91.6973	28.5634	-2.7955	.1387	8.9233
BOOSTER IMP.	575.334	12.471	99.1003	27.2367	-16.9889	.0000	-37.1899
START COV	214.752	15.785	91.6973	28.5634	-2.7955	10.3336	19.2569
	230.000	15.972	92.0247	28.5439	-3.4091	11.0158	18.2288
	246.000	16.186	92.3721	28.5195	-4.0621	11.6913	17.1503
	262.000	16.416	92.7236	28.4909	-4.7247	12.3240	16.0718
	278.000	16.664	93.0795	28.4577	-5.3975	12.9126	14.9928
	294.000	16.927	93.4399	28.4200	-6.0308	13.4564	13.9131
	310.000	17.207	93.8048	28.3774	-6.7751	13.9546	12.8321
	326.000	17.503	94.1745	28.3297	-7.4808	14.4070	11.7496
	342.000	17.815	94.5490	28.2767	-8.1983	14.8134	10.6650
	358.000	18.142	94.9235	28.2183	-8.9281	15.1738	9.5780
	374.000	18.484	95.3132	28.1540	-9.6706	15.4885	8.4883
	390.000	18.842	95.7031	28.0837	-10.4263	15.7577	7.3953
	406.000	19.215	96.0982	28.0070	-11.1956	15.9820	6.2988
	422.000	19.603	96.4987	27.9237	-11.9789	16.1618	5.1983
	438.000	20.006	96.9050	27.8333	-12.7769	16.2978	4.0935
	454.000	20.425	97.3108	27.7357	-13.5898	16.3907	2.9840
	466.352	20.759	97.6386	27.6551	-14.2280	16.4333	2.1240
	470.000	20.860	97.7700	27.6304	-14.4183	16.4411	1.8694
	486.000	21.310	98.1575	27.5171	-15.2627	16.4497	.7493
	502.000	21.776	98.5865	27.3953	-16.1235	16.4173	-.3766
	518.000	22.259	99.0214	27.2646	-17.0014	16.3446	-1.5086
	534.000	22.759	99.4621	27.1246	-17.8966	16.2323	-2.6472
	550.000	23.277	99.9087	26.9748	-18.8098	16.0811	-3.7926
	566.000	23.813	100.3611	26.8148	-19.7415	15.8917	-4.9452
	582.000	24.369	100.8194	26.6439	-20.6921	15.6646	-6.1052
	598.000	24.944	101.2835	26.4618	-21.6623	15.4007	-7.2731
	614.000	25.541	101.7533	26.2677	-22.6524	15.1003	-8.4491
	630.000	26.160	102.2288	26.0612	-23.6631	14.7642	-9.6334
	646.000	26.803	102.7097	25.8415	-24.6948	14.3927	-10.8263
	662.000	27.473	103.1961	25.6081	-25.7483	13.9865	-12.0281
	674.000	28.170	103.6876	25.3602	-26.8239	13.5459	-13.2389
	687.127	28.580	103.9701	25.2121	-27.4477	13.2794	-13.9338
CUTOFF	687.137	28.581	103.9703	25.2119	-27.4484	13.2791	-13.9346

TABLE 4.1.1.4-II (Continued)

	TIME SEC.	DXX MT/SEC.	DXX FT/SEC.	DYY MT/SEC.	DYY FT/SEC.	DZZ MT/SEC.	DZZ FT/SEC.
LIFT OFF	.000	408.641	1340.687	.000	.000	-.000	-.000
	8.000	408.621	1340.621	21.321	69.951	-.114	-.375
	16.000	408.705	1340.895	46.819	153.607	-.228	-.749
	24.000	411.698	1350.714	77.136	253.072	-.343	-1.124
	32.000	420.466	1379.483	112.585	369.373	-.457	-1.498
	43.000	446.637	1465.344	169.625	556.513	-.613	-2.012
	51.000	480.663	1576.978	216.083	708.935	-.727	-2.386
	59.000	530.005	1738.860	264.685	868.390	-.841	-2.759
	67.000	594.429	1950.226	310.453	1018.546	-.955	-3.132
10 KMS	70.161	624.316	2048.282	327.767	1075.416	-.999	-3.279
	75.000	675.965	2217.732	354.674	1163.629	-1.068	-3.504
Q MAX	79.000	724.443	2376.782	377.371	1238.095	-1.125	-3.689
14 KMS	81.196	753.372	2471.693	389.984	1279.474	-1.156	-3.791
	81.196	753.372	2471.693	389.984	1279.474	-1.156	-3.791
	91.000	903.662	2964.769	447.502	1468.183	-1.294	-4.246
	99.000	1052.504	3453.097	494.891	1623.658	-1.407	-4.615
	107.000	1225.182	4019.627	541.705	1777.247	-1.519	-4.984
	115.000	1422.421	4666.736	587.723	1928.226	-1.631	-5.352
	123.000	1645.820	5399.673	633.332	2077.862	-1.743	-5.718
	131.000	1898.005	6227.051	679.332	2228.780	-1.854	-6.084
	139.000	2183.082	7162.343	726.971	2385.075	-1.965	-6.448
	146.000	2464.267	8084.366	771.354	2530.689	-2.062	-6.766
1 ENG CO	146.000	2464.267	8084.866	771.354	2530.689	-2.062	-6.766
	146.000	2464.267	8084.866	771.354	2530.689	-2.062	-6.766
	155.000	2662.811	8736.255	756.695	2482.595	-2.186	-7.173
	163.000	2854.297	9364.493	742.471	2435.930	-2.299	-7.541
	171.000	3061.694	10044.926	727.245	2385.976	-2.410	-7.905
	179.000	3287.205	10784.792	711.182	2333.275	-2.519	-8.266
	187.000	3533.676	11593.427	694.504	2278.555	-2.628	-8.623
	195.000	3804.820	12483.006	677.498	2222.763	-2.736	-8.977
	203.000	4105.572	13469.725	660.543	2167.135	-2.844	-9.331
CUTOFF	210.952	4440.510	14568.602	644.236	2113.635	-2.950	-9.680
SEPARATION	210.952	4440.510	14568.602	644.236	2113.635	-2.950	-9.680
	210.952	4440.510	14568.602	644.236	2113.635	-2.950	-9.680
END COAST	214.752	4438.514	14562.054	608.223	1995.483	-3.001	-9.846

TABLE 4.1.1.4-II (Continued)

	TIME	DXX		DYY		DZZ	
	SEC.	MT/SEC.	FT/SEC.	MT/SEC.	FT/SEC.	MT/SEC.	FT/SEC.
	214.752	4438.514	14562.054	608.223	1995.483	-3.001	-9.846
BOOSTER IMP.	575.334	3836.063	12585.507	-2735.116	-8973.478	-7.426	-24.365
START COV	214.752	4438.514	14562.054	608.223	1995.483	-3.001	-9.846
	230.000	4504.023	14776.978	489.551	1606.138	-2.993	-9.821
	246.000	4573.203	15003.946	364.759	1196.716	-2.984	-9.791
	262.000	4642.875	15232.529	239.611	786.127	-2.974	-9.757
	278.000	4713.084	15462.375	114.023	374.092	-2.963	-9.721
	294.000	4783.876	15695.131	-12.090	-39.666	-2.951	-9.680
	310.000	4855.298	15929.457	-138.814	-455.425	-2.937	-9.637
	326.000	4927.401	16166.015	-266.233	-873.469	-2.923	-9.589
	342.000	5000.238	16404.980	-394.436	-1294.082	-2.908	-9.539
	358.000	5073.864	16646.536	-523.511	-1717.557	-2.891	-9.485
	374.000	5148.340	16890.881	-653.549	-2144.189	-2.874	-9.428
	390.000	5223.731	17138.224	-784.642	-2574.284	-2.855	-9.368
	406.000	5300.105	17388.795	-916.885	-3008.154	-2.836	-9.305
	422.000	5377.538	17642.841	-1050.379	-3446.125	-2.816	-9.238
	438.000	5456.112	17900.629	-1185.225	-3888.532	-2.794	-9.168
	454.000	5535.916	18162.452	-1321.530	-4335.727	-2.772	-9.094
	466.352	5598.425	18367.536	-1427.822	-4684.456	-2.754	-9.035
	470.000	5617.048	18428.634	-1459.406	-4788.077	-2.749	-9.018
	486.000	5699.616	18699.527	-1598.972	-5245.972	-2.724	-8.938
	502.000	5783.740	18975.523	-1740.353	-5709.821	-2.699	-8.855
	518.000	5869.551	19257.056	-1883.683	-6180.063	-2.673	-8.768
	534.000	5957.196	19544.607	-2029.105	-6657.168	-2.645	-8.679
	550.000	6046.841	19838.716	-2176.772	-7141.640	-2.617	-8.586
	566.000	6138.667	20139.984	-2326.852	-7634.028	-2.588	-8.491
	582.000	6232.582	20449.085	-2479.527	-8134.930	-2.558	-8.392
	598.000	6329.718	20766.791	-2634.997	-8645.003	-2.527	-8.290
	614.000	6429.437	21093.954	-2793.484	-9164.974	-2.495	-8.184
	630.000	6532.338	21431.555	-2955.234	-9695.650	-2.462	-8.076
	646.000	6638.759	21780.707	-3120.523	-10237.937	-2.428	-7.965
	662.000	6749.089	22142.680	-3289.663	-10792.859	-2.393	-7.851
	673.000	6863.772	22518.936	-3463.008	-11361.575	-2.357	-7.734
	687.127	6931.334	22740.597	-3563.927	-11692.675	-2.337	-7.666
CUTOFF	687.137	6931.413	22740.855	-3564.044	-11693.058	-2.337	-7.666

TABLE 4.1.1.4-II (Continued)

	TIME SEC.	SOLID THRUST NEWTONS	SOLID THRUST LBS.	SOLID WDOT KG/SEC	SOLID WDOT LB/SEC	LIQUID WDOT KG/SEC	LIQUID WDOT LB/SEC
LIFT OFF	.000	-.0	-.0	-.00	-.00	10476.78	23097.35
	8.000	-.0	-.0	-.00	-.00	10476.78	23097.35
	16.000	-.0	-.0	-.00	-.00	10476.78	23097.35
	24.000	-.0	-.0	-.00	-.00	10476.78	23097.35
	32.000	-.0	-.0	-.00	-.00	10476.78	23097.35
	43.000	-.0	-.0	-.00	-.00	10476.78	23097.35
	51.000	-.0	-.0	-.00	-.00	10476.78	23097.35
	59.000	-.0	-.0	-.00	-.00	10476.78	23097.35
	67.000	-.0	-.0	-.00	-.00	10476.78	23097.35
10 KMS	73.161	-.0	-.0	-.00	-.00	10476.78	23097.35
	75.000	-.0	-.0	-.00	-.00	10476.78	23097.35
Q MAX	79.000	-.0	-.0	-.00	-.00	10476.78	23097.35
14 KMS	81.196	-.0	-.0	-.00	-.00	10476.78	23097.35
	81.196	-.0	-.0	-.00	-.00	10476.78	23097.35
	91.000	-.0	-.0	-.00	-.00	10476.78	23097.35
	99.000	-.0	-.0	-.00	-.00	10476.78	23097.35
	107.000	-.0	-.0	-.00	-.00	10476.78	23097.35
	115.000	-.0	-.0	-.00	-.00	10476.78	23097.35
	123.000	-.0	-.0	-.00	-.00	10476.78	23097.35
	131.000	-.0	-.0	-.00	-.00	10476.78	23097.35
	139.000	-.0	-.0	-.00	-.00	10476.78	23097.35
	146.000	-.0	-.0	-.00	-.00	10476.78	23097.35
1 ENG CO	146.000	-.0	-.0	-.00	-.00	5238.39	11548.68
	148.000	-.0	-.0	-.00	-.00	5238.39	11548.68
	155.000	-.0	-.0	-.00	-.00	5238.39	11548.68
	163.000	-.0	-.0	-.00	-.00	5238.39	11548.68
	171.000	-.0	-.0	-.00	-.00	5238.39	11548.68
	179.000	-.0	-.0	-.00	-.00	5238.39	11548.68
	187.000	-.0	-.0	-.00	-.00	5238.39	11548.68
	195.000	-.0	-.0	-.00	-.00	5238.39	11548.68
	203.000	-.0	-.0	-.00	-.00	5238.39	11548.68
CUTOFF	210.952	-.0	-.0	-.00	-.00	5238.39	11548.68
SEPARATION	210.952	-.0	-.0	-.00	-.00	-.00	-.00
	210.952	-.0	-.0	-.00	-.00	-.00	-.00
END COAST	214.752	-.0	-.0	-.00	-.00	-.00	-.00

TABLE 4.1.1.4-II (Continued)

	TIME SEC.	SOLID THRUST NEWTONS	SOLID THRUST LBS.	SOLID WDOT KG/SEC	SOLID WDOT LB/SEC	LIQUID WDOT KG/SEC	LIQUID WDOT LB/SEC
	214.752	-.0	-.0	-.00	-.00	-.00	-.00
BOOSTER IMP.	575.334	-.0	-.0	-.00	-.00	-.00	-.00
START COV	214.752	-.0	-.0	-.00	-.00	218.28	481.22
	230.000	-.0	-.0	-.00	-.00	218.28	481.22
	246.000	-.0	-.0	-.00	-.00	218.28	481.22
	262.000	-.0	-.0	-.00	-.00	218.28	481.22
	278.000	-.0	-.0	-.00	-.00	218.28	481.22
	294.000	-.0	-.0	-.00	-.00	218.28	481.22
	310.000	-.0	-.0	-.00	-.00	218.28	481.22
	326.000	-.0	-.0	-.00	-.00	218.28	481.22
	342.000	-.0	-.0	-.00	-.00	218.28	481.22
	358.000	-.0	-.0	-.00	-.00	218.28	481.22
	374.000	-.0	-.0	-.00	-.00	218.28	481.22
	390.000	-.0	-.0	-.00	-.00	218.28	481.22
	406.000	-.0	-.0	-.00	-.00	218.28	481.22
	422.000	-.0	-.0	-.00	-.00	218.28	481.22
	438.000	-.0	-.0	-.00	-.00	218.28	481.22
	454.000	-.0	-.0	-.00	-.00	218.28	481.22
	465.352	-.0	-.0	-.00	-.00	218.28	481.22
	470.000	-.0	-.0	-.00	-.00	218.28	481.22
	486.000	-.0	-.0	-.00	-.00	218.28	481.22
	502.000	-.0	-.0	-.00	-.00	218.28	481.22
	518.000	-.0	-.0	-.00	-.00	218.28	481.22
	534.000	-.0	-.0	-.00	-.00	218.28	481.22
	550.000	-.0	-.0	-.00	-.00	218.28	481.22
	566.000	-.0	-.0	-.00	-.00	218.28	481.22
	582.000	-.0	-.0	-.00	-.00	218.28	481.22
	598.000	-.0	-.0	-.00	-.00	218.28	481.22
	614.000	-.0	-.0	-.00	-.00	218.28	481.22
	630.000	-.0	-.0	-.00	-.00	218.28	481.22
	646.000	-.0	-.0	-.00	-.00	218.28	481.22
	662.000	-.0	-.0	-.00	-.00	218.28	481.22
	678.000	-.0	-.0	-.00	-.00	218.28	481.22
	687.127	-.0	-.0	-.00	-.00	218.28	481.22
CUTOFF	687.137	-.0	-.0	-.00	-.00	218.28	481.22

617 LINES OUTPUT THIS JOB

SPOOK

SPOOK ENDSPOOK

TABLE 4.1.1.4-III EXPLANATION OF TRAJECTORY TABLE DATA

COLUMN	DEFINITION
Mass	Vehicle weight in kilograms
Weight	Vehicle weight in pounds
Thrust	Operating stage thrust
Longit. Accel.	Vehicle acceleration along the vehicle longitudinal axis
Inertial Velocity	Vehicle velocity in the space fixed coordinate system
Relative Velocity	Vehicle velocity in the earth fixed coordinate system
Theta S	Flight path angle, measured from the local horizontal to the inertial velocity vector
Theta R	Flight, path angle, measured from the local horizontal to the relative velocity vector
XXX	Displacement along the space-fixed X-axis, which has its origin at the launch point and lies in the horizontal plane pointing in the direction of the aiming azimuth
YYY	Displacement along the space-fixed Y-axis, which has its origin at the launch point and is vertical to the horizontal plane (goedetic).
ZZZ	Displacement along the space-fixed Z-axis, which has its origin at the launch point and completes the right-handed coordinate system
Altitude	Altitude above earth surface
Down Range	Distance measured along earth's surface from launch site (rotating earth)

TABLE 4.1.1.4-III (Continued)

Dyn. Pressure	Dynamic pressure doe to the relative velocity
Mach	Mach number, based on the relative velocity
AZ. S	Azimuth angle, measured in the space-fixed system from North to South over East
Latitude	Goedetic latitude of vehicle
Longitude	Longitude, measured positive from the Greenwich meridian due-West
Alpha	Angle of attack, measured from vehicle longitudinal axis to relative velocity vector
Chi	Thrust vector angle, measured from vehicle longitudinal axis to the horizontal at space-fixed launch point
DXX	Velocity component along space-fixed X-axis
DYY	Velocity component along space-fixed Y-axis

TABLE 4.1.1.4-IV

INT-20, RETROFIT BASELINE
100 N.M. CIRCULAR ORBIT, LAUNCH AZ. = 90°

	Liftoff Weight	lbs	4,870,400
	Sea Level Thrust	lbs	6,088,000
	Sea Level Specific Impulse	sec	263.58
	Propellant Consumed	lbs	4,122,325
S-IC	Stage Weight at Separation*	lbs	361,138
	Thrust-to-Weight Ratio at Liftoff		1.25
	Accel. at First Engine Shutdown	g's	3.68
	Accel. at Final Engine Shutdown	g's	4.68
	Weight at Ignition	lbs	386,937
	Vacuum Thrust	lbs	205,000
	Vacuum Specific Impulse	sec	426
S-IVB	Propellant Capacity	lbs	230,000
	Propellant Consumed	lbs	227,403
	Stage Weight at Separation	lbs	27,504
	Gross Payload	lbs	132,030
	Weight to be Subtracted		
	Astrionics Equipment	lbs	4,183
	Flight Performance Reserves (3/4%)	lbs	2,597
	Net Payload	lbs	125,250

* Includes 28,503 lbs of ballast

TABLE 4.1.1.4-V

INT-20, PRELIMINARY BASE LINE
100 N.M. CIRCULAR ORBIT, LAUNCH AZ. = 90^o

Liftoff Weight	lbs	4,870,400
Sea Level Thrust	lbs	6,088,000
Sea Level Specific Impulse	sec	263.58
Propellant Consumed	lbs	4,122,325
Stage Weight at Separation*	lbs	354,362
Thrust-to-Weight Ratio at Liftoff		1.25
Weight at Ignition	lbs	393,713
Vacuum Thrust	lbs	205,000
Vacuum Specific Impulse	sec	426
Propellant Capacity	lbs	230,000
Propellant Consumed	lbs	227,322
Stage Weight at Separation	lbs	27,504
Gross Payload	lbs	138,887
Weight to be Subtracted		
Astrionics Equipment	lbs	4,183
Flight Performance Reserves (3/4%)	lbs	2,678
Net Payload	lbs	132,026

* Includes 21,727 lbs of ballast

TABLE 4.1.1.4-VI INT-20 RETROFIT TRAJECTORY

	TIME SEC.	MASS KGS.	WEIGHT LBS.	THRUST NEWTØNS	THRUST LBS.	LØNGIT. ACCEL. MT/S.SQ.	LØNGIT. ACCEL. FT/S.SQ.
LIFT ØFF	.000	2209176.4	4870400.2	27080974.5	6088045.1	12.242	40.165
	8.000	2125362.1	4685621.3	27119343.5	6096670.9	12.734	41.777
	16.000	2041547.8	4500842.4	27242467.3	6124350.2	13.295	43.617
	24.000	1957733.4	4316063.3	27459273.3	6173090.1	13.940	45.735
	32.000	1873919.0	4131284.3	27772375.3	6243478.2	14.655	48.080
	43.000	1758674.3	3877213.2	28343532.0	6371879.4	15.763	51.717
	51.000	1674860.0	3692434.3	28831206.3	6481512.9	16.648	54.618
	59.000	1591045.7	3507655.4	29341301.5	6596186.9	17.446	57.236
	67.000	1507231.5	3322876.5	29832255.5	6706557.8	17.828	58.491
10 KMS	69.911	1476730.0	3255632.4	29997822.3	6743778.6	18.244	59.855
	75.000	1423417.1	3138097.6	30263836.3	6803580.9	19.308	63.347
G MAX	79.000	1381510.0	3045708.2	30447526.3	6844876.1	20.215	66.322
14 KMS	80.666	1363848.8	3006771.8	30517371.3	6860577.8	20.629	67.679
	80.666	1363848.8	3006771.8	30517371.3	6860577.8	20.629	67.679
	91.000	1255788.6	2768539.9	30838522.8	6932775.6	23.393	76.749
	99.000	1171974.3	2583761.0	30974348.8	6963310.5	25.645	84.138
	107.000	1088160.0	2398982.2	31048228.5	6979919.3	28.023	91.940
	115.000	1004345.8	2214203.4	31086688.5	6988565.5	30.640	100.525
	123.000	920531.5	2029424.5	31105350.5	6992760.9	33.609	110.267
	129.000	857670.8	1890840.4	31112390.5	6994343.5	36.157	118.625
ENG CØ	129.000	857670.8	1890840.4	15556195.3	3497171.8	18.019	59.118
	129.000	857670.8	1890840.4	15556195.3	3497171.8	18.019	59.118
	139.000	805286.9	1775353.6	15558898.5	3497779.5	19.270	63.221
	147.000	763379.7	1682964.2	15559793.4	3497980.7	20.356	66.785
	155.000	721472.6	1590574.8	15560236.9	3498080.3	21.555	70.718
	163.000	679565.5	1498185.4	15560444.0	3498126.9	22.889	75.094
	171.000	637658.3	1405796.0	15560532.0	3498146.7	24.398	80.047
	179.000	595751.2	1313406.5	15560564.9	3498154.1	26.117	85.687
	187.000	553844.0	1221017.1	15560574.9	3498156.3	28.095	92.175
	195.000	511936.9	1128627.7	15560577.6	3498156.9	30.395	99.722
	203.000	470029.8	1036238.3	15560578.4	3498157.1	33.105	108.614
	211.000	428122.7	943848.9	15560578.5	3498157.2	36.346	119.246
	219.000	386215.5	851459.5	15560578.8	3498157.2	40.290	132.185
	227.000	344308.4	759070.0	15560578.8	3498157.2	45.194	148.273
CUTØFF	227.952	339320.5	748073.7	15560578.8	3498157.2	45.858	150.453

TABLE 4.1.1.4-VI (Continued)

	TIME SEC.	MASS KGS.	WEIGHT LBS.	THRUST NEWTØNS	THRUST LBS.	LØNGIT. ACCEL. MT/S.SQ.	LØNGIT. ACCEL. FT/S.SQ.
SEPARATIØN	227.952	175511.2	386936.0	.0	.0	-.000	-.000
END CØAST	231.752	175511.2	386936.0	.0	.0	-.000	-.000
	231.752	175511.2	386936.0	.0	.0	-.000	-.000
BØØSTER IMP.	586.807	163809.3	361137.7	.0	.0	-.000	-.000
START CØV	231.752	175511.2	386936.0	911885.4	205000.0	5.196	17.046
	247.000	172183.0	379598.4	911885.4	205000.0	5.296	17.375
	263.000	168690.5	371898.9	911885.4	205000.0	5.406	17.735
	279.000	165198.1	364199.4	911885.4	205000.0	5.520	18.110
	295.000	161705.6	356499.9	911885.4	205000.0	5.639	18.501
	311.000	158213.2	348800.3	911885.4	205000.0	5.764	18.910
	327.000	154720.7	341100.8	911885.4	205000.0	5.894	19.336
	343.000	151228.3	333401.3	911885.4	205000.0	6.030	19.783
	359.000	147735.8	325701.7	911885.4	205000.0	6.172	20.251
	375.000	144243.4	318002.2	911885.4	205000.0	6.322	20.741
	391.000	140750.9	310302.7	911885.4	205000.0	6.479	21.256
	407.000	137258.5	302603.1	911885.4	205000.0	6.644	21.796
	423.000	133766.0	294903.6	911885.4	205000.0	6.817	22.366
	439.000	130273.6	287204.1	911885.4	205000.0	7.000	22.965
	455.000	126781.1	279504.5	911885.4	205000.0	7.193	23.598
	468.202	123899.5	273151.6	911885.4	205000.0	7.360	24.147
	471.000	123288.7	271805.0	911885.4	205000.0	7.396	24.266
	487.000	119796.2	264105.5	911885.4	205000.0	7.612	24.974
	503.000	116303.8	256406.0	911885.4	205000.0	7.841	25.724
	519.000	112811.3	248706.4	911885.4	205000.0	8.083	26.520
	535.000	109318.9	241006.9	911885.4	205000.0	8.342	27.367
	551.000	105826.4	233307.4	911885.4	205000.0	8.617	28.270
	567.000	102334.0	225607.8	911885.4	205000.0	8.911	29.235
	583.000	98841.5	217908.3	911885.4	205000.0	9.226	30.268
	599.000	95349.1	210208.8	911885.4	205000.0	9.564	31.377
	615.000	91856.6	202509.2	911885.4	205000.0	9.927	32.570
	631.000	88364.2	194809.7	911885.4	205000.0	10.320	33.857
	647.000	84871.8	187110.2	911885.4	205000.0	10.744	35.250
	663.000	81379.3	179410.7	911885.4	205000.0	11.205	36.763
	679.000	77886.9	171711.1	911885.4	205000.0	11.708	38.411
	695.000	74394.4	164011.6	911885.4	205000.0	12.257	40.215

TABLE 4.1.1.4-VI (Continued)

	TIME	MASS	WEIGHT	THRUST		LØNGIT. ACCEL.	
	SEC.	KGS.	LBS.	NEWTØNS	LBS.	MT/S.SQ.	FT/S.SQ.
	704.325	72359.0	159524.3	911885.4	205000.0	12.602	41.346
CUTØFF	704.306	72363.1	159533.3	911885.4	205000.0	12.602	41.344

TABLE 4.1.1.4-VI (Continued)

	TIME SEC.	INERTIAL VELØCITY MT/SEC.	INERTIAL VELØCITY FT/SEC.	RELATIVE VELØCITY MT/SEC.	RELATIVE VELØCITY FT/SEC.	THETA S DEG.	THETA R DEG.
LIFT ØFF	.000	438.641	1340.687	.000	.000	.0000	90.0000
	8.000	409.177	1342.445	21.530	70.637	3.0162	89.8091
	16.000	411.357	1349.596	47.238	154.981	6.5941	89.8321
	24.000	418.567	1373.251	77.820	255.317	10.7088	88.0923
	32.000	434.213	1424.583	113.993	373.993	15.1555	84.7801
	43.000	474.537	1556.880	174.623	572.911	21.1664	78.8792
	51.000	521.336	1710.420	228.235	748.802	24.8435	73.6764
	59.000	583.784	1915.303	290.915	954.444	27.5364	68.0840
	67.000	658.872	2161.652	360.449	1182.575	28.9971	62.3886
10 KMS	69.911	689.397	2261.800	387.989	1272.931	29.2747	60.3279
	75.000	748.308	2455.079	441.074	1447.093	29.5476	56.7884
Q MAX	79.000	800.105	2625.018	487.967	1600.942	29.6032	54.0942
14 KMS	80.666	823.427	2701.531	509.171	1670.509	29.5884	52.9883
	80.666	823.427	2701.531	509.171	1670.509	29.5884	52.9883
	91.000	986.633	3236.986	659.353	2163.232	29.0948	46.6870
	99.000	1137.858	3733.130	800.940	2627.756	28.3409	42.4083
	107.000	1310.797	4300.515	965.041	3166.144	27.3845	38.6641
	115.000	1506.430	4942.357	1152.798	3782.145	26.3262	35.4170
	123.000	1726.736	5665.146	1366.221	4482.352	25.2445	32.6171
	129.000	1909.943	6266.218	1544.890	5068.537	24.4522	30.7802
ENG CØ	129.000	1909.943	6266.218	1544.890	5068.537	24.4522	30.7802
	129.000	1909.943	6266.218	1544.890	5068.537	24.4522	30.7802
	139.000	2056.737	6747.825	1683.944	5524.752	22.5978	27.9907
	147.000	2185.772	7171.166	1807.491	5930.088	21.1780	25.9046
	155.000	2325.842	7630.717	1942.664	6373.568	19.8215	23.9522
	163.000	2477.831	8129.366	2090.304	6857.954	18.5335	22.1348
	171.000	2642.901	8670.937	2251.534	7386.924	17.3188	20.4524
	179.000	2822.470	9260.072	2427.725	7964.976	16.1804	18.9034
	187.000	3018.290	9902.527	2620.583	8597.713	15.1220	17.4856
	195.000	3222.561	10605.516	2832.265	9292.208	14.1457	16.1963
	203.000	3468.061	11378.219	3065.526	10057.501	13.2536	15.0326
	211.000	3728.466	12232.501	3323.945	10905.331	12.4478	13.9918
	219.000	4018.490	13184.023	3612.258	11851.240	11.7302	13.0714
	227.000	4344.614	14253.982	3936.894	12916.319	11.1029	12.2698
CUTØFF	227.952	4386.224	14390.497	3978.340	13052.298	11.0344	12.1822

TABLE 4.1.1.4-VI (Continued)

	TIME SEC.	INERTIAL VELØCITY MT/SEC.	INERTIAL VELØCITY FT/SEC.	RELATIVE VELØCITY MT/SEC.	RELATIVE VELØCITY FT/SEC.	THETA S DEG.	THETA R DEG.
SEPARATIØN	227.952	4386.224	14390.497	3978.340	13052.298	11.0344	12.1822
END CØAST	231.752	4379.449	14368.270	3970.893	13027.863	10.7176	11.8355
	231.752	4379.449	14368.270	3970.893	13027.863	10.7176	11.8355
BØØSTER IMP.	586.807	4644.266	15237.092	4260.869	13979.228	-19.1559	-20.9569
START CØV	231.752	4379.449	14368.270	3970.893	13027.863	10.7176	11.8355
	247.000	4431.851	14540.195	4021.101	13192.590	9.7007	10.7028
	263.000	4491.003	14734.261	4078.227	13380.011	8.6798	9.5662
	279.000	4554.344	14942.073	4139.808	13582.047	7.7071	8.4842
	295.000	4621.806	15163.406	4205.755	13798.410	6.7834	7.4581
	311.000	4693.328	15398.056	4275.988	14028.831	5.9095	6.4886
	327.000	4768.854	15645.847	4350.432	14273.071	5.0857	5.5764
	343.000	4848.342	15906.633	4429.025	14530.921	4.3124	4.7216
	359.000	4931.754	16180.296	4511.711	14802.203	3.5897	3.9244
	375.000	5019.067	16466.754	4598.449	15086.776	2.9175	3.1846
	391.000	5110.265	16765.961	4689.208	15384.539	2.2959	2.5021
	407.000	5205.347	17077.909	4783.968	15695.431	1.7246	1.8766
	423.000	5304.322	17402.631	4882.724	16019.437	1.2035	1.3074
	439.000	5407.215	17740.205	4985.488	16356.587	.7323	.7942
	455.000	5514.063	18090.757	5092.282	16706.963	.3106	.3364
	468.202	5605.241	18389.897	5183.465	17006.118	-.0000	-.0001
	471.000	5624.920	18454.463	5203.150	17070.703	-.0008	-.0667
	487.000	5739.857	18831.553	5318.151	17448.001	-.3849	-.4155
	503.000	5858.964	19222.321	5437.362	17839.115	-.6595	-.7106
	519.000	5982.347	19627.124	5560.884	18244.371	-.8855	-.9526
	535.000	6110.140	20046.392	5688.840	18664.173	-1.0633	-1.1421
	551.000	6242.498	20480.637	5821.377	19099.004	-1.1930	-1.2794
	567.000	6379.604	20930.458	5958.670	19549.442	-1.2748	-1.3649
	583.000	6521.671	21396.558	6100.928	20016.168	-1.3088	-1.3991
	599.000	6668.948	21879.751	6248.393	20499.976	-1.2949	-1.3821
	615.000	6821.724	22380.983	6401.348	21001.796	-1.2332	-1.3142
	631.000	6980.332	22901.351	6560.121	21522.707	-1.1235	-1.1954
	647.000	7145.158	23442.121	6725.096	22063.962	-.9656	-1.0259
	663.000	7316.652	24004.762	6896.715	22627.018	-.7592	-.8055
	679.000	7495.331	24590.981	7075.496	23213.570	-.5041	-.5340
	695.000	7681.802	25202.763	7262.040	23825.592	-.1998	-.2113

TABLE 4.1.1.4-VI (Continued)

	TIME SEC.	INERTIAL VELØCITY MT/SEC.	FT/SEC.	RELATIVE VELØCITY MT/SEC.	FT/SEC.	THETA S DEG.	THETA R DEG.
	704.325	7794.348	25572.006	7374.614	24194.929	.COCO	.CC05
CUTØFF	7C4.3C6	7794.118	25571.252	7374.385	24194.175	.CCCC	.CCC1

TABLE 4.1.1.4-VI (Continued)

	TIME	XXX		YYY		ZZZ	
	SEC.	MT.	FT.	MT.	FT.	MT.	FT.
LIFT ØFF	.000	.0	.0	.0	.0	-.0	-.0
	8.000	3269.1	10725.3	82.7	271.2	-.5	-1.5
	16.000	6537.9	21449.7	352.2	1155.6	-1.8	-6.0
	24.000	9815.6	32203.3	844.7	2771.4	-4.1	-13.5
	32.000	13134.4	43091.9	1600.4	5250.7	-7.3	-24.0
	43.000	17860.4	58597.1	3146.7	10324.0	-13.2	-43.3
	51.000	21518.9	70600.1	4694.2	15401.1	-18.6	-60.9
	59.000	25479.3	83593.4	6634.2	21765.7	-24.8	-81.5
	67.000	29856.0	97952.9	8974.8	29444.8	-32.0	-105.0
10 KMS	69.911	31574.3	103590.3	9922.1	32553.0	-34.9	-114.3
	75.000	34766.6	114063.7	11701.6	38391.0	-40.1	-131.6
G MAX	79.000	37466.8	122922.6	13214.6	43354.9	-44.5	-146.0
14 KMS	80.666	38660.6	126839.2	13883.3	45548.9	-46.4	-152.2
	80.666	38660.6	126839.2	13883.3	45548.9	-46.4	-152.2
	91.000	46797.8	153535.9	18394.7	60350.1	-59.0	-193.6
	99.000	54268.8	178047.4	22413.8	73536.0	-69.8	-229.0
	107.000	62956.2	206549.2	26905.3	88271.9	-81.5	-267.4
	115.000	73049.6	239664.1	31880.6	104595.1	-94.1	-308.7
	123.000	84748.2	278045.4	37352.5	122547.6	-107.6	-353.0
	129.000	94704.5	310710.3	41793.3	137117.1	-118.3	-388.1
ENG CØ	129.000	94704.5	310710.3	41793.3	137117.1	-118.3	-388.1
	129.000	94704.5	310710.3	41793.3	137117.1	-118.3	-388.1
	139.000	113004.5	370749.6	49403.4	162084.8	-137.3	-450.3
	147.000	128861.6	422774.2	55426.3	181844.9	-153.4	-503.4
	155.000	145889.9	478641.4	61377.5	201369.9	-170.5	-559.3
	163.000	164177.8	538641.2	67244.1	220617.1	-188.4	-618.2
	171.000	183821.5	603088.8	73013.1	239544.4	-207.3	-680.0
	179.000	204926.7	672331.5	78672.5	258111.9	-227.0	-744.6
	187.000	227611.4	746756.7	84210.8	276282.1	-247.5	-812.2
	195.000	252009.3	826802.1	89617.7	294021.4	-269.0	-882.5
	203.000	278273.3	912970.0	94884.5	311301.0	-291.3	-955.7
	211.000	306581.8	1005845.7	100004.2	328097.9	-314.5	-1031.7
	219.000	337146.4	1106123.5	104972.1	344396.7	-338.5	-1110.5
	227.000	370223.2	1214643.1	109786.4	360191.7	-363.4	-1192.1
CUTØFF	227.952	374341.5	1228154.6	110349.3	362038.3	-366.4	-1202.0

TABLE 4.1.1.4-VI (Continued)

	TIME	XXX		YYY		ZZZ	
	SEC.	MT.	FT.	MT.	FT.	MT.	FT.
SEPARATIØN	227.952	374341.5	1228154.6	110349.3	362038.3	-366.4	-1202.0
END CØAST	231.752	390853.7	1282328.4	112523.1	369170.1	-378.5	-1241.9
	231.752	390853.7	1282328.4	112523.1	369170.1	-378.5	-1241.9
BØØSTER IMP.	586.807	1852270.8	6077004.1	-274665.3	-901133.0	-2308.0	-7572.3
START CØV	231.752	390853.7	1282328.4	112523.1	369170.1	-378.5	-1241.9
	247.000	457596.0	1501299.2	120086.1	393983.3	-427.6	-1403.0
	263.000	528715.8	1734631.7	126113.8	413759.1	-479.0	-1571.5
	279.000	600956.1	1971640.9	130182.1	427106.7	-530.2	-1739.5
	295.000	674327.5	2212360.5	132284.8	434005.2	-581.2	-1906.8
	311.000	748840.9	2456826.9	132413.9	434429.0	-632.0	-2073.4
	327.000	824508.2	2705079.3	130560.5	428348.2	-682.5	-2239.3
	343.000	901342.2	2957159.5	126714.0	415728.4	-732.8	-2404.3
	359.000	979356.7	3213112.4	120862.5	396530.5	-782.9	-2568.5
	375.000	1058566.1	3472986.0	112992.6	370710.7	-832.6	-2731.8
	391.000	1138986.3	3736831.8	103089.6	338220.3	-882.1	-2894.1
	407.000	1220634.0	4004704.8	91137.0	299005.9	-931.3	-3055.3
	423.000	1303527.2	4276664.1	77117.1	253008.8	-980.1	-3215.5
	439.000	1387685.2	4552772.9	61010.3	200165.1	-1028.5	-3374.5
	455.000	1473128.6	4833099.1	42795.6	140405.5	-1076.6	-3532.2
	468.202	1544612.9	5067627.4	26163.0	85836.5	-1116.0	-3661.5
	471.000	1559879.7	5117715.6	22450.0	73654.9	-1124.3	-3688.8
	487.000	1647962.4	5406700.8	-51.1	-167.7	-1171.6	-3844.0
	503.000	1737402.4	5700139.2	-24734.5	-81149.8	-1218.5	-3997.8
	519.000	1828227.6	5998122.1	-51628.7	-169385.4	-1265.0	-4150.2
	535.000	1920468.1	6300748.2	-80764.6	-264975.8	-1311.0	-4301.2
	551.000	2014156.3	6608124.4	-112175.5	-368030.0	-1356.5	-4450.5
	567.000	2109327.9	6920367.0	-145897.1	-478665.1	-1401.6	-4598.3
	583.000	2206021.3	7237602.5	-181967.9	-597007.5	-1446.1	-4744.5
	599.000	2304278.5	7559968.9	-220429.3	-723193.2	-1490.1	-4888.9
	615.000	2404145.8	7887617.6	-261326.1	-857369.0	-1533.6	-5031.6
	631.000	2505673.7	8220714.3	-304706.7	-999693.9	-1576.6	-5172.4
	647.000	2608917.9	8559441.8	-350623.6	-1150339.9	-1618.9	-5311.5
	663.000	2713939.8	8904001.9	-399133.9	-1309494.3	-1660.7	-5448.5
	679.000	2820807.7	9254618.4	-450299.7	-1477361.1	-1701.9	-5583.7
	695.000	2929597.3	9611539.8	-504189.1	-1654163.6	-1742.5	-5716.8

TABLE 4.1.1.4-VI (Continued)

	TIME	XXX		YYY		ZZZ	
	SEC.	MT.	FT.	MT.	FT.	MT.	FT.
	704.325	2993919.8	9822571.6	-536881.6	-1761422.6	-1765.8	-5793.4
CUTØFF	704.306	2993789.6	9822144.3	-536814.8	-1761203.4	-1765.8	-5793.2

TABLE 4.1.1.4-VI (Continued)

	TIME SEC.	ALTITUDE MT	ALTITUDE FT	DØWN RANGE KM	DØWN RANGE NAUT. MI.	DYN. PRESSURE KG/M.SQ.	DYN. PRESSURE LB/F.SQ.
LIFT ØFF	.000	-.125	-.410	.000	.000	.00	.00
	8.000	83.313	273.335	.000	.000	27.77	5.69
	16.000	355.438	1166.134	.000	.000	130.42	26.71
	24.000	852.063	2795.481	.000	.000	337.98	69.22
	32.000	1613.750	5294.455	.045	.024	673.86	138.02
	43.000	3171.500	10405.184	.278	.150	1352.51	277.02
	51.000	4730.375	15519.603	.661	.357	1968.59	403.20
	59.000	6684.813	21931.799	1.342	.725	2610.68	534.71
	67.000	9044.375	29673.146	2.434	1.314	3114.69	637.94
10 KMS	69.911	10000.000	32808.398	2.956	1.596	3243.19	664.26
	75.000	11796.000	38700.787	4.054	2.189	3388.85	694.09
G MAX	79.000	13324.188	43714.525	5.106	2.757	3395.67	695.49
14 KMS	80.686	14000.000	45931.758	5.604	3.026	3361.77	688.55
	80.686	14000.000	45931.758	5.604	3.026	3361.77	688.55
	91.000	18565.750	60911.253	9.476	5.116	2654.96	543.78
	99.000	22643.563	74289.902	13.622	7.355	1942.37	398.04
	107.000	27214.375	89286.007	18.965	10.240	1347.55	276.00
	115.000	32296.500	105959.646	25.689	13.871	869.60	178.11
	123.000	37911.813	124382.586	33.987	18.351	521.15	106.74
	129.000	42491.250	139406.986	41.366	22.336	348.14	71.31
ENG CØ	129.000	42491.250	139406.986	41.366	22.336	348.14	71.31
	129.000	42491.250	139406.986	41.366	22.336	348.14	71.31
	139.000	50395.938	165341.002	55.323	29.872	150.74	30.87
	147.000	56715.750	186075.295	67.663	36.535	85.49	17.51
	155.000	63028.688	206787.031	81.134	43.809	47.96	9.82
	163.000	69333.250	227471.291	95.820	51.739	24.74	5.07
	171.000	75629.688	248128.895	111.813	60.374	11.46	2.35
	179.000	81921.313	268770.707	129.214	69.770	4.41	.90
	187.000	88215.000	289419.289	148.135	79.986	1.45	.30
	195.000	94521.875	310111.137	168.705	91.093	.46	.09
	203.000	100858.813	330901.613	191.069	103.169	.16	.03
	211.000	107249.375	351868.027	215.398	116.305	.06	.01
	219.000	113726.125	373117.207	241.893	130.612	.02	.00
	227.000	120333.063	394793.508	270.799	146.220	.01	.00
CUTØFF	227.952	121130.813	397410.801	274.413	148.171	.01	.00

TABLE 4.1.1.4-VI (Continued)

	TIME	ALTITUDE		DØWN RANGE		DYN. PRESSURE	
	SEC.	MT	FT	KM	NAUT. MI.	KG/M.SQ.	LB/F.SQ.
SEPARATIØN	227.952	121130.813	397410.801	274.413	148.171	.01	.00
END CØAST	231.752	124271.875	407716.121	288.907	155.997	.01	.00
	231.752	124271.875	407716.121	288.907	155.997	.01	.00
BØØSTER IMP.	586.807	-.000	-.000	1639.541	885.282	1095546.98	224386.54
START CØV	231.752	124271.875	407716.121	288.915	156.001	.01	.00
	247.000	136165.750	446738.023	347.465	187.616	.00	.00
	263.000	147551.375	484092.434	409.852	221.302	.00	.00
	279.000	157847.625	517872.781	473.256	255.538	.00	.00
	295.000	167087.813	548188.359	537.724	290.348	.00	.00
	311.000	175306.063	575151.117	603.307	325.760	.00	.00
	327.000	182537.188	598875.281	670.053	361.800	.00	.00
	343.000	188817.688	619480.594	738.018	398.498	.00	.00
	359.000	194184.813	637089.273	807.252	435.881	.00	.00
	375.000	198677.125	651827.836	877.811	473.980	.00	.00
	391.000	202335.125	663829.148	949.749	512.824	.00	.00
	407.000	205200.688	673230.594	1023.125	552.443	.00	.00
	423.000	207317.750	680176.344	1097.997	592.871	.00	.00
	439.000	208732.250	684817.086	1174.426	634.140	.00	.00
	455.000	209492.250	687310.523	1252.475	676.283	.00	.00
	468.202	209662.500	687869.086	1318.139	711.738	.00	.00
	471.000	209648.500	687823.156	1332.208	719.335	.00	.00
	487.000	209254.750	686531.328	1413.690	763.332	.00	.00
	503.000	208367.375	683619.992	1496.992	808.311	.00	.00
	519.000	207045.875	679284.359	1582.183	854.310	.00	.00
	535.000	205353.938	673733.383	1669.336	901.369	.00	.00
	551.000	203358.688	667187.289	1758.527	949.529	.00	.00
	567.000	201131.938	659881.680	1849.834	998.831	.00	.00
	583.000	198749.875	652066.516	1943.339	1049.319	.00	.00
	599.000	196294.250	644010.000	2039.125	1101.039	.00	.00
	615.000	193852.000	635997.367	2137.279	1154.038	.00	.00
	631.000	191516.563	628335.172	2237.892	1208.365	.00	.00
	647.000	189388.438	621353.133	2341.059	1264.071	.00	.00
	663.000	187575.375	615404.766	2446.878	1321.208	.00	.00
	679.000	186193.500	610871.055	2555.451	1379.833	.00	.00
	695.000	185368.188	608163.344	2666.887	1440.004	.00	.00

TABLE 4.1.1.4-VI (Continued)

	TIME	ALTITUDE		DOWN RANGE		DYN. PRESSURE	
	SEC.	MT	FT	KM	NAUT. MI.	KG/M.SQ.	LB/F.SQ.
	704.325	185197.438	607603.133	2733.197	1475.808	.00	.00
CUTØFF	704.306	185197.500	607603.344	2733.062	1475.735	.00	.00

TABLE 4.1.1.4-VI (Continued)

	TIME SEC.	MACH	AZ. S DEG.	LATITUDE DEG.	LØNGITUDE DEG.	ALPHA DEG.	CHI DEG.
LIFT ØFF	.000	.000	90.0000	28.6080	.0000	.0000	90.0000
	8.000	.062	89.9914	28.6080	.0003	.0000	90.0000
	16.000	.137	89.9811	28.6080	.0000	.4797	89.4964
	24.000	.227	89.9695	28.6080	-.0001	.5996	87.4116
	32.000	.335	89.9577	28.6080	-.0006	.2000	84.4648
	43.000	.521	89.9448	28.6081	-.0029	.0000	78.7199
	51.000	.692	89.9417	28.6081	-.0068	.0000	73.4838
	59.000	.904	89.9466	28.6081	-.0137	.0000	67.8557
	67.000	1.160	89.9608	28.6082	-.0249	.0000	62.1209
10 KMS	69.911	1.268	89.9682	28.6082	-.0302	.0000	60.0448
	75.000	1.483	89.9837	28.6082	-.0415	.0000	56.4767
Q MAX	79.000	1.675	89.9981	28.6082	-.0522	.0000	53.7583
14 KMS	80.666	1.760	90.0047	28.6082	-.0573	.0000	52.6417
	80.666	1.760	90.0047	28.6082	-.0573	.0000	52.6417
	91.000	2.288	90.0521	28.6082	-.0969	.0000	46.2677
	99.000	2.713	90.0968	28.6081	-.1393	.0000	41.9222
	107.000	3.207	90.1483	28.6079	-.1939	.0000	38.1006
	115.000	3.749	90.2066	28.6076	-.2627	.0000	34.7636
	123.000	4.280	90.2724	28.6072	-.3475	.0000	31.8597
	129.000	4.706	90.3271	28.6067	-.4229	.0000	29.9344
ENG CØ	129.000	4.706	90.3271	28.6067	-.4229	.0000	29.9344
	129.000	4.706	90.3271	28.6067	-.4229	.0000	29.9344
	139.000	4.996	90.4227	28.6057	-.5656	.0000	26.9826
	147.000	5.504	90.5048	28.6045	-.6918	.0000	24.7562
	155.000	6.248	90.5922	28.6031	-.8295	.0862	22.6532
	163.000	7.148	90.6854	28.6013	-.9796	.0941	20.6744
	171.000	8.255	90.7849	28.5990	-1.1431	.1023	18.8188
	179.000	9.409	90.8910	28.5963	-1.3210	.1097	17.0839
	187.000	10.157	91.0044	28.5930	-1.5144	.1164	15.4666
	195.000	10.977	91.1256	28.5890	-1.7246	.1227	13.9630
	203.000	11.881	91.2554	28.5842	-1.9532	.1286	12.5688
	211.000	12.883	91.3946	28.5784	-2.2018	.1334	11.2799
	219.000	14.000	91.5441	28.5715	-2.4725	.1383	10.0920
	227.000	15.258	91.7051	28.5633	-2.7677	.1423	9.0011
CUTØFF	227.952	15.419	91.7251	28.5622	-2.8047	.1427	8.8775

TABLE 4.1.1.4-VI (Continued)

	TIME SEC.	MACH	AZ. S DEG.	LATITUDE DEG.	LØNGITUDE DEG.	ALPHA DEG.	CHI DEG.
SEPARATIØN	227.952	15.419	91.7251	28.5622	-2.8047	.1427	8.8775
END CØAST	231.752	15.390	91.8037	28.5578	-2.9527	.1507	8.3865
	231.752	15.390	91.8037	28.5578	-2.9527	.1507	8.3865
BØØSTER IMP.	566.807	12.284	98.9461	27.2852	-16.6066	.0000	-37.8557
START CØV	231.752	15.390	91.8037	28.5578	-2.9527	11.8621	20.2486
	247.000	15.585	92.1236	28.5377	-3.5506	12.5524	19.2237
	263.000	15.806	92.4633	28.5129	-4.1873	13.2329	18.1479
	279.000	16.045	92.8073	28.4838	-4.8341	13.8672	17.0714
	295.000	16.300	93.1558	28.4505	-5.4913	14.4542	15.9939
	311.000	16.573	93.5088	28.4127	-6.1594	14.9931	14.9148
	327.000	16.861	93.8667	28.3701	-6.8387	15.4833	13.8339
	343.000	17.166	94.2294	28.3227	-7.5299	15.9247	12.7507
	359.000	17.486	94.5972	28.2701	-8.2332	16.3173	11.6647
	375.000	17.822	94.9702	28.2121	-8.9491	16.6611	10.5756
	391.000	18.174	95.3485	28.1484	-9.6782	16.9565	9.4830
	407.000	18.541	95.7322	28.0787	-10.4208	17.2041	8.3865
	423.000	18.924	96.1214	28.0029	-11.1775	17.4043	7.2857
	439.000	19.323	96.5164	27.9205	-11.9487	17.5578	6.1801
	455.000	19.736	96.9170	27.8312	-12.7349	17.6654	5.0694
	468.202	20.090	97.2521	27.7521	-13.3952	17.7201	4.1487
	471.000	20.166	97.3556	27.7347	-13.5366	17.7278	3.9531
	487.000	20.612	97.7361	27.6306	-14.3543	17.7457	2.8309
	503.000	21.074	98.1546	27.5185	-15.1884	17.7200	1.7024
	519.000	21.553	98.5792	27.3981	-16.0396	17.6514	.5671
	535.000	22.049	99.0100	27.2688	-16.9084	17.5408	-.5752
	551.000	22.562	99.4469	27.1302	-17.7952	17.3888	-1.7251
	567.000	23.094	99.8900	26.9819	-18.7005	17.1963	-2.8827
	583.000	23.646	100.3394	26.8234	-19.6251	16.9639	-4.0486
	599.000	24.217	100.7950	26.6540	-20.5693	16.6923	-5.2230
	615.000	24.810	101.2567	26.4733	-21.5337	16.3822	-6.4063
	631.000	25.425	101.7245	26.2806	-22.5190	16.0341	-7.5987
	647.000	26.065	102.1984	26.0754	-23.5256	15.6487	-8.8007
	663.000	26.730	102.6783	25.8569	-24.5543	15.2264	-10.0125
	679.000	27.423	103.1640	25.6246	-25.6056	14.7678	-11.2344
	695.000	28.146	103.6553	25.3776	-26.6801	14.2733	-12.4665

TABLE 4.1.1.4-VI (Continued)

	TIME SEC.	MACH	AZ. S DEG.	LATITUDE DEG.	LØNGITUDE DEG.	ALPHA DEG.	CHI DEG.
	704.325	28.582	103.9440	25.2266	-27.3173	13.9687	-13.1895
CUTØFF	704.306	28.581	103.9436	25.2269	-27.3160	13.9694	-13.1880

TABLE 4.1.1.4-VI (Continued)

	TIME	DXX		DYY		DZZ	
	SEC.	MT/SEC.	FT/SEC.	MT/SEC.	FT/SEC.	MT/SEC.	FT/SEC.
LIFT ØFF	.000	408.641	1340.687	.000	.000	-.000	-.000
	8.000	408.621	1340.621	21.321	69.951	-.114	-.375
	16.000	408.684	1340.826	46.820	153.607	-.228	-.749
	24.000	411.396	1349.725	77.145	253.101	-.343	-1.124
	32.000	419.343	1375.798	112.658	369.614	-.457	-1.498
	43.000	443.000	1453.411	170.107	558.093	-.613	-2.012
	51.000	473.825	1554.545	217.440	713.387	-.727	-2.386
	59.000	518.723	1701.846	267.825	878.691	-.841	-2.759
	67.000	577.763	1895.549	316.703	1039.050	-.955	-3.132
10 KMS	69.911	603.007	1978.371	334.139	1096.256	-.996	-3.267
	75.000	652.984	2142.336	365.480	1199.083	-1.068	-3.504
G MAX	79.000	697.969	2289.923	391.162	1283.339	-1.125	-3.689
14 KMS	80.686	718.492	2357.257	402.243	1319.695	-1.148	-3.768
	80.686	718.492	2357.257	402.243	1319.695	-1.148	-3.768
	91.000	865.621	2839.965	473.438	1553.273	-1.294	-4.245
	99.000	1006.015	3300.576	531.650	1744.259	-1.407	-4.615
	107.000	1169.779	3837.857	591.442	1940.427	-1.519	-4.984
	115.000	1357.716	4454.448	652.639	2141.203	-1.631	-5.351
	123.000	1571.419	5155.576	715.721	2348.166	-1.743	-5.718
	129.000	1750.111	5741.833	764.848	2509.345	-1.826	-5.992
END CØ	129.000	1750.111	5741.833	764.848	2509.345	-1.826	-5.992
	129.000	1750.111	5741.833	764.848	2509.345	-1.826	-5.992
	139.000	1912.435	6274.394	756.805	2482.957	-1.965	-6.447
	147.000	2053.561	6737.406	748.651	2456.204	-2.076	-6.810
	155.000	2205.355	7235.418	738.882	2424.153	-2.188	-7.178
	163.000	2368.629	7771.093	727.486	2386.765	-2.299	-7.542
	171.000	2544.480	8348.033	714.522	2344.233	-2.409	-7.903
	179.000	2734.268	8970.695	700.078	2296.844	-2.518	-8.262
	187.000	2939.700	9644.684	684.277	2245.002	-2.627	-8.618
	195.000	3162.940	10377.098	667.277	2189.228	-2.735	-8.972
	203.000	3406.760	11177.034	649.281	2130.186	-2.842	-9.325
	211.000	3674.161	12056.303	630.543	2068.710	-2.949	-9.676
	219.000	3971.707	13030.536	611.387	2005.863	-3.056	-10.025
	227.000	4304.058	14120.925	592.237	1943.033	-3.162	-10.373
CLTØFF	227.952	4346.363	14259.720	589.982	1935.638	-3.174	-10.414

TABLE 4.1.1.4-VI (Continued)

	TIME	DXX		DYY		DZZ	
	SEC.	MT/SEC.	FT/SEC.	MT/SEC.	FT/SEC.	MT/SEC.	FT/SEC.
SEPARATIØN	227.952	4346.363	14259.720	589.982	1935.638	-3.174	-10.414
END CØAST	231.752	4344.249	14252.785	554.132	1818.017	-3.225	-10.579
	231.752	4344.249	14252.785	554.132	1818.017	-3.225	-10.579
BØØSTER IMP.	566.807	3754.883	12319.169	-2733.131	-8966.964	-7.534	-24.719
START CØV	231.752	4344.249	14252.785	554.132	1818.017	-3.225	-10.579
	247.000	4410.168	14469.057	437.848	1436.508	-3.216	-10.550
	263.000	4479.901	14697.839	315.561	1035.304	-3.205	-10.516
	279.000	4550.255	14928.659	192.917	632.930	-3.194	-10.478
	295.000	4621.278	15161.672	69.832	229.108	-3.181	-10.436
	311.000	4693.018	15397.042	-53.780	-176.444	-3.167	-10.391
	327.000	4765.530	15634.941	-178.006	-584.008	-3.152	-10.342
	343.000	4838.868	15875.550	-302.932	-993.871	-3.136	-10.289
	359.000	4913.090	16119.061	-428.647	-1406.323	-3.119	-10.233
	375.000	4988.259	16365.679	-555.243	-1821.662	-3.101	-10.174
	391.000	5064.442	16615.622	-682.810	-2240.191	-3.082	-10.111
	407.000	5141.710	16869.127	-811.446	-2662.223	-3.061	-10.044
	423.000	5220.141	17126.448	-941.247	-3088.082	-3.040	-9.974
	439.000	5299.821	17387.863	-1072.317	-3518.101	-3.018	-9.900
	455.000	5380.839	17653.671	-1204.762	-3952.631	-2.994	-9.823
	468.202	5448.768	17876.536	-1315.158	-4314.823	-2.974	-9.757
	471.000	5463.298	17924.205	-1338.693	-4392.037	-2.970	-9.743
	487.000	5547.307	18199.825	-1474.227	-4836.703	-2.944	-9.658
	503.000	5632.988	18480.931	-1611.489	-5287.036	-2.917	-9.571
	519.000	5720.476	18767.964	-1750.609	-5743.469	-2.889	-9.480
	535.000	5809.919	19061.415	-1891.731	-6206.466	-2.861	-9.385
	551.000	5901.486	19361.829	-2035.005	-6676.525	-2.831	-9.287
	567.000	5995.360	19669.816	-2180.595	-7154.183	-2.800	-9.186
	583.000	6091.751	19986.059	-2328.681	-7640.030	-2.768	-9.081
	599.000	6190.892	20311.326	-2479.459	-8134.707	-2.735	-8.973
	615.000	6293.049	20646.485	-2633.145	-8638.926	-2.701	-8.861
	631.000	6398.519	20992.516	-2789.979	-9153.473	-2.666	-8.746
	647.000	6507.644	21350.537	-2950.230	-9679.231	-2.630	-8.628
	663.000	6620.811	21721.822	-3114.200	-10217.192	-2.593	-8.507
	679.000	6738.466	22107.828	-3282.234	-10768.483	-2.555	-8.383
	695.000	6861.120	22510.238	-3454.722	-11334.390	-2.516	-8.255

TABLE 4.1.1.4-VI (Continued)

	TIME	DXX		DYY		DZZ	
	SEC.	MT/SEC.	FT/SEC.	MT/SEC.	FT/SEC.	MT/SEC.	FT/SEC.
	704.325	6935.141	22753.087	-3557.480	-11671.523	-2.493	-8.179
CUTØFF	704.306	6934.990	22752.592	-3557.272	-11670.839	-2.493	-8.179

TABLE 4.1.1.4-VI (Continued)

	TIME SEC.	SØLID THRUST NEWTØNS	SØLID THRUST LBS.	SØLID WDØT KG/SEC	SØLID WDØT LB/SEC	LIQUID WDØT KG/SEC	LIQUID WDØT LB/SEC
LIFT ØFF	.000	-.0	-.0	-.00	-.00	10476.78	23097.35
	8.000	-.0	-.0	-.00	-.00	10476.78	23097.35
	16.000	-.0	-.0	-.00	-.00	10476.78	23097.35
	24.000	-.0	-.0	-.00	-.00	10476.78	23097.35
	32.000	-.0	-.0	-.00	-.00	10476.78	23097.35
	43.000	-.0	-.0	-.00	-.00	10476.78	23097.35
	51.000	-.0	-.0	-.00	-.00	10476.78	23097.35
	59.000	-.0	-.0	-.00	-.00	10476.78	23097.35
	67.000	-.0	-.0	-.00	-.00	10476.78	23097.35
10 KMS	69.911	-.0	-.0	-.00	-.00	10476.78	23097.35
	75.000	-.0	-.0	-.00	-.00	10476.78	23097.35
Q MAX	79.000	-.0	-.0	-.00	-.00	10476.78	23097.35
14 KMS	80.686	-.0	-.0	-.00	-.00	10476.78	23097.35
	80.686	-.0	-.0	-.00	-.00	10476.78	23097.35
	91.000	-.0	-.0	-.00	-.00	10476.78	23097.35
	99.000	-.0	-.0	-.00	-.00	10476.78	23097.35
	107.000	-.0	-.0	-.00	-.00	10476.78	23097.35
	115.000	-.0	-.0	-.00	-.00	10476.78	23097.35
	123.000	-.0	-.0	-.00	-.00	10476.78	23097.35
	129.000	-.0	-.0	-.00	-.00	10476.78	23097.35
ENG CØ	129.000	-.0	-.0	-.00	-.00	5238.39	11548.68
	129.000	-.0	-.0	-.00	-.00	5238.39	11548.68
	139.000	-.0	-.0	-.00	-.00	5238.39	11548.68
	147.000	-.0	-.0	-.00	-.00	5238.39	11548.68
	155.000	-.0	-.0	-.00	-.00	5238.39	11548.68
	163.000	-.0	-.0	-.00	-.00	5238.39	11548.68
	171.000	-.0	-.0	-.00	-.00	5238.39	11548.68
	179.000	-.0	-.0	-.00	-.00	5238.39	11548.68
	187.000	-.0	-.0	-.00	-.00	5238.39	11548.68
	195.000	-.0	-.0	-.00	-.00	5238.39	11548.68
	203.000	-.0	-.0	-.00	-.00	5238.39	11548.68
	211.000	-.0	-.0	-.00	-.00	5238.39	11548.68
	219.000	-.0	-.0	-.00	-.00	5238.39	11548.68
	227.000	-.0	-.0	-.00	-.00	5238.39	11548.68
CUTØFF	227.952	-.0	-.0	-.00	-.00	5238.39	11548.68

TABLE 4.1.1.4-VI (Continued)

	TIME SEC.	SØLID THRUST NEWTØNS	SØLID THRUST LBS.	SØLID WDØT KG/SEC	SØLID WDØT LB/SEC	LIQUID WDØT KG/SEC	LIQUID WDØT LB/SEC
SEPARATIØN	227.552	-.0	-.0	-.00	-.00	-.00	-.00
END CØAST	231.752	-.0	-.0	-.00	-.00	-.00	-.00
	231.752	-.0	-.0	-.00	-.00	-.00	-.00
BØØSTER IMP.	586.807	-.0	-.0	-.00	-.00	-.00	-.00
START CØV	231.752	-.0	-.0	-.00	-.00	218.28	481.22
	247.000	-.0	-.0	-.00	-.00	218.28	481.22
	263.000	-.0	-.0	-.00	-.00	218.28	481.22
	279.000	-.0	-.0	-.00	-.00	218.28	481.22
	295.000	-.0	-.0	-.00	-.00	218.28	481.22
	311.000	-.0	-.0	-.00	-.00	218.28	481.22
	327.000	-.0	-.0	-.00	-.00	218.28	481.22
	343.000	-.0	-.0	-.00	-.00	218.28	481.22
	359.000	-.0	-.0	-.00	-.00	218.28	481.22
	375.000	-.0	-.0	-.00	-.00	218.28	481.22
	391.000	-.0	-.0	-.00	-.00	218.28	481.22
	407.000	-.0	-.0	-.00	-.00	218.28	481.22
	423.000	-.0	-.0	-.00	-.00	218.28	481.22
	439.000	-.0	-.0	-.00	-.00	218.28	481.22
	455.000	-.0	-.0	-.00	-.00	218.28	481.22
	468.202	-.0	-.0	-.00	-.00	218.28	481.22
	471.000	-.0	-.0	-.00	-.00	218.28	481.22
	487.000	-.0	-.0	-.00	-.00	218.28	481.22
	503.000	-.0	-.0	-.00	-.00	218.28	481.22
	519.000	-.0	-.0	-.00	-.00	218.28	481.22
	535.000	-.0	-.0	-.00	-.00	218.28	481.22
	551.000	-.0	-.0	-.00	-.00	218.28	481.22
	567.000	-.0	-.0	-.00	-.00	218.28	481.22
	583.000	-.0	-.0	-.00	-.00	218.28	481.22
	599.000	-.0	-.0	-.00	-.00	218.28	481.22
	615.000	-.0	-.0	-.00	-.00	218.28	481.22
	631.000	-.0	-.0	-.00	-.00	218.28	481.22
	647.000	-.0	-.0	-.00	-.00	218.28	481.22
	663.000	-.0	-.0	-.00	-.00	218.28	481.22
	679.000	-.0	-.0	-.00	-.00	218.28	481.22
	695.000	-.0	-.0	-.00	-.00	218.28	481.22

TABLE 4.1.1.4-VI (Continued)

	TIME	SØLID THRUST		SØLID WDØT		LIQUID WDØT	
	SEC.	NEWTØNS	LBS.	KG/SEC	LB/SEC	KG/SEC	LB/SEC
	704.325	-.0	-.0	-.00	-.00	218.28	481.22
CUTØFF	704.306	-.0	-.0	-.00	-.00	218.28	481.22

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4.1.2 Aerodynamics

The static aerodynamic characteristics of the baseline INT-20 vehicle with 4 F-1 engines and a 43 foot (13.1 m) payload shroud are presented in this section. The baseline vehicle is pictured in Figure 4.1.2-1. The on-pad and lift-off vehicle aerodynamics (normal force coefficient and center of pressure location) are presented in Figure 4.1.2-2 as a function of angle of attack. The normal force coefficient (C_N) was calculated using the modification of Allen's equation (excluding fin and shroud effects) proposed by Kelly (reference 4.1.2-1):

$$C_N = C_{N_\alpha} \, \text{Sin}\,\alpha \, \cos\alpha + C_{N_{90^o}} \, \text{Sin}^3\alpha = C_{N_{fins+shrouds}}$$

Where: α is the angle of attack;

C_{N_α} is the normal force coefficient gradient at $\alpha = 0^o$ for the clean forebody ($C_{N_\alpha} = 2.61$);

$C_{N_{90^o}}$ is the normal force coefficient at $\alpha = 90^o$ for the clean forebody ($C_{N_{90^o}} = 4.98$);

$C_{N_{fins+shrouds}}$ is the normal force coefficient contribution of fins and shrouds together at the angle of attack.

The normal force coefficient at $\alpha = 90^o$ derives from the cross-flow drag coefficient for a two-dimensional cylinder. The other coefficients, C_{N_α} and $C_{N_{fins+shrouds}}$, are based on data from references 4.1.2-2, 4.1.2-3, and 4.1.2-4. The center of pressure was calculated using a segmented breakdown of the forebody normal force distributions:

$$CP/D = \sum_{i=1}^{n} C_{N_i} \, (CP/D)_i \Big/ \sum_{i=1}^{n} C_{N_i}$$

Where:

(n) is the number of segments into which the forebody is divided.

The total vehicle axial force coefficient (as a function of Mach number) is given in Figure 4.1.2-3. It is based on the predicted INT-20 trajectory. The axial force coefficient was obtained by summing the base axial force coefficient with power-on (Figure 4.1.2-4), and the total forebody contribution including fins and shrouds (Figure 4.1.2-5). Figure 4.1.2-6 presents the contribution of the fins and shrouds alone. The base axial force contribution (Figure 4.1.2-4) was also calculated for the predicted INT-20 trajectory. It was determined by comparing power-off and power-on base pressure coefficients and an analysis of plume

4.1.2 (Continued)

characteristics and flow interaction. It was assumed that the difference between power-off and power-on base pressure coefficients was dependent upon the number of lines of interaction between adjacent engines. Removal of the center engine decreases the base pressure coefficient to a level between the power-off and power-on values. Therefore, a new effective base area was determined for the INT-20 vehicle and base drag calculated. At a Mach number of 6.51, two of the four outboard engines are cut off to limit the g level to 4.68 g's. This results in a further lowering of the base pressure and a resultant jump in a base axial force contribution at M = 6.51. In the base flow analysis, use was made of the AS-502 flight test base pressure data. The AS-502 vehicle used the S-IC stage without air scoops on the fairings. Figures 4.1.2-7 through 4.1.2-10 present the distributions of local axial force coefficients at Mach numbers of 0.9, 1.0, 1.29, and 1.69 for zero angle of attack. These data are also applicable for angles of attack of up to about 15 degrees.

The total vehicle normal force coefficient gradient (C_{N_α}) and center of pressure location (CP/D) are presented in Figure 4.1.2-11 as a function of Mach number. Figure 4.1.2-12 presents the fins and shrouds C_{N_α} contribution. The contribution of the INT-20 clean forebody (no fins or shrouds) is shown in Figure 4.1.2-13 and represents the difference of Figures 4.1.2-11 and 4.1.2-12. The distributed normal force coefficient gradients are presented for Mach numbers of 0.9, 1.0, 1.29, and 1.69 in Figures 4.1.2-14 through 4.1.2-17, respectively. The maximum dynamic pressure (max q) condition occurs at a Mach number of 1.69, while the maximum ($q\alpha$) condition occurs at a Mach number of 1.29.

Table 4.1.2-I gives the modified Newtonian aerodynamic characteristics (C_{N_α}, CP/D, and C_A) for the S-IC stage by itself, and the S-IVB stage plus payload by itself. These are applicable in the hypersonic regime for these stages after separation ($M_\infty \geq 15.8$ and $h \geq 352,000$ feet (107290 m), since the dynamic pressure is so low at separation conditions, $q \leq 0.02$ lb/ft^2 (.957 N/M^2), that the external aerodynamic forces on these stages will be extremely small.

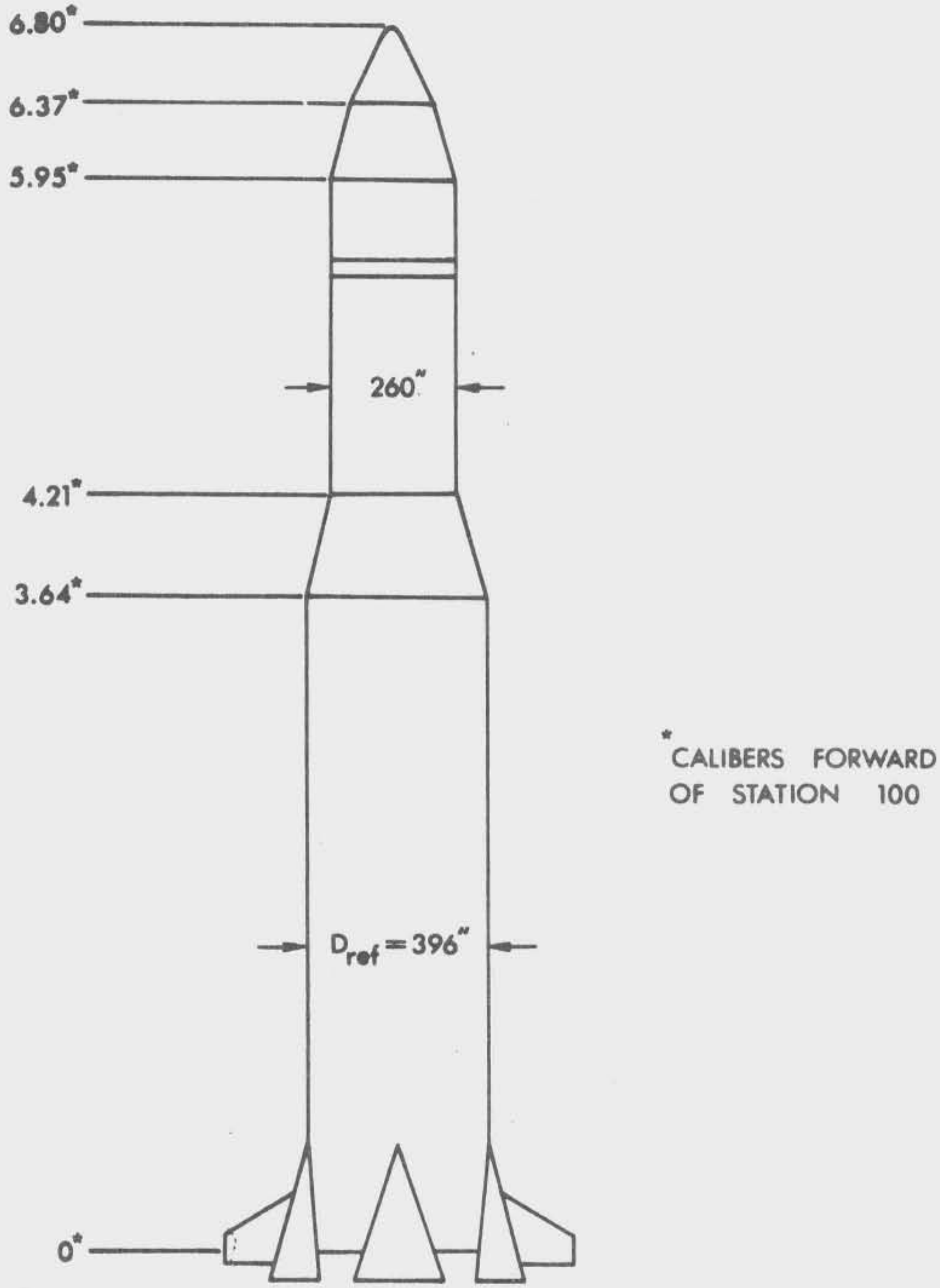


FIGURE 4.1.2-1 INT-20 VEHICLE

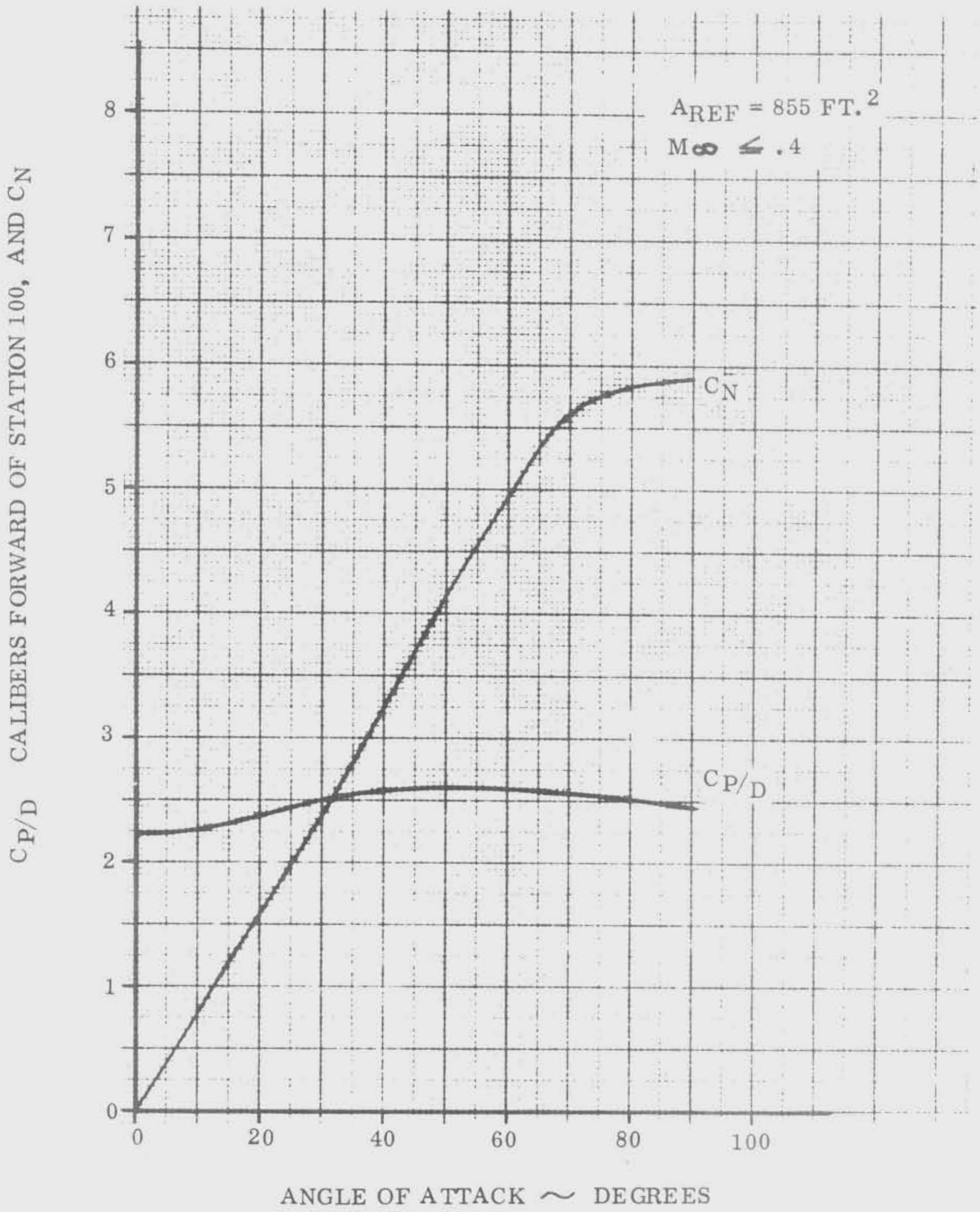


FIGURE 4.1.2-2 ON PAD AND LIFT OFF AERODYNAMICS, 43 FT. PAYLOAD

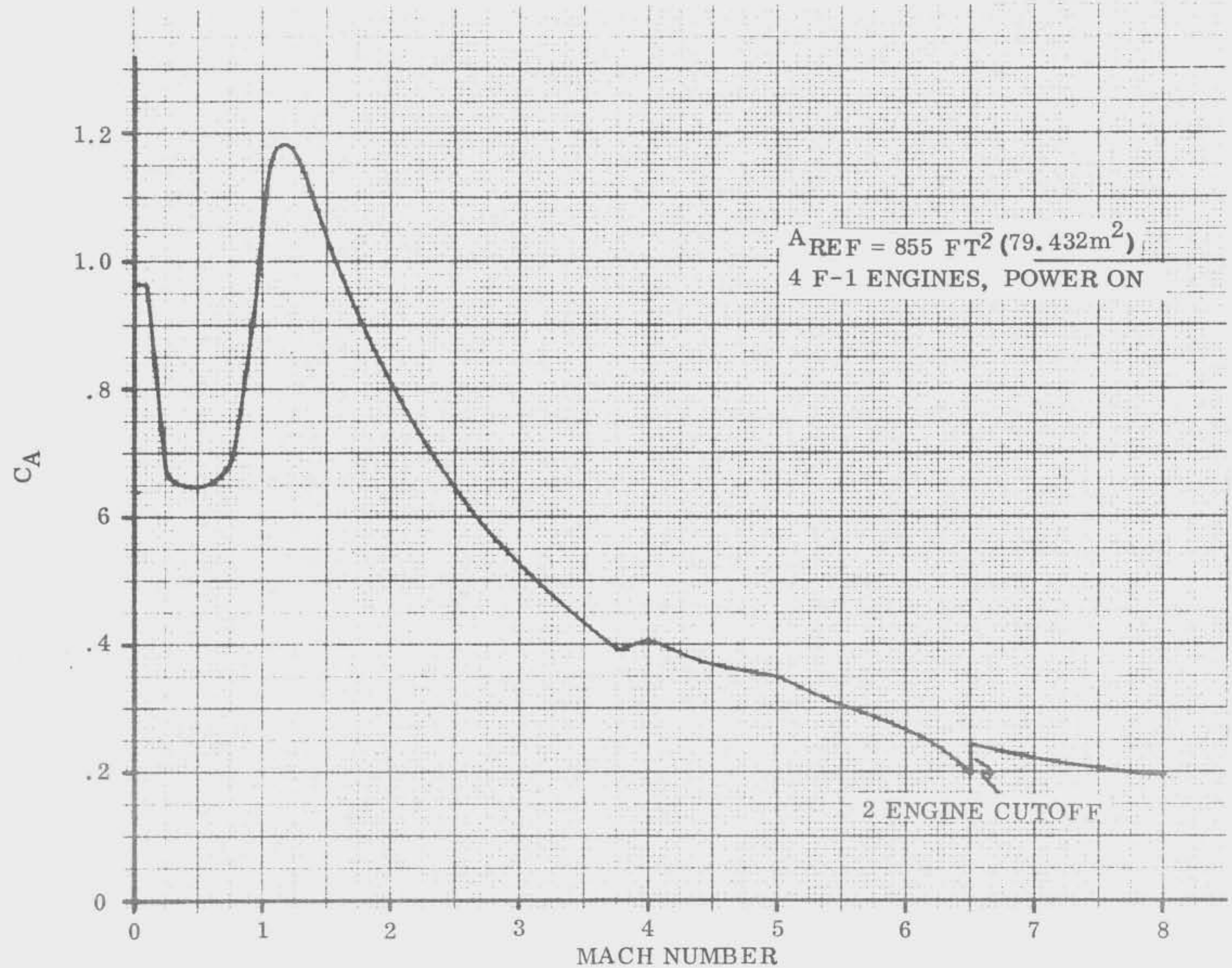


FIGURE 4.1.2-3 VEHICLE AXIAL FORCE COEFFICIENT VS. MACH NUMBER AT ZERO ANGLE OF ATTACK, 43 FT. (13.1m) PAYLOAD

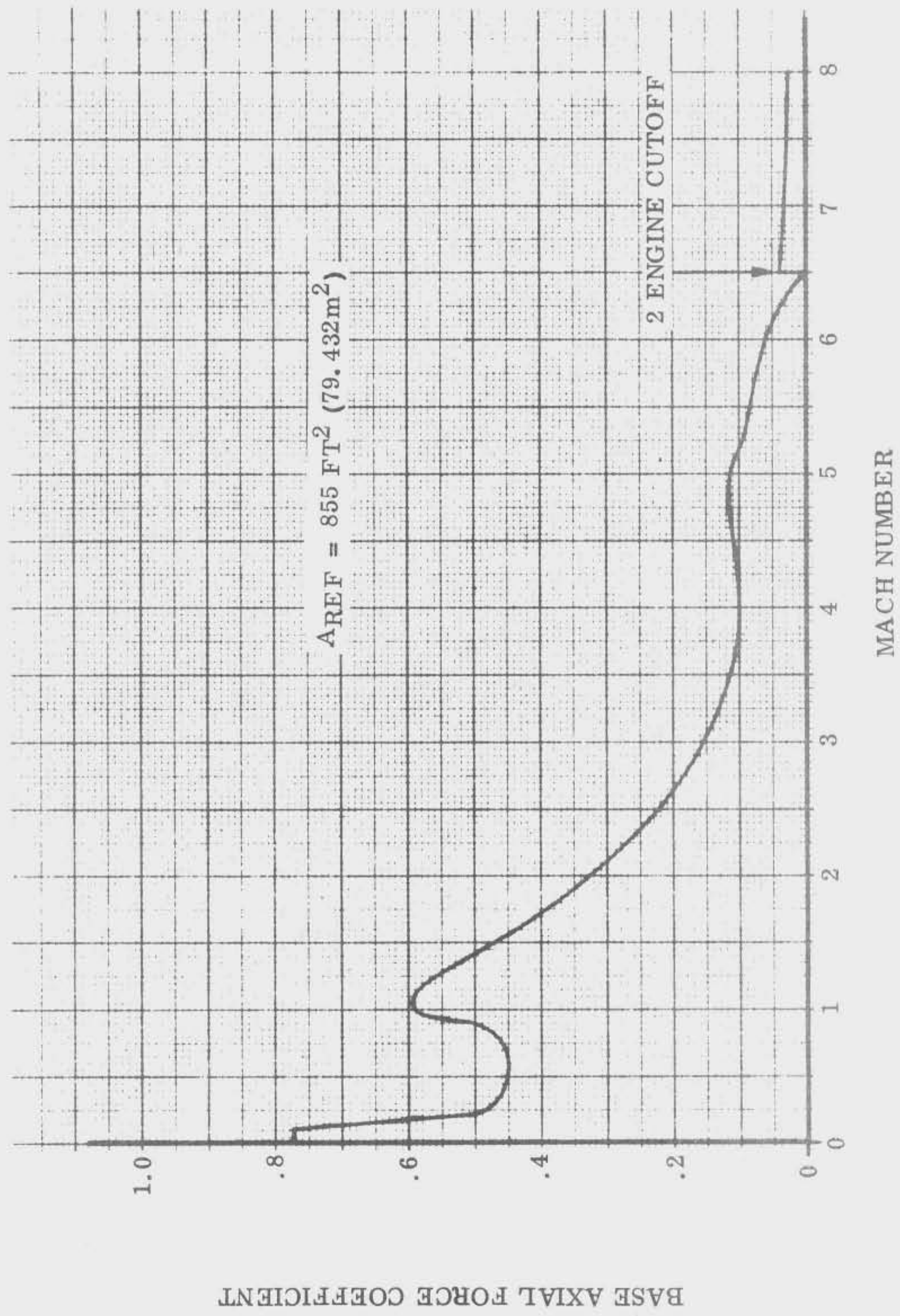


FIGURE 4.1.2-4 BASE AXIAL FORCE COEFFICIENT VS. MACH NUMBER (POWER ON)

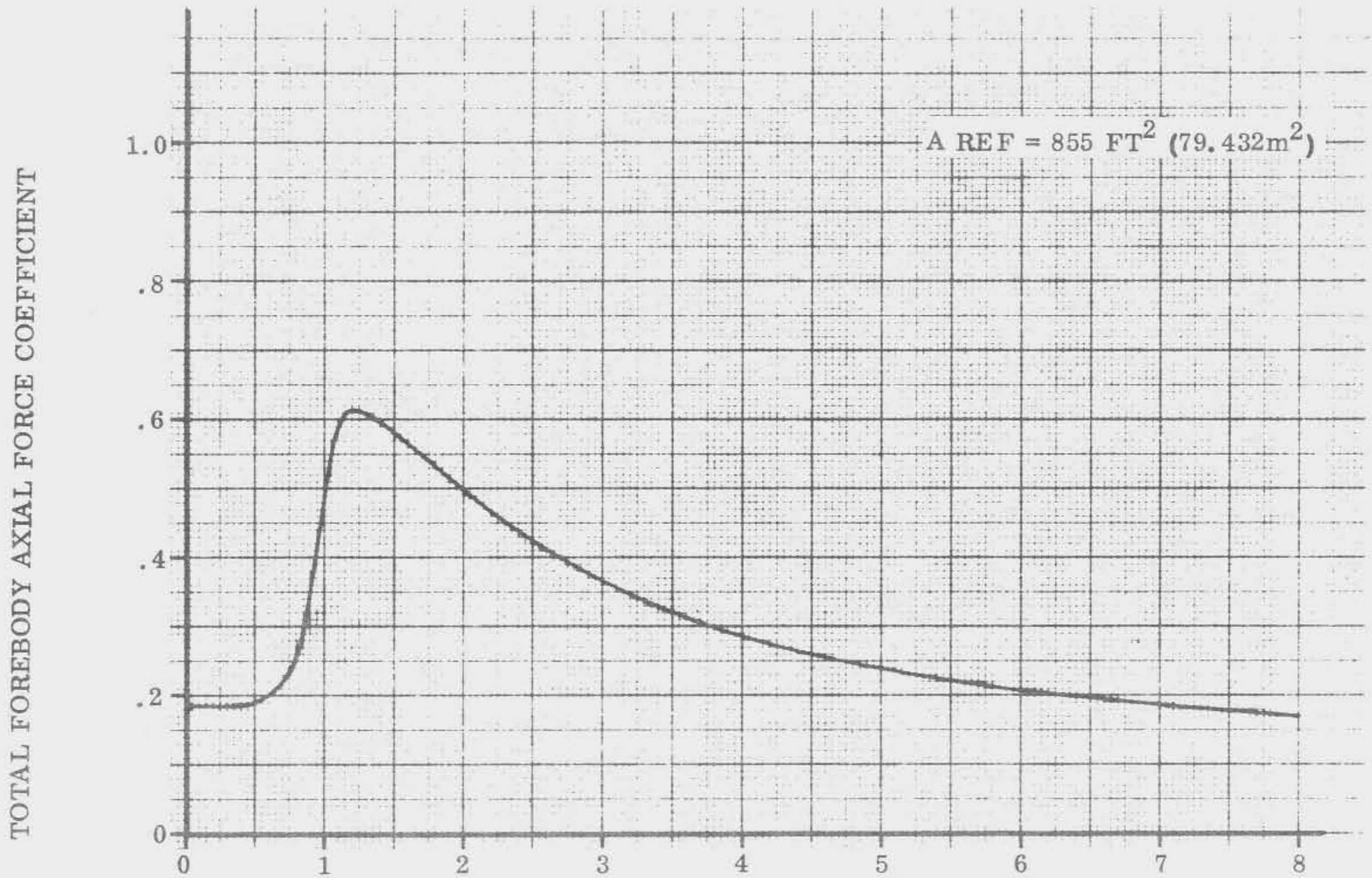


FIGURE 4.1.2-5 AXIAL FORCE COEFFICIENT OF TOTAL FOREBODY VS. MACH NUMBER (INCLUDES FINS PLUS SHROUDS)

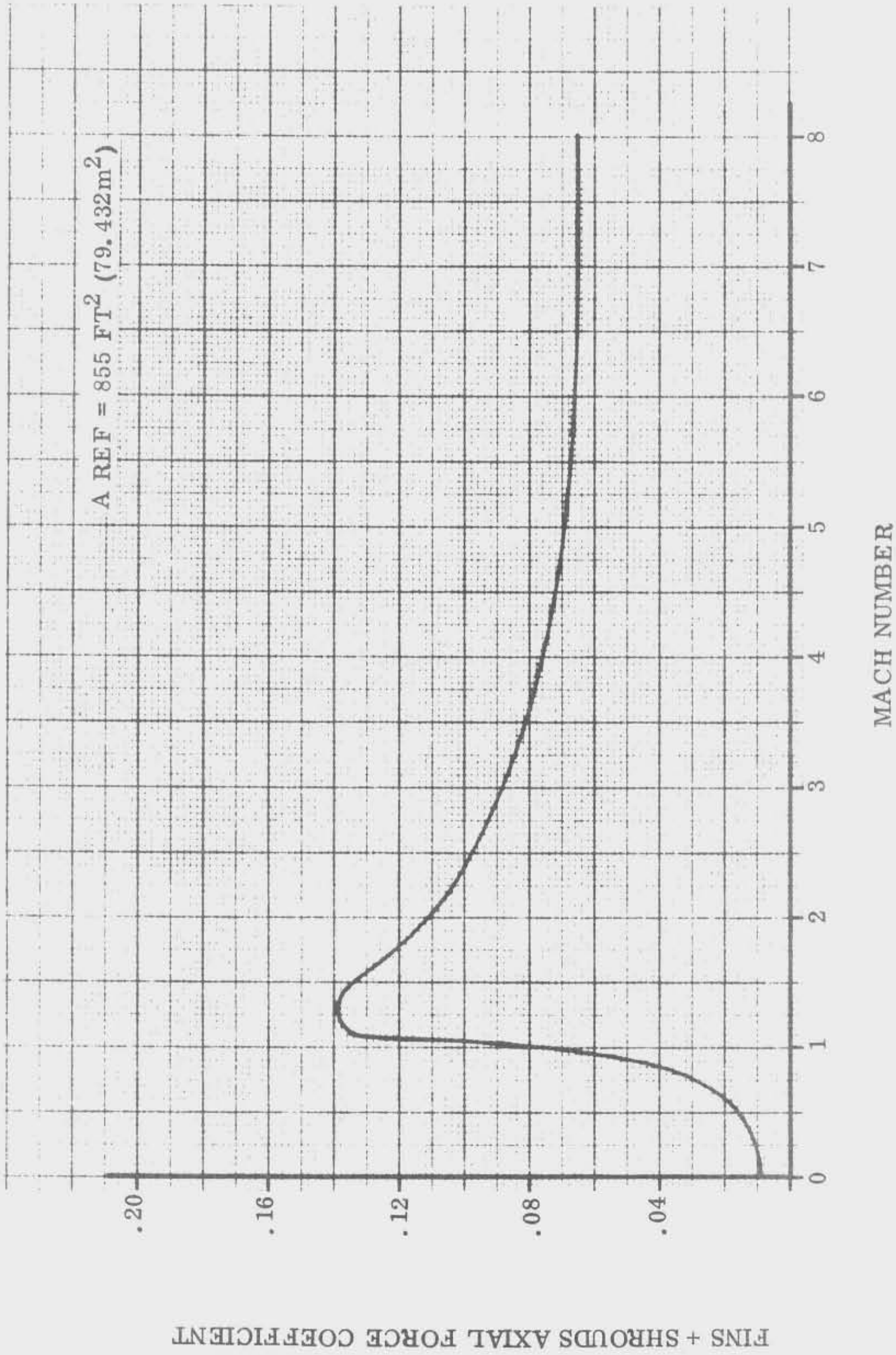


FIGURE 4.1.2-6 AXIAL FORCE COEFFICIENT OF FINS PLUS SHROUDS VS. MACH NUMBER

TABLE 4.1.2-I MODIFIED NEWTONIAN AERODYNAMICS OF INT-20 STAGES AFTER SEPARATION

STAGE	C_N $(\frac{1}{RAD.})$	CP/D (CALIBERS)	C_A
S-IC	0.88	0.13*	1.84
S-IVB (PLUS 43 FT)	1.65	1.20**	0.19

NOTE : ALL COEFFICIENTS ARE BASED ON 855 FT^2 REFERENCE AREA AND 33 FT. REFERENCE DIAMETER

* CALIBERS FORWARD OF VEHICLE STATION 100

** CALIBERS FORWARD OF VEHICLE STATION 1541

4.1.3 Vehicle Environment

The acoustic and thermal environments expected on the INT-20 vehicle were extrapolated from Saturn V flight data. In general, the severity of environment for the INT-20 is equal to or less than that of the Saturn V.

4.1.3.1 Acoustics and Vibration

The acoustics environment for the baseline 4 F-1 INT-20 launch vehicle having an MLV payload shape is shown in Figure 4.1.3.1-1. These data represent the maximum acoustic level envelope encountered throughout flight. In all cases the sound pressure levels are below the present design specification. The near-field launch (lift-off) environment is less than that for the Saturn V. The in-flight acoustic environment was determined by extrapolating data from the Saturn IB AS-203, AS-204 and Saturn V AS-501, AS-502 flights. The flight data provided information on configurations similar to the MLV nose and the S-IVB-20 to S-IC-20 transition respectively. Basic aerodynamic phenomena such as turbulent attached boundary layer, shock-induced boundary layer separation, shoulder expansion-induced boundary layer separation, and oscillating shock waves were considered. Differences in trajectory characteristics are accounted for in the data extrapolation.

Component vibration is basically proportional to the acoustic excitation. The maximum acoustic environment of the INT-20 launch vehicle (shown in Figure 4.1.3-1) is equal to or less than the acoustic environment on corresponding components of the Saturn V - Apollo vehicle. Therefore, it is assumed that the vibration environments on the S-IC stage of the INT-20 are within the qualification levels of corresponding Saturn V S-IC stage. However, Saturn flight data indicate that low frequency vibration levels (below 100 hertz) at several locations on the S-IVB stage exceeded qualification test specifications. Also since the S-IVB stage is physically closer to the acoustic excitation source, predicted vibration levels are estimated to be 25 percent higher for the INT-20 than for the Saturn V. Consequently, some critical components on the S-IVB stage will require requalification (See Section 4.2.4). Resulting additional test requirements are discussed in Section 5.2.3.

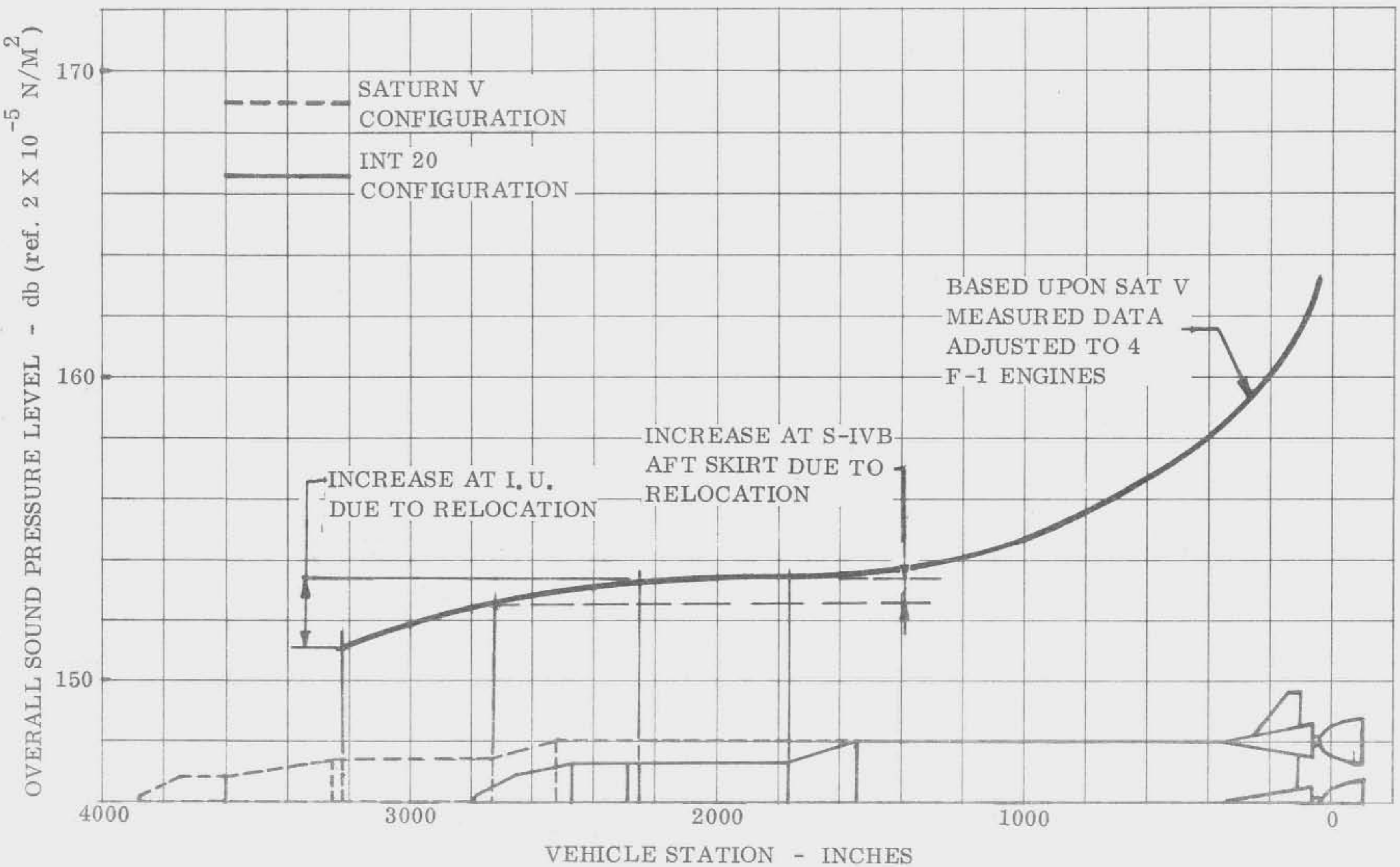


FIGURE 4.1.3.1-1 ACOUSTIC ENVIRONMENT - SAT-INT 20 WITH 4 F-1 ENGINES AND MLV NOSE

4.1.3.2 Thermal

The INT-20 vehicle thermal environment was analyzed to determine heating effects upon S-IC and S-IVB stage design. Base heating and aerodynamic heating were investigated. The data resulting from analyses included the base region radiation heating rate, gas recovery temperature, and heat transfer coefficient. Convective film coefficients and recovery temperatures were also calculated and are contained in Appendix D.5, Figures D.5-1 through D.5-12.

a. S-IC Stage

1. Base Heat Shield

The maximum temperature of the forward surface of the INT-20 heat shield panels has increased to 527°F because of different form factors, longer soak time, and higher k factor. The higher k factor results from the use of FTA-442A heat shield material. This material replaced M-31 (effective with SA-510) because the fiberous potassium titanate matrix material used to make M-31 is no longer available. A "pigmentary" potassium titanate is used in the new FTA-422A heat shield. This requires the use of a different production process and a change in material composition. FTA-442A is denser and stronger, and has a slightly higher conductivity (k).

The base heat shield thermal environment is described in Appendix A-2, Paragraph 2.1.1.11. Base region radiant heat flux, gas recovery temperatures, and convective heating coefficient are shown in Figures A-36 through A-38 of the appendix, document no. D5-17009-3. A diagram of the heat shield panel cross section is shown in Figure A-41.

2. Forward Skirt

INT-20 thermal data and comparative INT-20 and Saturn V thermal responses of the forward skirt skin and hat sections are contained in Appendix A-2, D5-17009-3, Figures A-30 through A-35.

b. S-IVB

An analysis of the thermal environment for the S-IVB stage is contained in Section 4.2.4.8. This analysis was based upon film coefficients and recovery temperatures shown in document D5-17009-3, Appendix D.5, Figures D.5-1 through D.5-12 Insulation (Korotherm TC-320) is required to cover certain protuberance heating areas on the S-IVB forward skirt and over the entire surface (.01 inches thick) for the aft interstage, in order to meet the structural requirements shown in Paragraph 4.1.6.3.

4.1.4 Controls

The INT-20 vehicle control parameters were defined using rigid and flexible body analyses. Study was also made of liftoff dynamics and S-IC/S-IVB separation clearance and control. INT-20 controllability compared satisfactorily with that of the Saturn V except in one area. The rigid body analysis indicated that engine-out control capability at the trajectory time point of maximum dynamic pressure is less than that of the Saturn V, and is marginal.

4.1.4.1 Rigid Body

The rigid body control analysis for the INT-20 baseline configuration was made using the environmental conditions and vehicle characteristics existing in the region of maximum dynamic pressure (q max) during first stage flight. The results show the rigid body control gains (Figure 4.1.4.1-1), the time to double amplitude and control authority ratio (Figure 4.1.4.1-2) and the angle of attack and thrust deflection (Figure 4.1.4.1-3) resulting from a simulation of the flight from liftoff to first stage burnout. Root-sum-square (RSS) envelopes were obtained for basic vehicle dynamic parameters (thrust deflection angle, angle of attack, and lateral accelerations) for the time of flight where dynamic pressure is maximum (Figure 4.1.4.1-4 and 4.1.4.1-5). Also, an engine-out study was made to determine the controllability of the INT-20 when subjected to a potential abort situation at q max. Saturn V/Apollo design wind and linear aerodynamics were used.

a. Control Equation

The basic attitude-attitude rate control equation used for these analyses is:

$$\beta_c = A_0 \phi_\epsilon + A_1 \dot{\phi}$$

where:

β_c = command thrust deflection

ϕ_ϵ = attitude error

$\dot{\phi}$ = attitude rate

A_0 = attitude error gain

A_1 = attitude rate gain

b. Flight Control System Gains

The flight control system gains were determined based on a rigid body model. The equations (see Reference 4.1.1.1-1) are:

4.1.4.1 (Continued)

$$A_0 = \frac{\omega_n^2 - C_1}{C_2} \quad ; \qquad A_1 = \frac{2\zeta_n \omega_n}{C_2}$$

where ω_n = undamped natural frequency

ζ_n = damping ratio

C_1 = aerodynamic disturbing moment coefficient

C_2 = control restoring moment coefficient

The undamped natural frequency was chosen as 0.2 Hertz, which is well within the guideline of one-fifth of the first body bending mode frequency of 1.7 Hertz. The damping ratio of 0.7 was selected to yield a relatively fast response with very little overshoot.

Using this damping ratio, this frequency and the characteristics of the vehicle, the flight control system gains were computed as a function of flight time. Figure 4.1.4.1-1 shows the attitude error and attitude rate gains as a function of flight time. The discontinuity at approximately 146 seconds is caused by the reduction in the control restoring moment coefficient when the two engines are cut off. This discontinuity should not cause any problems if a piecewise continuous gains program is used for the INT-20 configuration as is used for the Saturn V.

c. Uncontrolled Divergence

The time required for the angular position to double in amplitude and the control authority ratio were determined in order to gain an insight into abort warning time. A rigid body model and static conditions were assumed. The time to double amplitude (TDA) and control authority ratio (CAR) are shown as a function of flight time in Figure 4.1.4.1-2. The TDA is greater than two seconds for all flight times, and the CAR is at least adequate. Comparison of the smallest TDA and CAR with those of SA-505 shows that the INT-20 baseline configuration has less capability to maintain control after an engine or actuator malfunction. The TDA and CAR are 2.95 sec. and 5.65 for the SA-505 and 2.01 and 5.3 for the INT-20 respectively. The effects of the smaller TDA for the INT-20 become evident in the abort study contained in sub-paragraph f, below.

d. Thrust Deflection Duty Cycle

A continuous-time, rigid body simulation was made to determine the thrust vector deflection duty cycle. The appropriate scatter was applied in the worst direction and the Apollo design wind was used to obtain these results.

4.1.4.1 (Continued)

The thrust deflection was smaller than that which was obtained from the rigid body point time study (RSS value) because the wind shear is less severe than it was for the point-time rigid body study. The thrust deflection duty cycle, angle of attack, and wind envelope are shown in Figure 4.1.4.1-3.

e. RSS Parameter Envelopes

The RSS envelopes for the basic flight parameters were obtained by applying the Saturn V/Apollo design wind at selected time points during the maximum dynamic pressure region of flight. Perturbations in the vehicle characteristics were applied to yield the most severe control requirements. The scatter parameters and their values are:

Thrust Imbalance	±	1.5%
Thrust Misalignment	±	0.122 degrees
Axial CG offset	±	7 inches
Lateral CG offset	±	2 inches
Normal Force Coefficient Variations	±	6 %
Center of Pressure Offset	±	79 inches
Control Gains Variation	±	10%

RSS envelopes represent the maximum values of the basic flight parameters regardless of when the maximum wind disturbance occurs. The gains used to obtain these data were taken from Figure 4.1.4.1-1, the values corresponding to the time point under investigation.

Figure 4.1.4.1-4 shows the RSS envelopes for the gimbal angle and angle of attack with the dynamic pressure shown for comparative purposes. The RSS gimbal angle does not include the factor of 0.465 degrees which is usually included to account for disturbances caused by vehicle bending modes. Figure 4.1.4.1-5 shows the RSS angular rate, angular acceleration, and lateral acceleration envelopes during the corresponding maximum dynamic pressure region of flight.

f. Abort

An abort analysis was made to compare the post-engine-out control capability of the INT-20 vehicle with that of a typical Saturn V (Reference 4.1.4.1-2). The analysis was made using vehicle conditions at the time of flight at which maximum dynamic pressure (q max) is experienced. The design wind, a 95 percentile wind profile for the month of March, was used to obtain the maximum wind speed expected at the altitude at which q max occurs. This wind speed was

4.1.4.1 (Continued)

246 feet/second (75 m/sec) and was applied as a tailwind with a 25.1 feet/second (7.65 m/sec) embedded gust of 0.6 seconds duration. Also a 108.3 feet/second (35m/sec) wind, with embedded gust, was used to approximate the angle of attack (α) which would be developed if a tailwind-biased trajectory (biased based on 50 percentile winds) were used with the 75 m/sec wind. Thus, the data shown for the 75 and 35 m/sec winds represent a comparison between a 75 m/sec wind with and without a tailwind-biased trajectory. The winds used are shown in Figure 4.1.4.1-6. Since the winds are applied at the q max region, t = 0 represents a reference time slightly before the time of q max.

A nominal pitch-plane response, all engines operating, was obtained for both the INT-20 and the Saturn V vehicles using both winds. Lower and upper controllability bounds were established by simulating cutoff of one engine at various times from the t = 0 reference at q max. The bounds define the time period during which the thrust vector control (TVC) cannot maintain vehicle control upon loss of an engine. These controllability boundaries were determined with and without engines canted.

The nominal pitch-plane responses, no engine out, for the INT-20 vehicle under the influence of the 35 m/sec wind are shown in Figure 4.1.4.1-7. Responses, obtained with the 35 m/sec wind, for the lower and upper engine-out bounds, engines not canted, are shown in Figures 4.1.4.1-8 and 4.1.4.1-9. The lower and upper bounds occur at 0.5 and 3.4 seconds, respectively, from the zero time reference. Dotted lines indicate uncontrollable response trends. Note that the upper bound is prior to the peak wind speed due to the lag between angle of attack and gimbal angle. The peak wind speed is reached prior to the maximum gimbal angle and by the time the maximum gimbal angle is reached, the wind speed is decreasing (refer to Figures 4.1.4.1-6 and 4.1.4.1-9). These boundaries indicate that the TVC would not be able to maintain or recover controllability for engine-out times between 0.5 and 3.4 seconds. However, with engines canted at 1.5 degrees, so that the thrust vector passes closer to the center of gravity, the INT-20 vehicle was controllable for all engine-out times. The responses for the engine-out time that resulted in maximum angle of attack and attitude error are shown, for engines canted, in Figure 4.1.4.1-10.

Nominal and engine out responses for the Saturn V, with the 35 m/sec wind, are shown in Figures 4.1.4.1-11 and 4.1.4.1-12. The engine out time was 2.4 seconds, which resulted in the maximum disturbances. At no time did cutoff of an engine cause an uncontrollable situation, whether engines were canted or not. Note that peak values of the responses for the Saturn V are smaller than for the INT-20. This is because the control authority ratio and time to double amplitude is smaller for the Saturn V than for the INT-20.

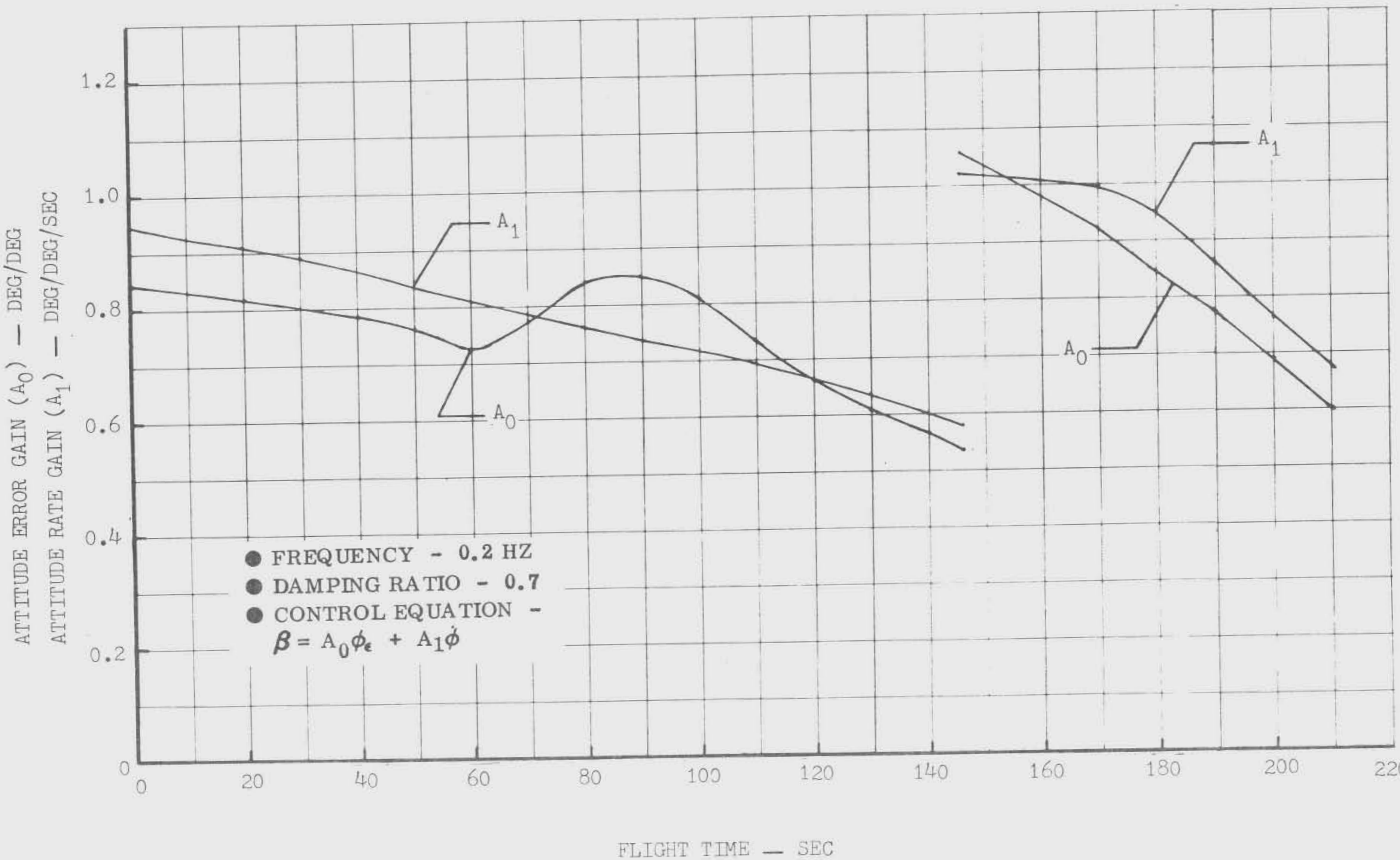


FIGURE 4.1.4.1-1 RIGID BODY FLIGHT CONTROL SYSTEM GAINS FOR INT-20 DURING FIRST STAGE FLIGHT

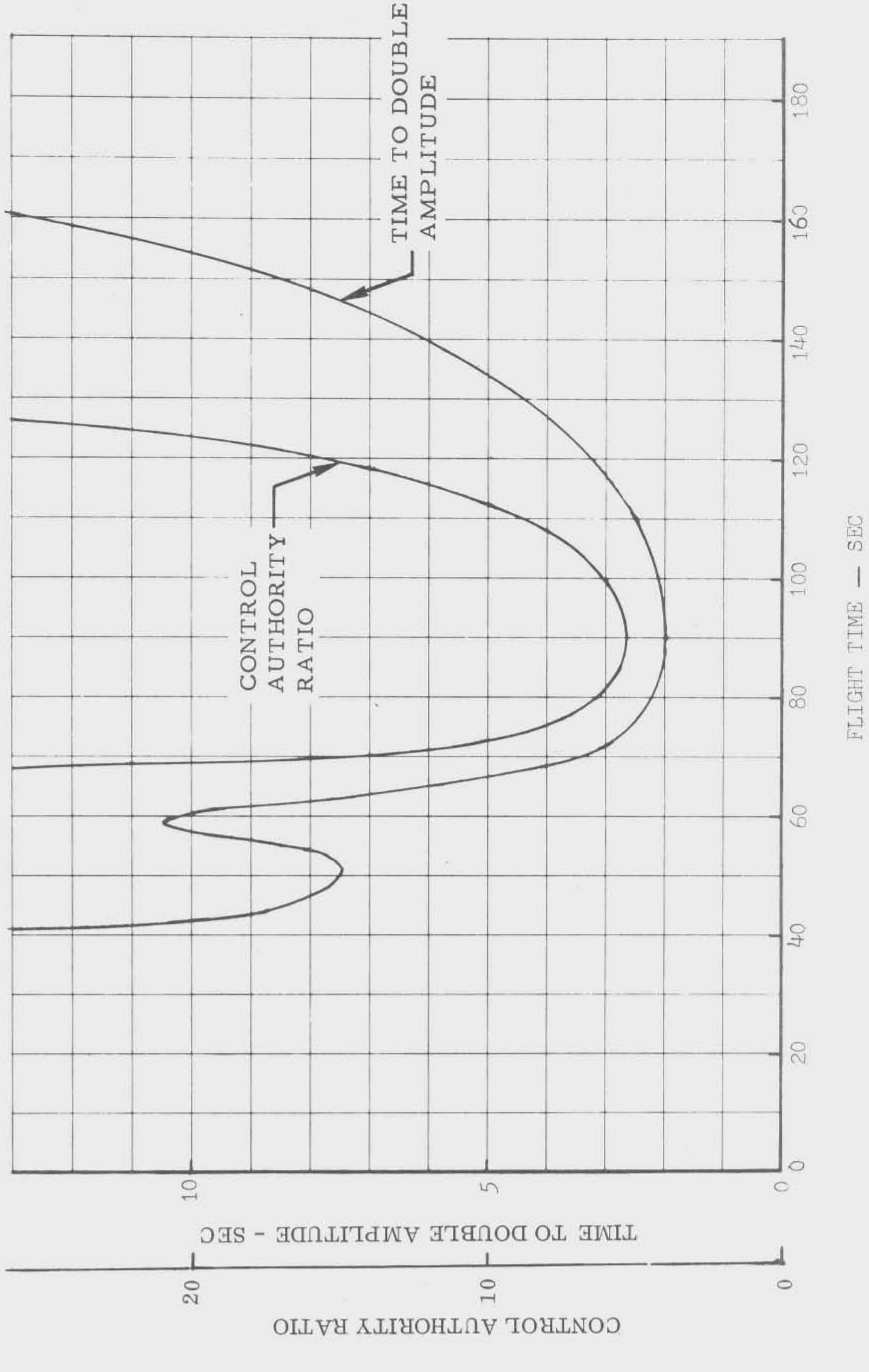


FIGURE 4.1.4.1-2 TIME TO DOUBLE AMPLITUDE AND CONTROL AUTHORITY RATIO

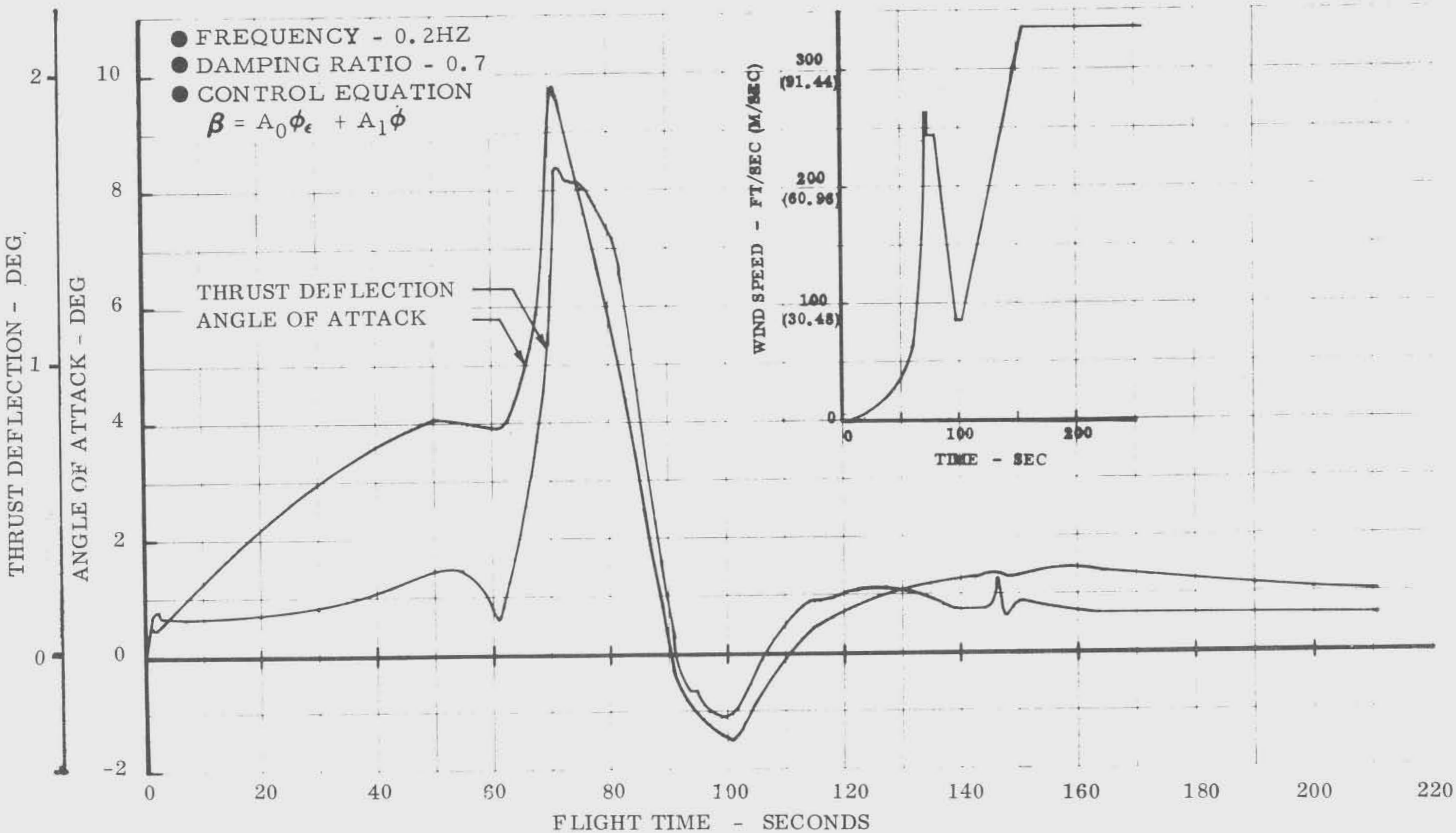


FIGURE 4.1.4.1-3 DUTY CYCLE AND ANGLE OF ATTACK TIME HISTORIES FOR THE INT-20 BASELINE CONFIGURATION

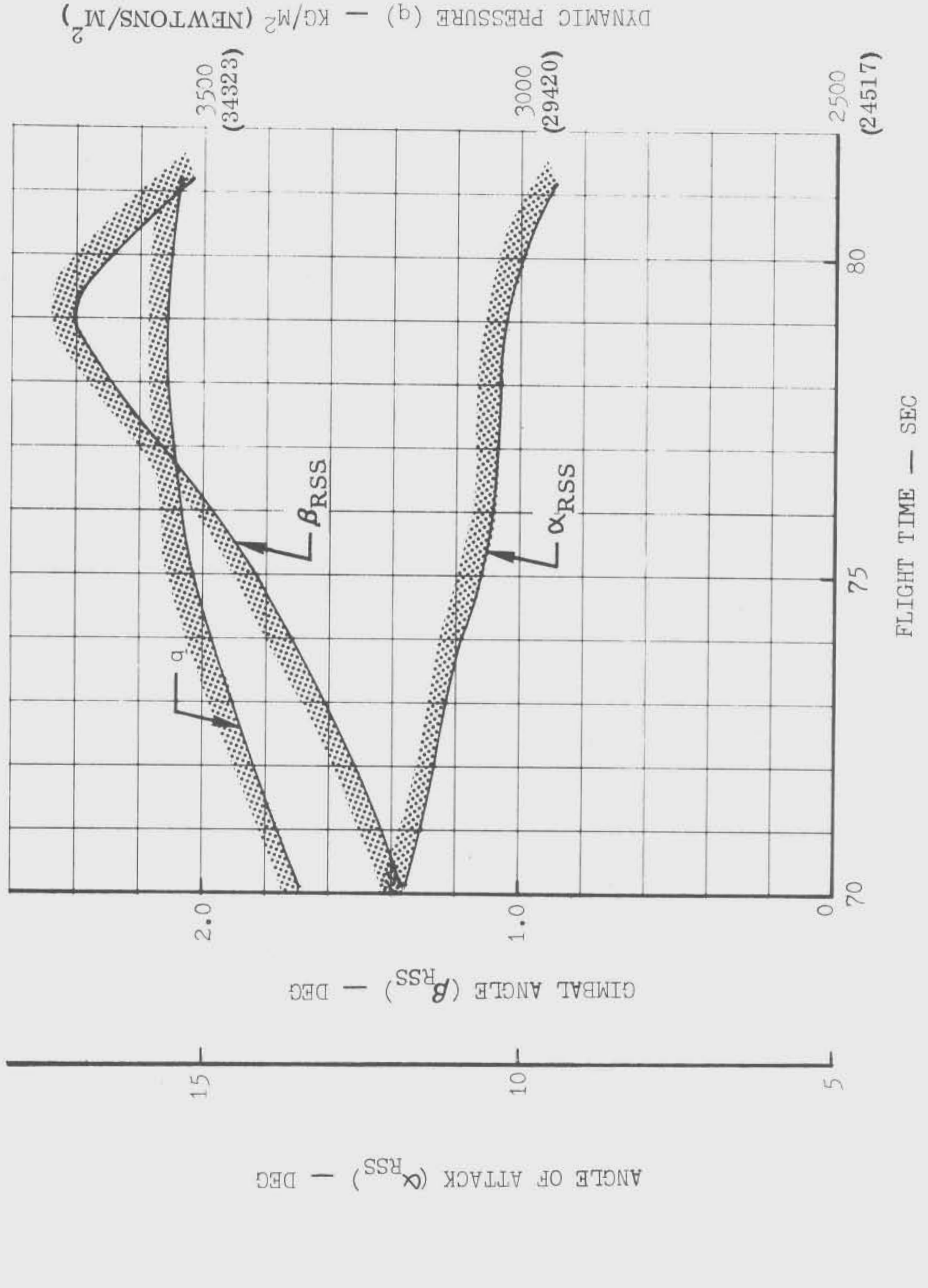


FIGURE 4.1.4.1-4 INT-20 DYNAMIC PRESSURE, ANGLE OF ATTACK, AND GIMBAL ANGLE FOR THE q_{MAX} REGION OF FLIGHT

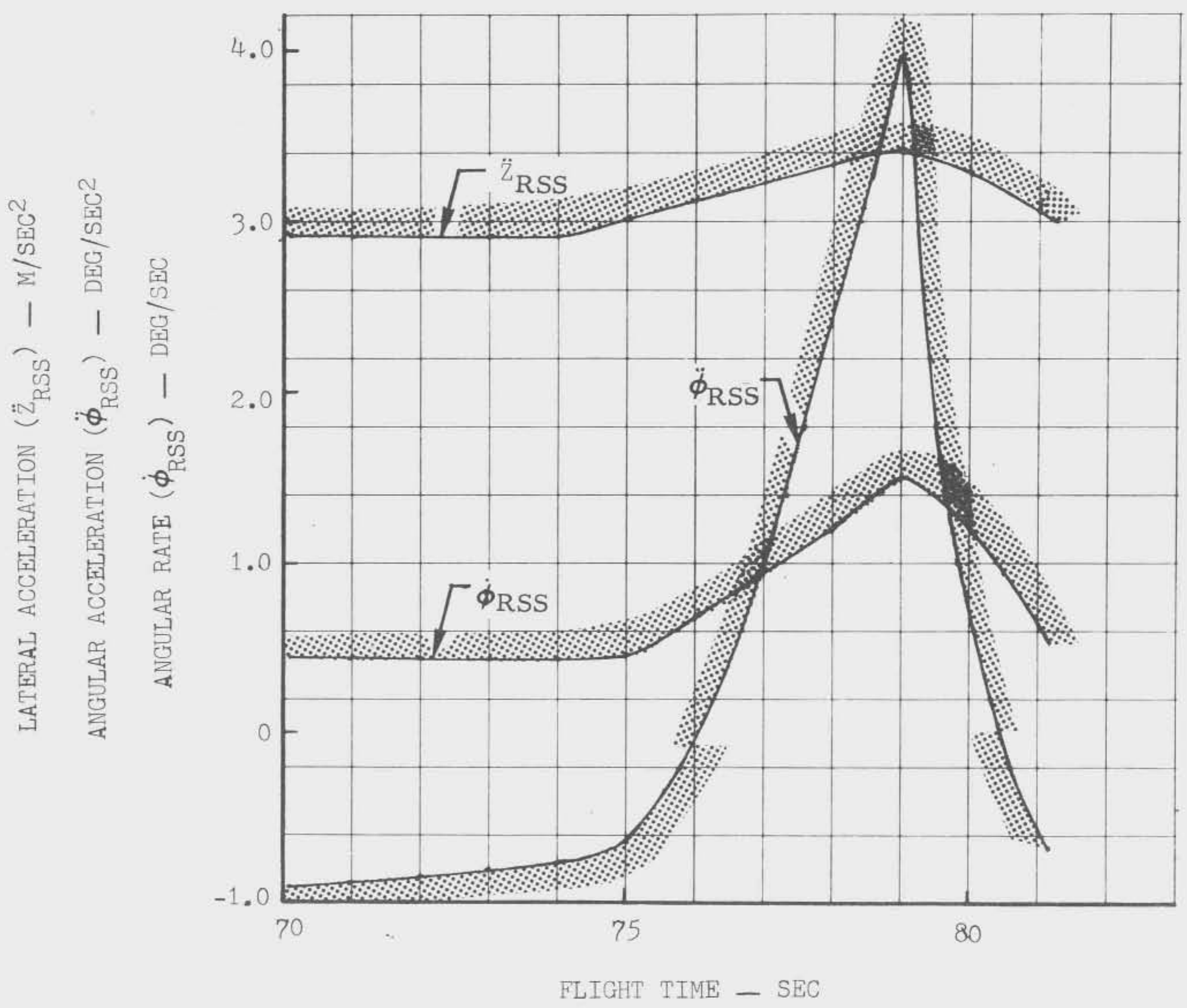


FIGURE 4.1.4.1-5 INT-20 LATERAL ACCELERATION, ANGULAR ACCELERATION, AND ANGULAR RATE FOR THE q_{MAX} REGION OF FLIGHT

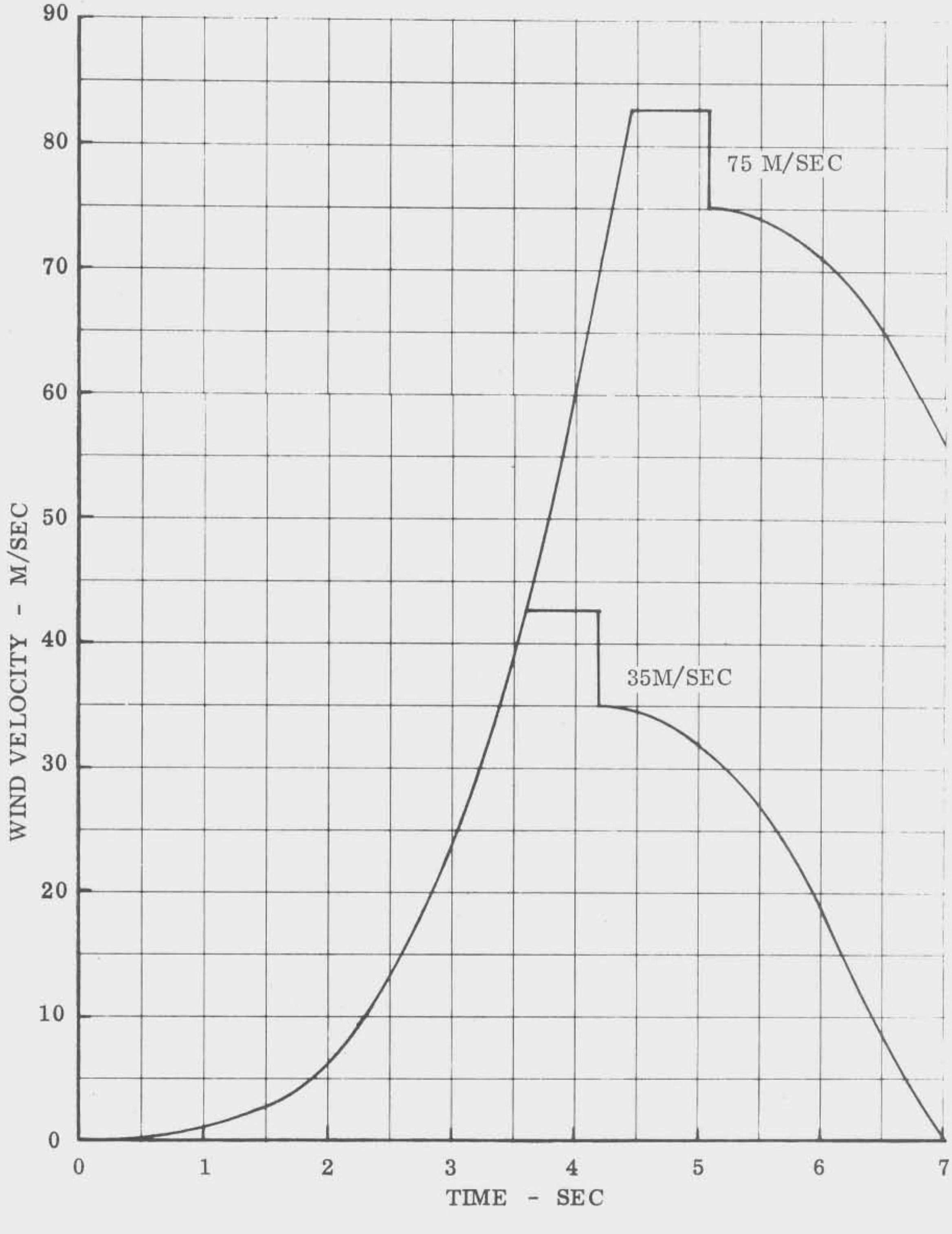


FIGURE 4.1.4.1-6 WIND USED FOR ENGINE OUT ANALYSIS FOR INT-20

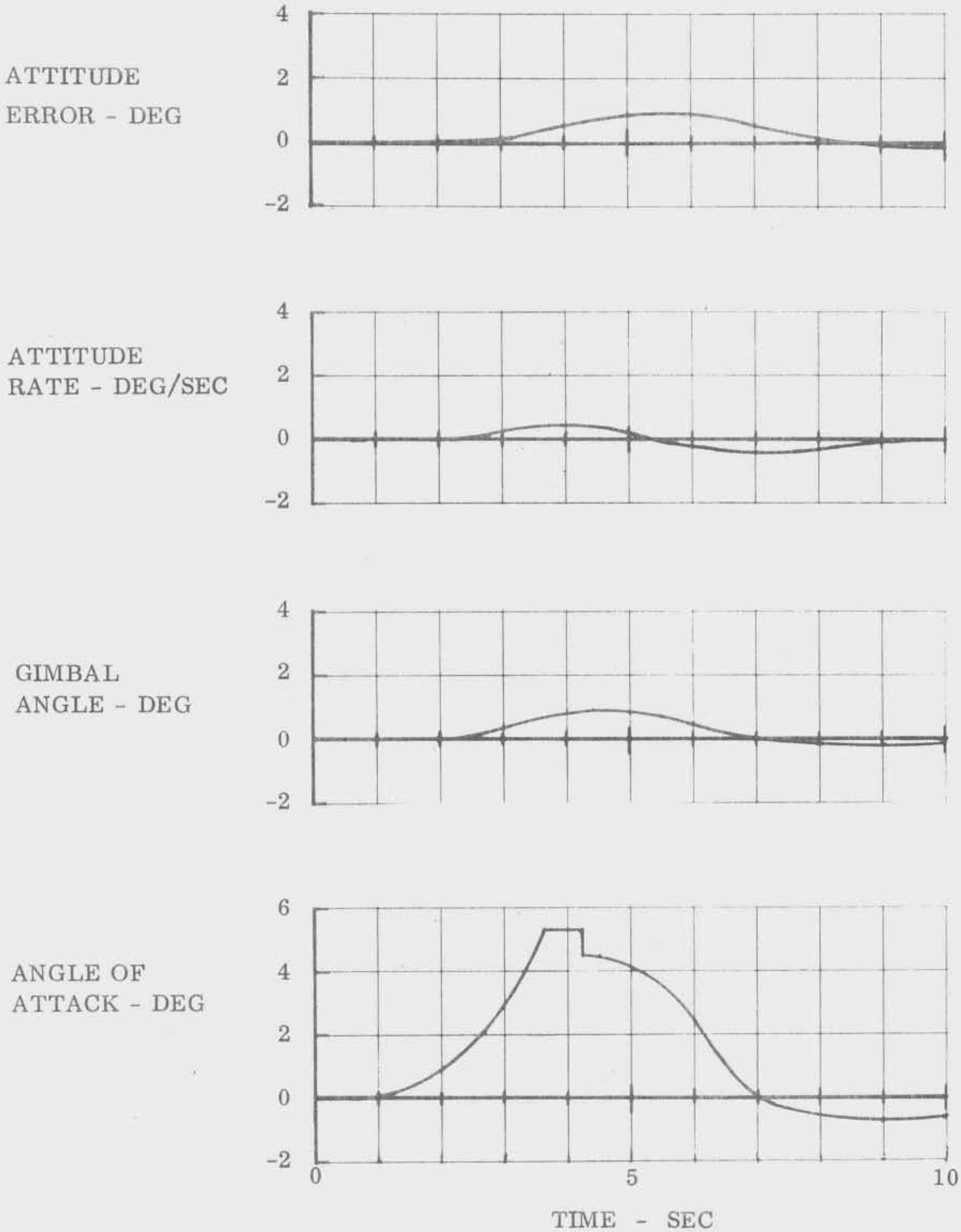


FIGURE 4.1.4.1-7 NOMINAL RESPONSES FOR INT-20, V_W= 35.0 M/SEC

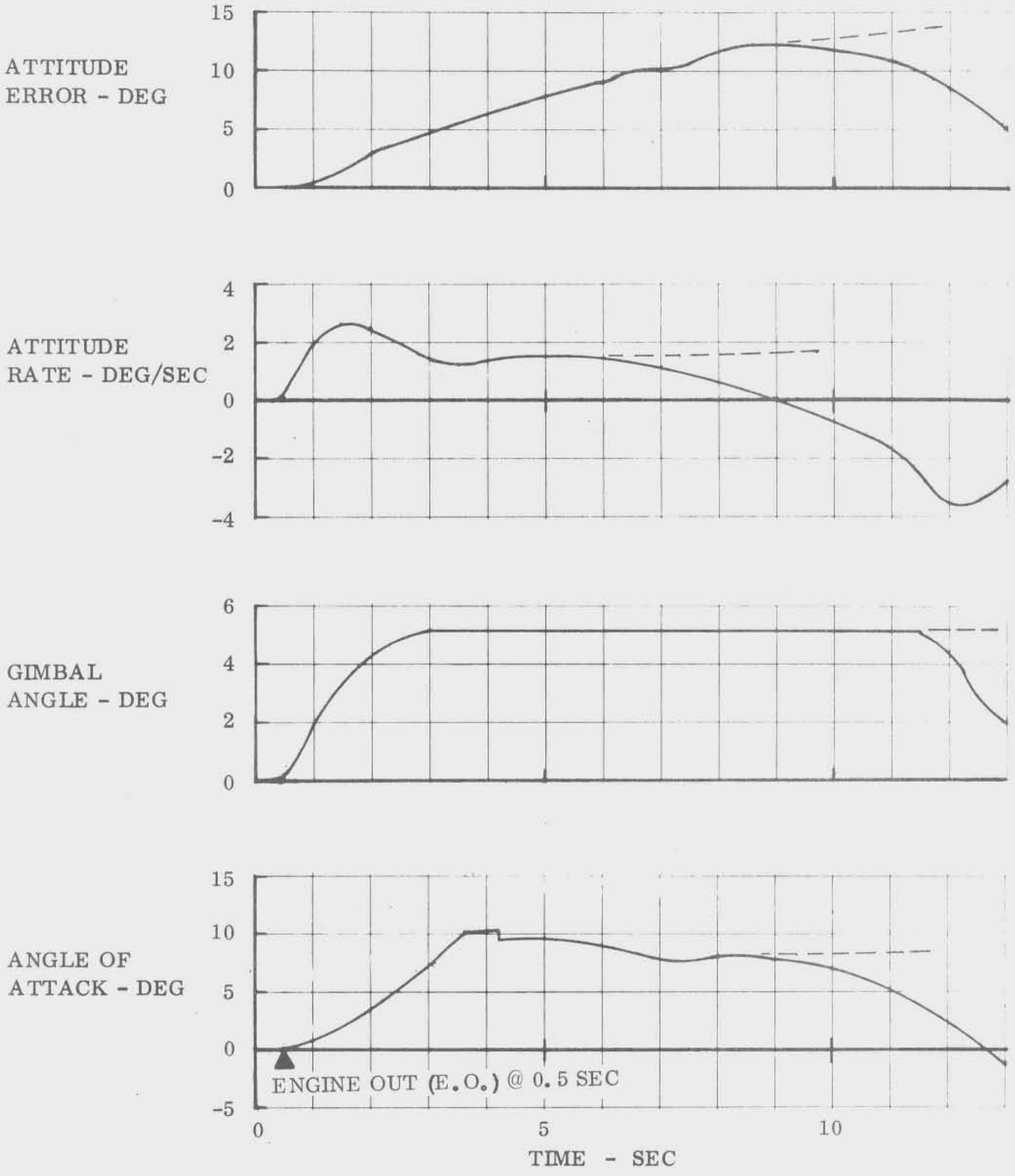


FIGURE 4.1.4.1-8 LOWER BOUND ENGINE OUT RESPONSES FOR INT-20, V_W = 35.0 M/SEC

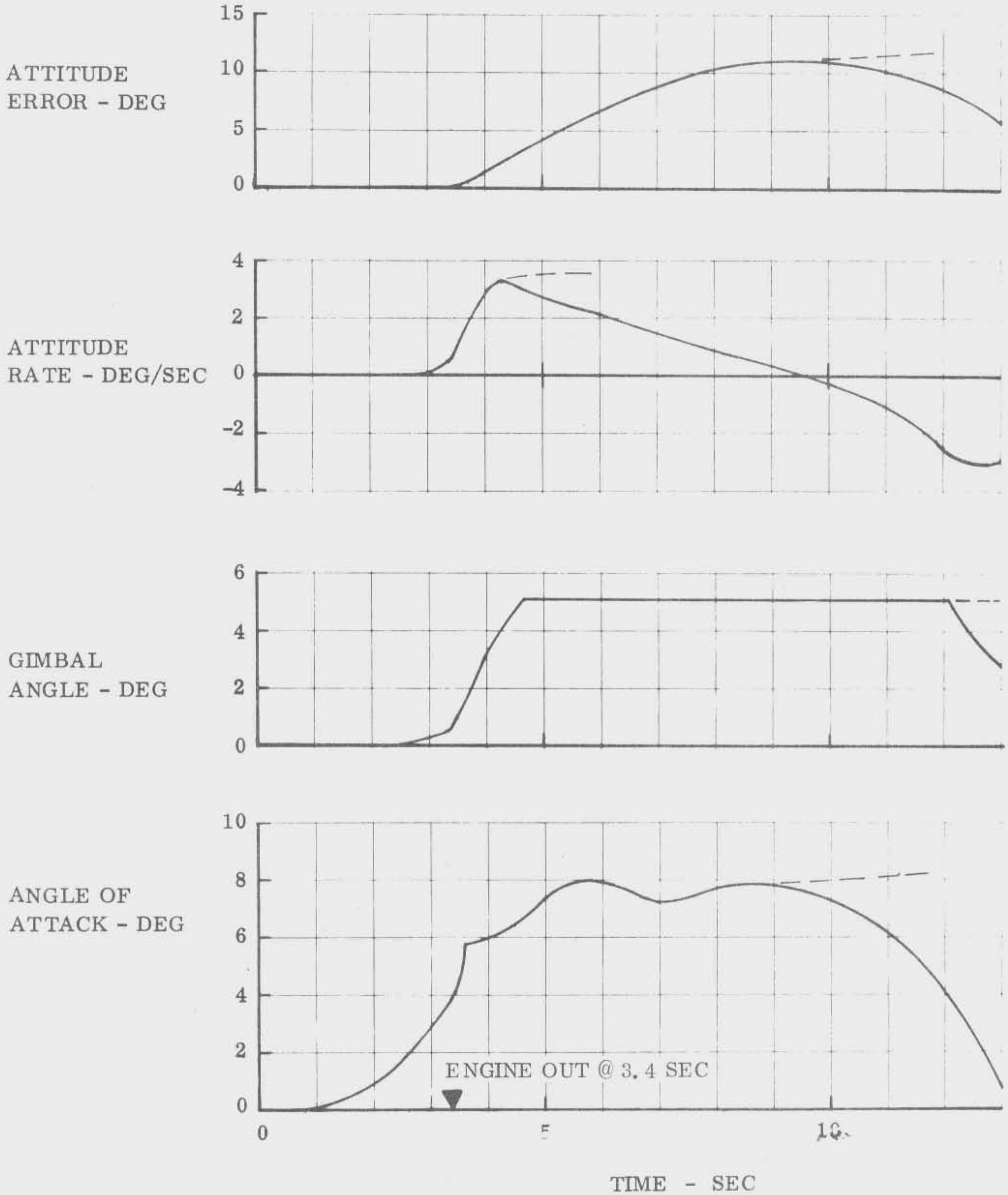


FIGURE 4.1.4.1-9 UPPER BOUND ENGINE OUT RESPONSES FOR INT-20, V_W= 35.0 M/SEC

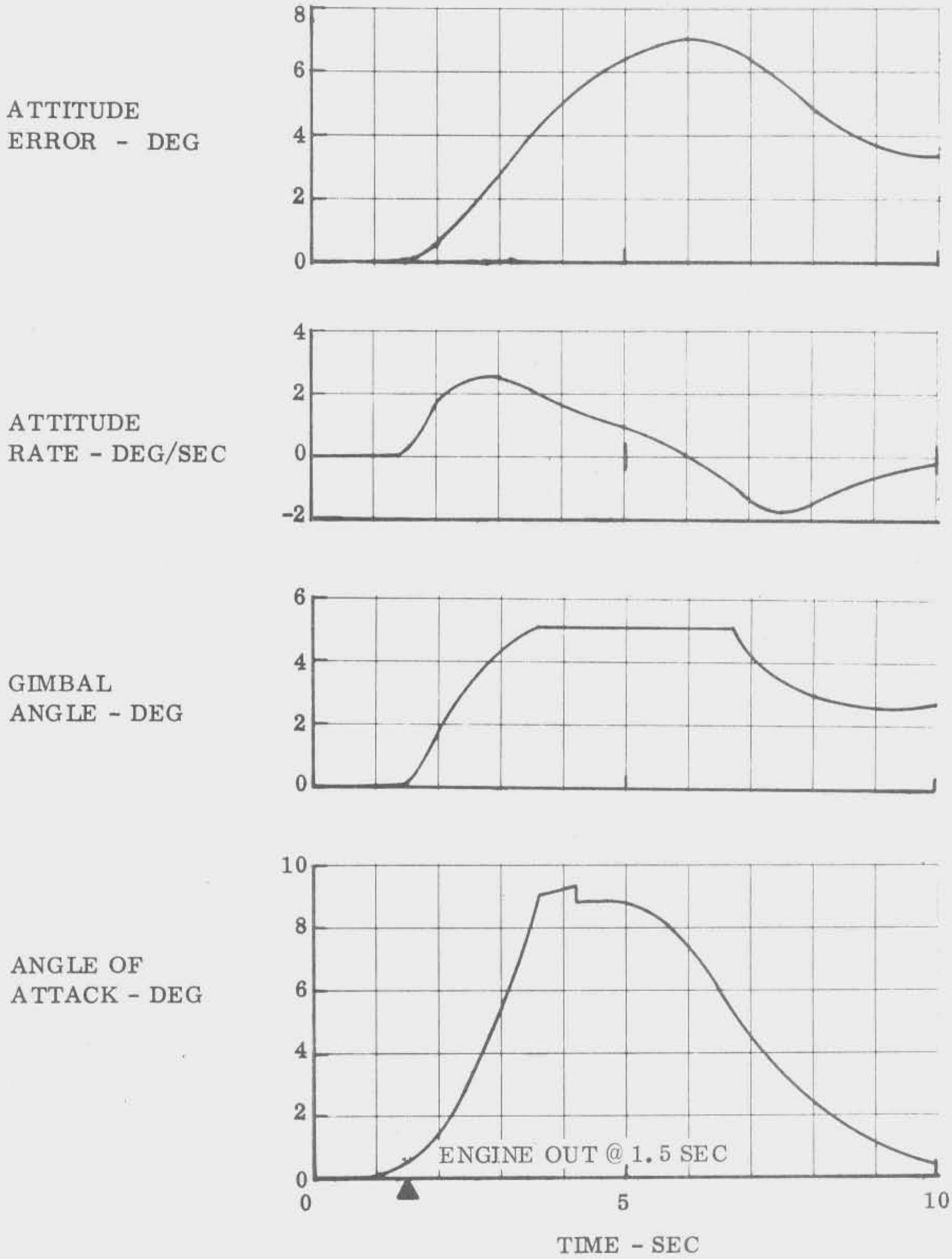


FIGURE 4.1.4.1-10 ENGINE OUT RESPONSES FOR INT-20 USING A 1.5 DEGREE CANT ANGLE, V_W= 35.0 M/SEC

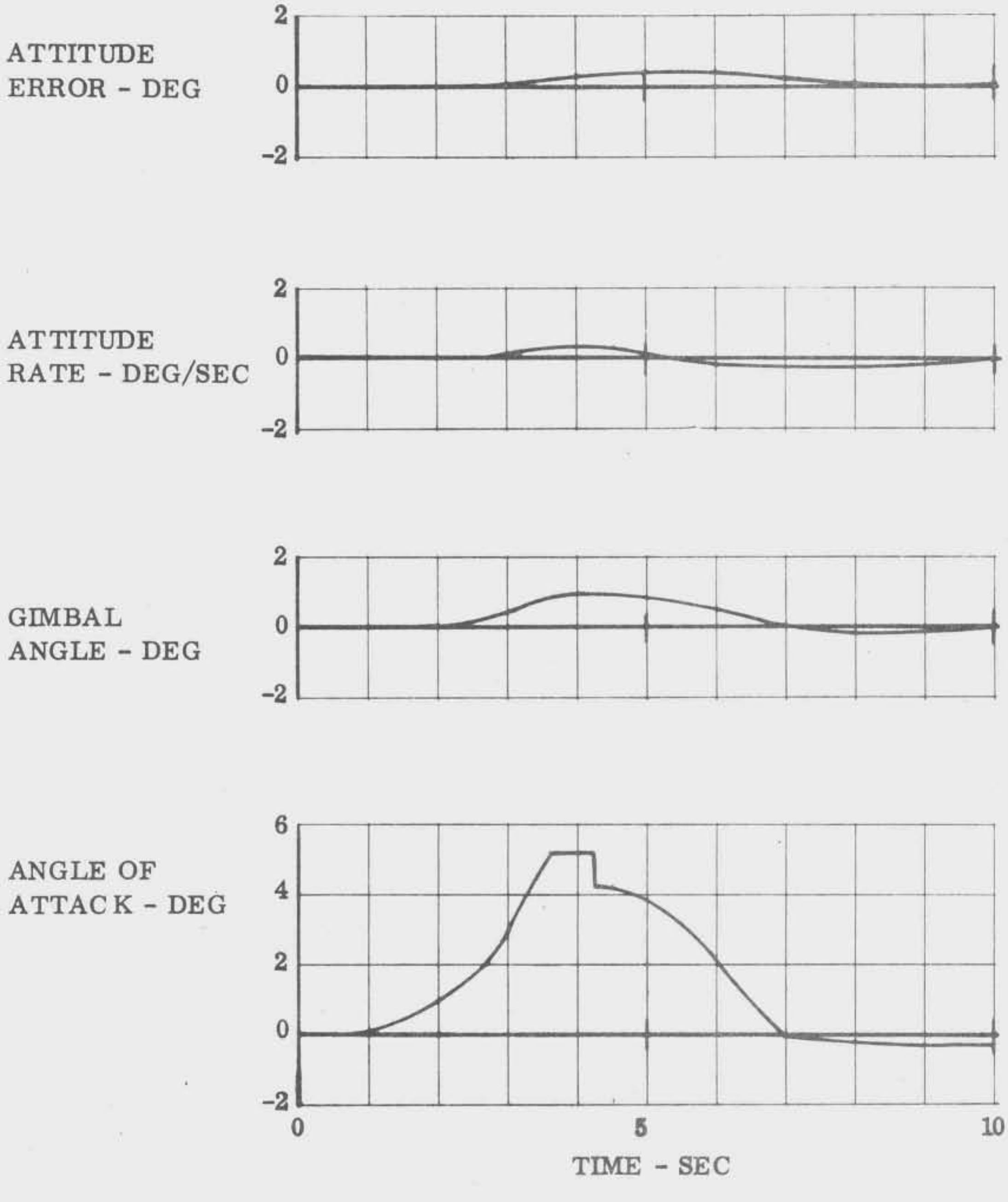


FIGURE 4.1.4.1-11 NOMINAL RESPONSES FOR SATURN V, V_W= 35.0 M/SEC

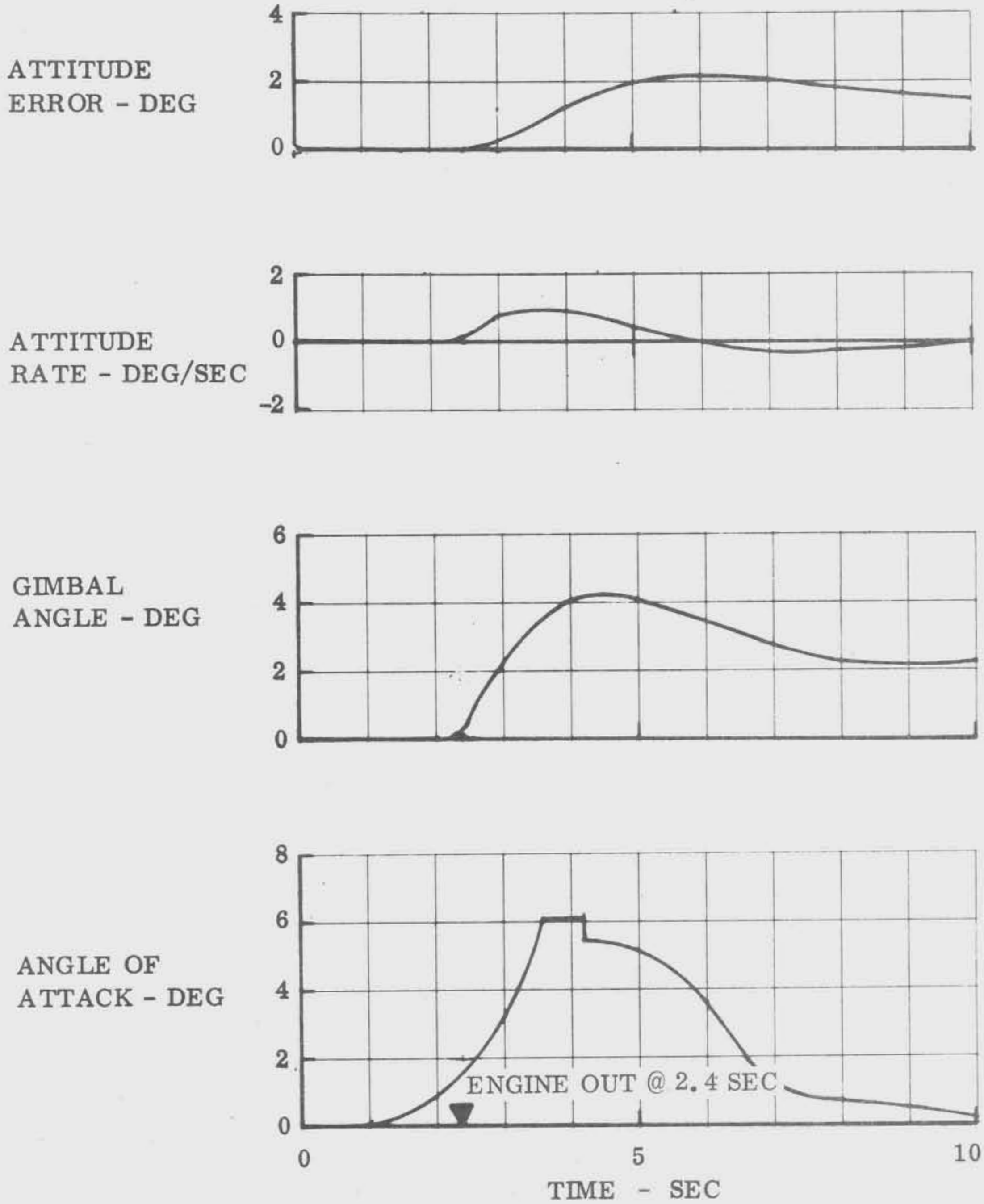


FIGURE 4.1.4.1-12 ENGINE OUT RESPONSES FOR SATURN V, V_W= 35.0 M/SEC

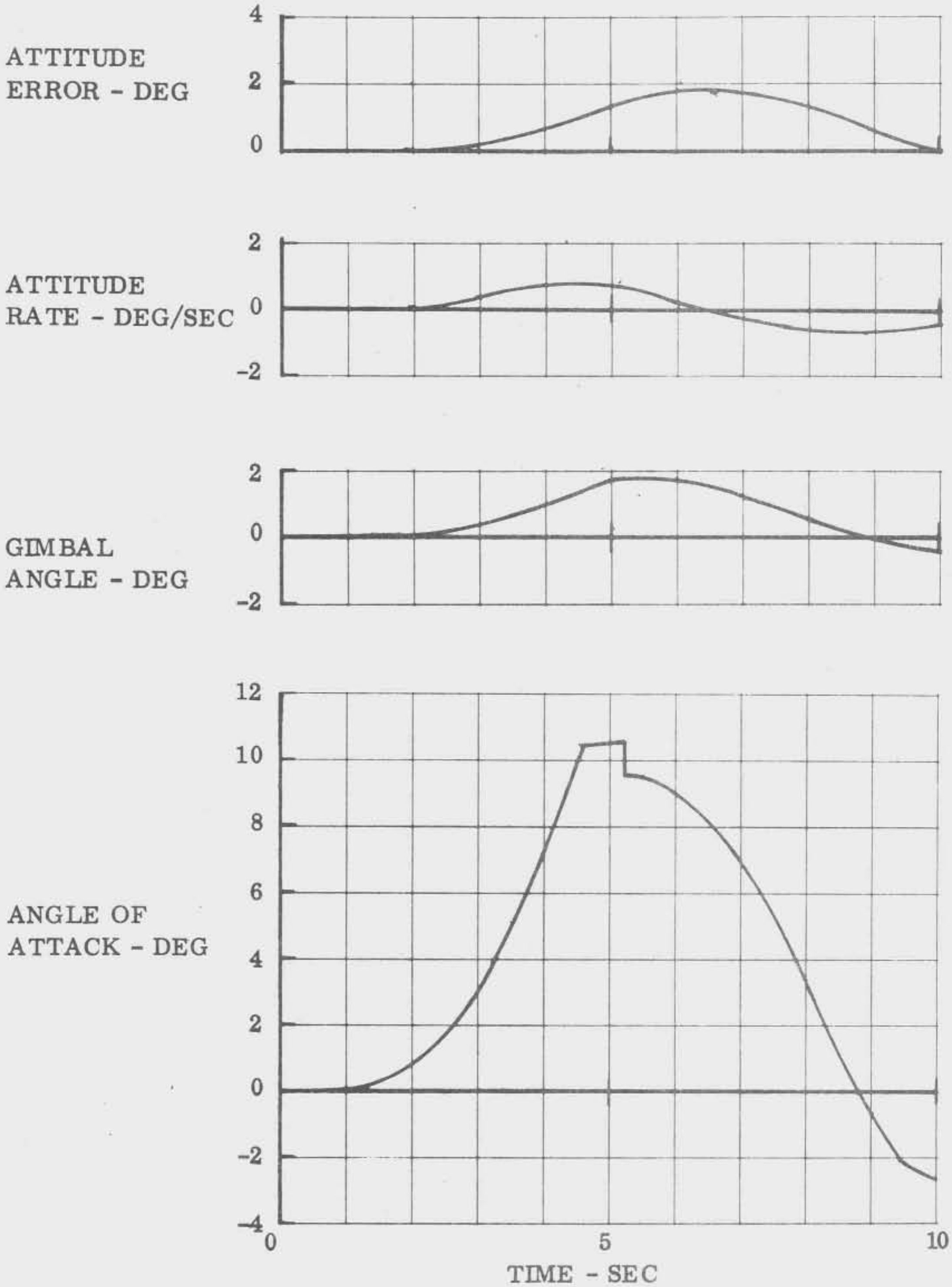


FIGURE 4.1.4.1-13 NOMINAL RESPONSES FOR INT-20, V_W= 75.0 M/SEC

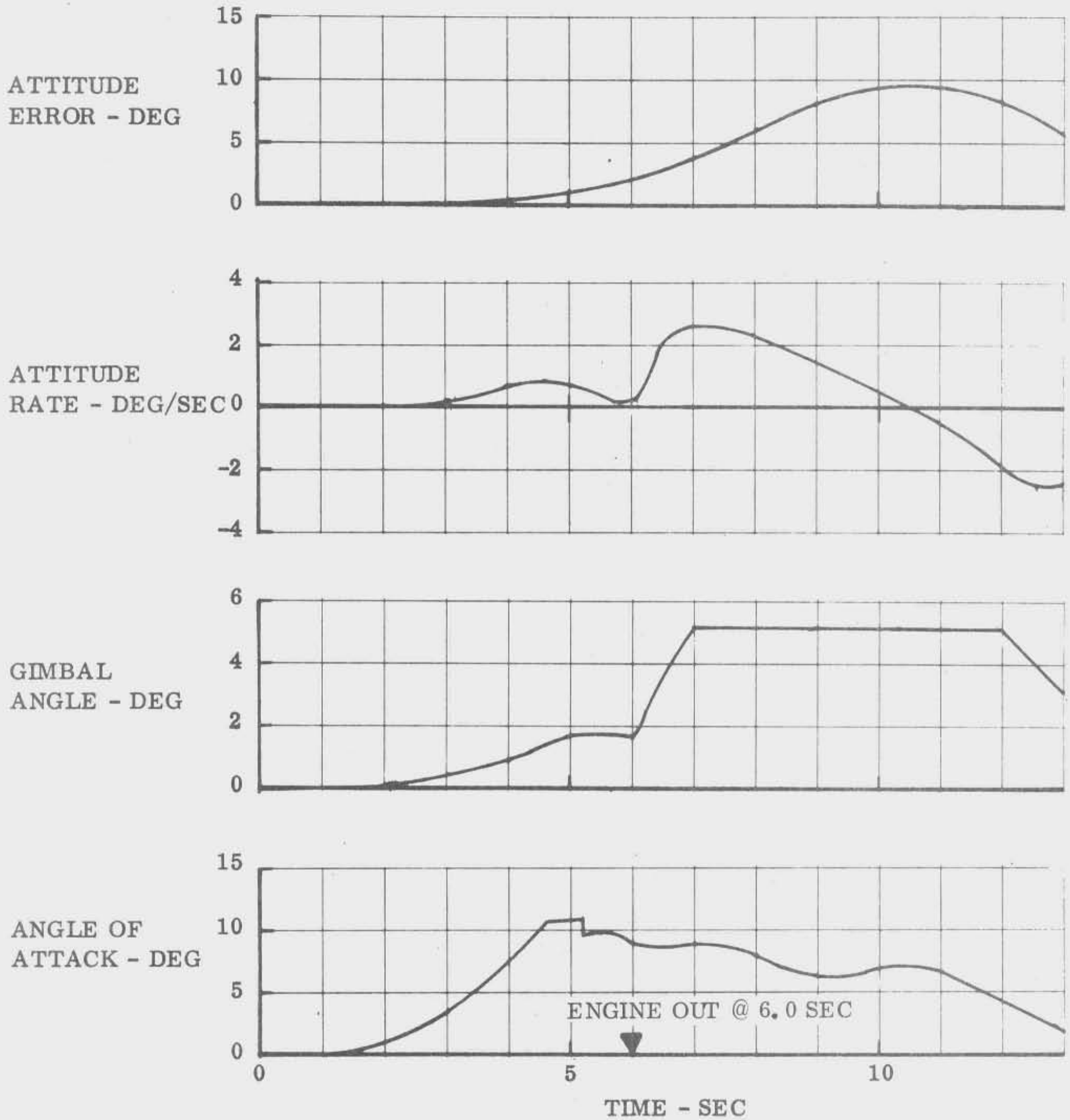


FIGURE 4.1.4-14 UPPER BOUND ENGINE OUT RESPONSES FOR INT-20, V_W= 75.0 M/SEC

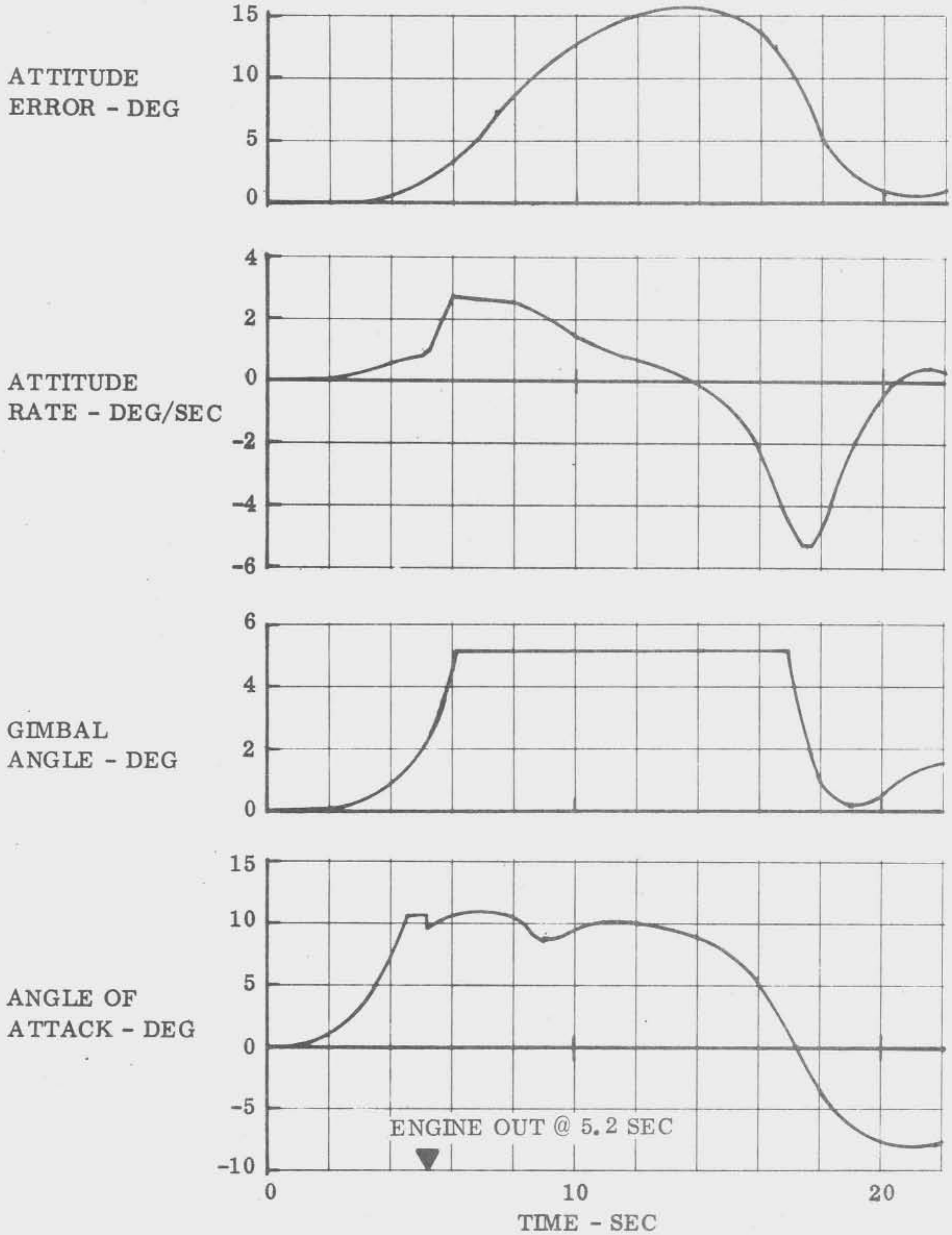


FIGURE 4.1.4.1-15 UPPER BOUND ENGINE OUT RESPONSES FOR INT-20 USING A 1.5 DEGREE CANT ANGLE, V_W= 75.0 M/SEC

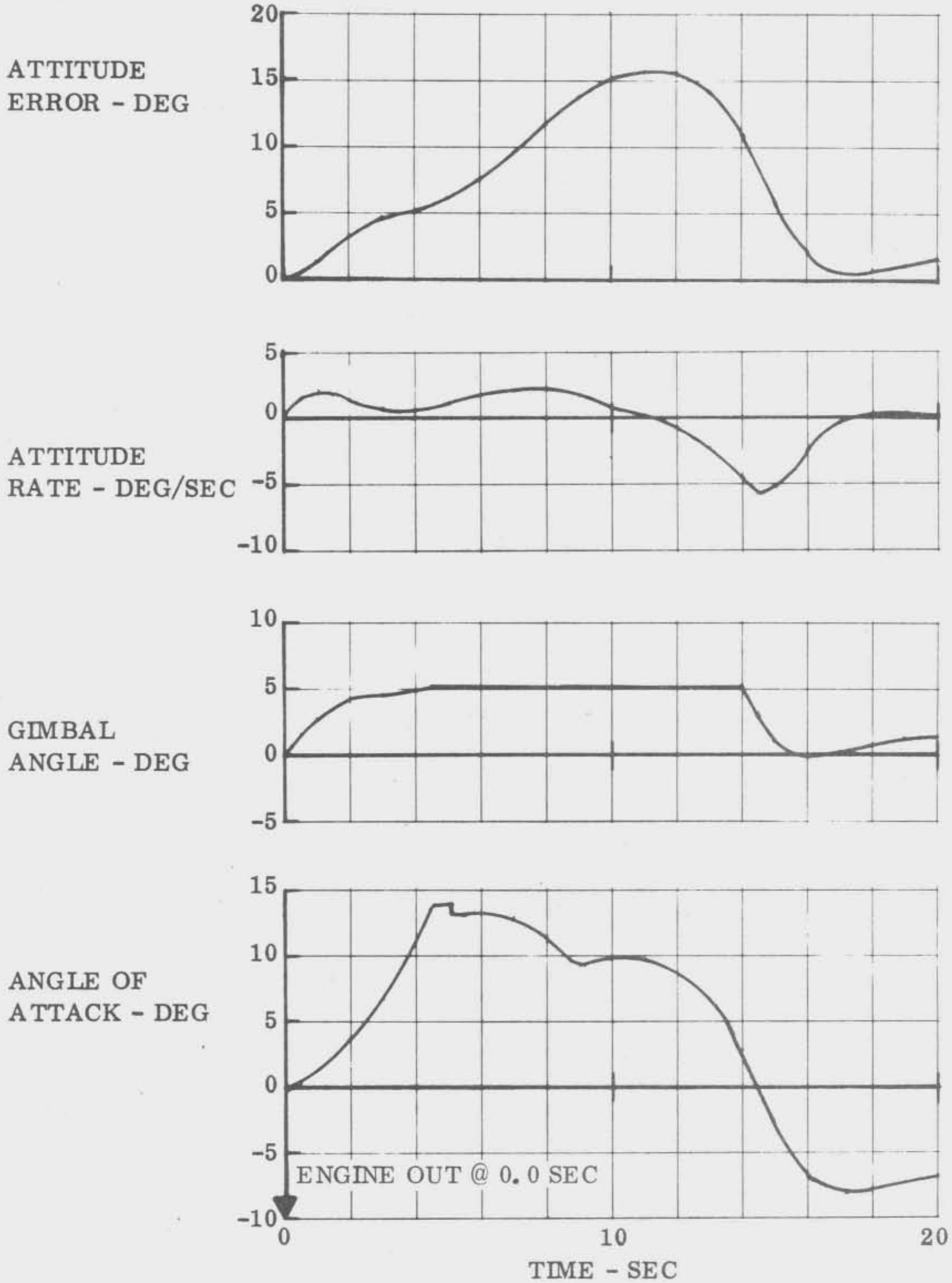


FIGURE 4.1.4.1-16 LOWER BOUND ENGINE OUT RESPONSES FOR INT-20 USING A 2.42 DEGREE CANT ANGLE, V_W= 75.0 M/SEC

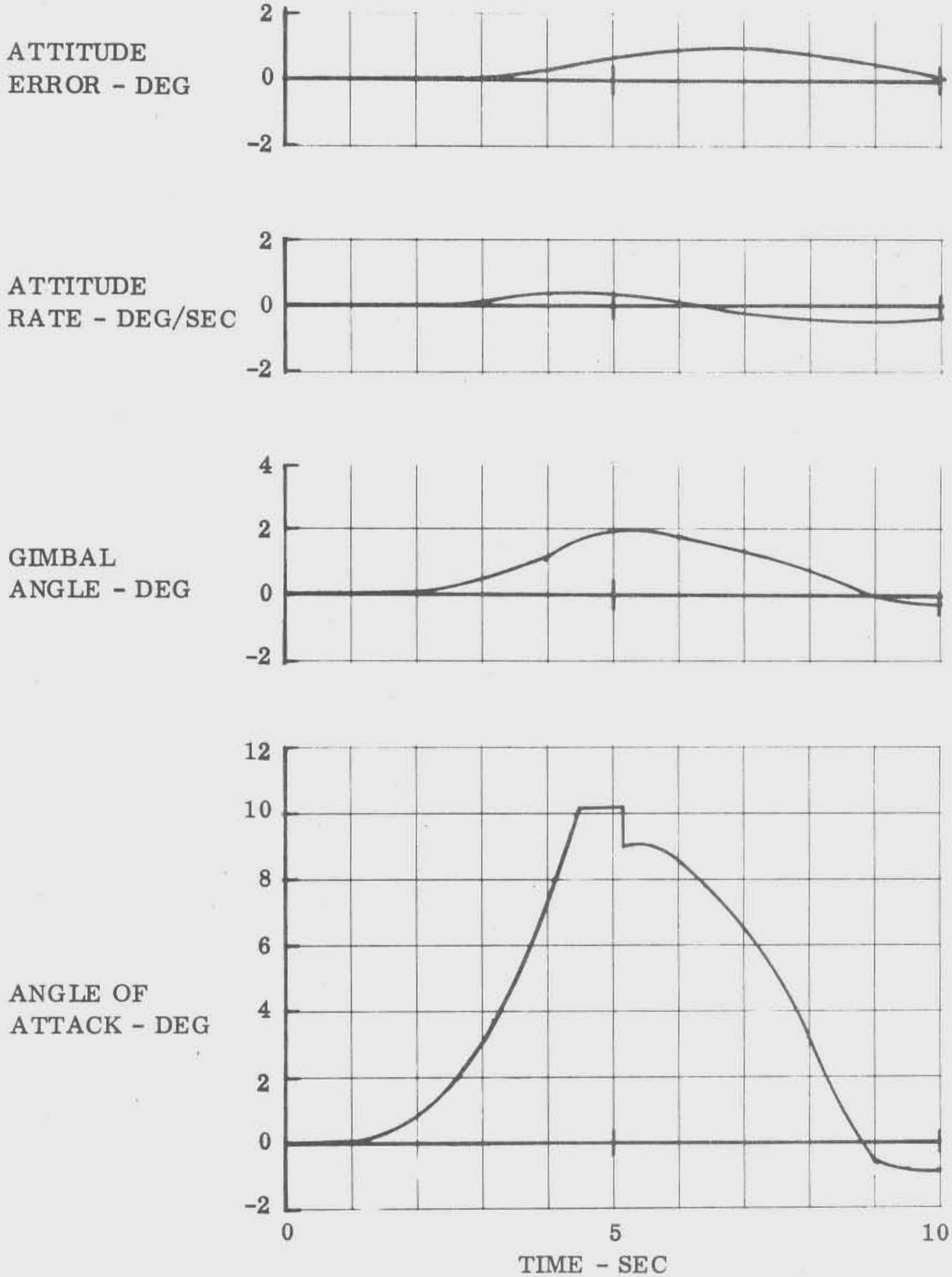


FIGURE 4.1.4.1-17 NOMINAL RESPONSES FOR SATURN V, V_W= 75.0 M/SEC

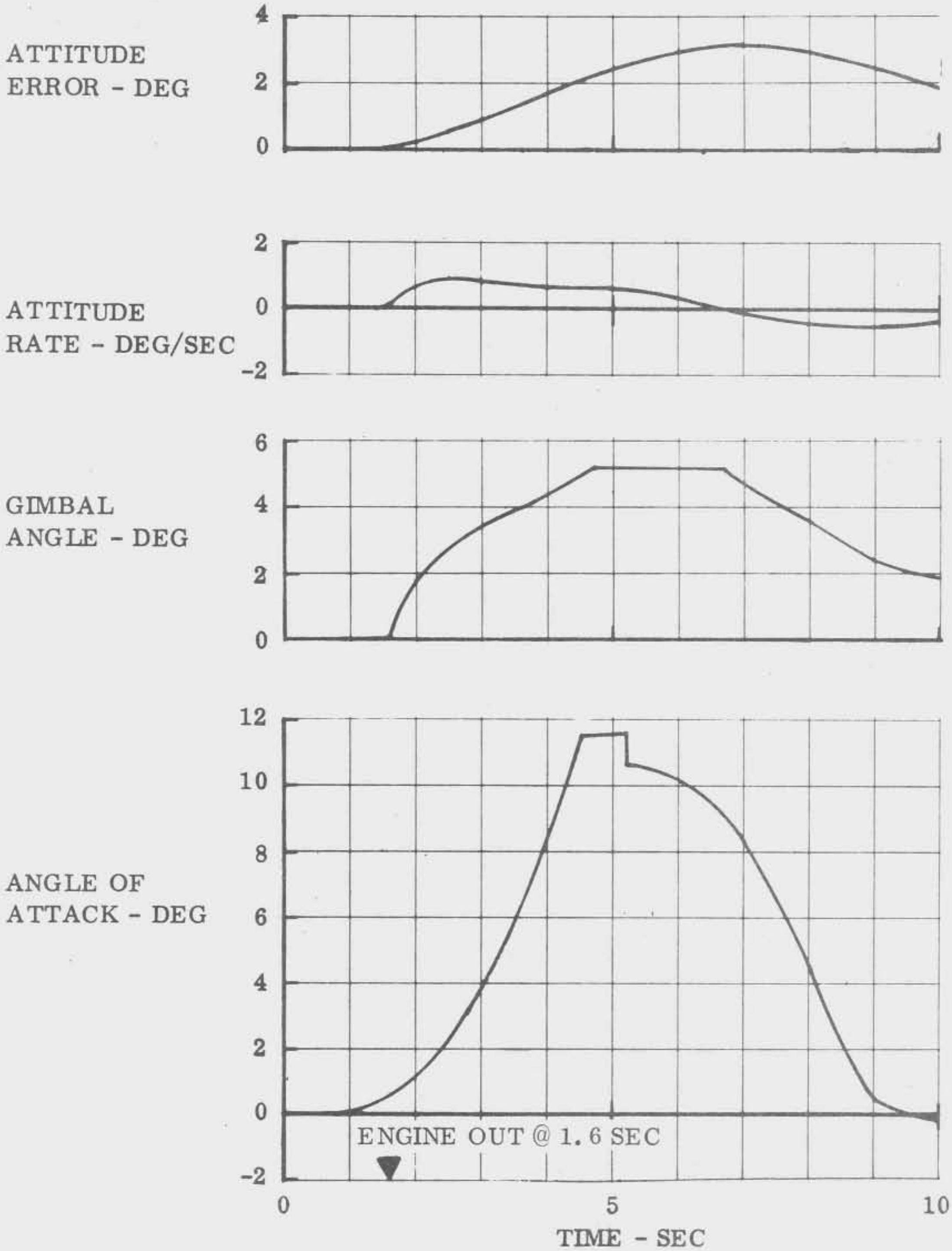


FIGURE 4.1.4.1-18 ENGINE OUT RESPONSES FOR SATURN V, V_W= 75.0 M/SEC

4.1.4.1 (Continued)

The nominal responses for the INT-20 when subjected to the 75 m/sec wind are shown in Figure 4.1.4.1-13. The responses, with no engine cant, for the upper bound of engine-out time, which is 6 seconds, are shown in Figure 4.1.4.1-14. No lower bound exists, i.e., if the engine is cut off between 0 and 6 seconds, the TVC system cannot maintain controllability. The upper bound was reduced from 6.0 to 5.2 seconds using an engine cant angle of 1.5 degrees (Figure 4.1.4.1-15). The lower bound of controllability was established at zero seconds (engine out at t = 0) with 2.42 degrees of engine cant. The responses are shown for 2.42 degrees of engine cant in Figure 4.1.4.1-16. Note that the attitude error is large.

The effects of the 75 m/sec wind upon the Saturn V, nominal and engine out, are shown in Figures 4.1.4.1-17 and 4.1.4.1-18. The responses shown in Figure 4.1.4.1-18 are for engine out at 1.6 seconds (worst case) and no cant angle.

The data described above indicate that post-engine-out control capability for the INT-20 is less than that of the typical Saturn V. Abort limits have not been defined for the INT-20 configurations; however, a detailed analysis should be made to determine the controllability in a potential abort situtation.

4.1.4.2 Lift-Off Dynamics

A launch tower clearance analysis was made to determine the lift-off characteristics of the INT-20. Phase I analyses showed that the INT-20, like Saturn V, would require preprogrammed yaw to avoid collision with the tower when under the influence of design ground winds and worst-cast scatter at lift-off. This analysis verified the Phase I conclusions.

The analysis was made using a rigid body vehicle, lumped mass, and a nine-panel aerodynamic mode. A 95 percentile ground wind, as defined in Reference 4.1.4.2-1 was used. The profile was constructed by superimposing a gust on the steady-state wind, as shown in Figure 4.1.4.2-1. The gust was simulated by causing the wind speed to build up from the steady-state value to a peak value and then return to steady-state. Two seconds were allowed for gust build-up and two seconds for gust decay. The gust was applied at vehicle liftoff because that resulted in the most severe drift. The effects of the gust were experienced at the vehicle center of pressure (cp) as shown in Figure 4.1.4.2-1. Wind direction was such that the vehicle was caused to drift toward the tower.

The scatter parameters applied with the wind were based on Saturn V data and are listed below with their values or percentages:

Engine Thrust Imbalance	1.5%
Axial CG Offset	7.75 inches

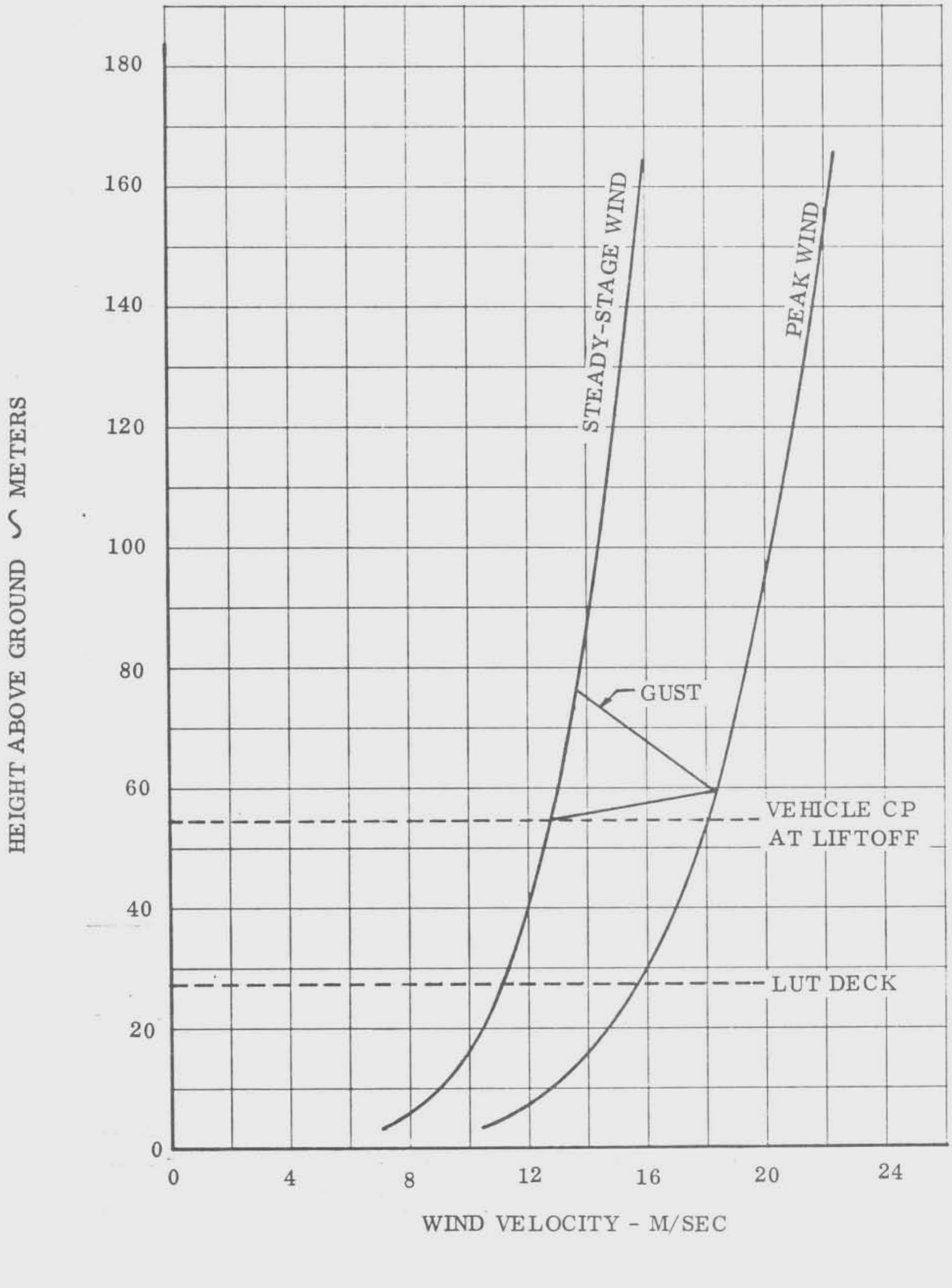


FIGURE 4.1.4.2-1 95 PERCENTILE GROUND DESIGN WIND WITH GUST

4.1.4.2 (Continued)

Radial CG Offset	2.0 inches
Thrust Vector Misalignment	0.442 degrees
Normal Force Coefficient	0.20 (1/RAD)
Center of Pressure Offset	61.0 inches
Control System Gains	10.0%

The scatter was combined (additive) so as to induce drift toward the launch tower.

The attitude-attitude rate control mode was used for this study. This control mode actually contributes to drift. The combined effects of the wind, the scatter parameters, and the control mode made it necessary to use a programmed yaw maneuver to reduce the vehicle drift so that collision with the tower was avoided. The programmed yaw maneuver was initiated one second after liftoff and was increased at a rate of one degree per second until maximum commanded yaw was reached. Then, beginning at 8 seconds after liftoff, the command was reduced one degree per second until it again became zero. The effect was to tilt the vehicle away from the tower and into the disturbances, thus reducing drift toward the tower.

The study results are shown in Figure 4.1.4.2-2. Trajectory traces are shown for the tip of vehicle fin A, the relative position of which is shown in Figure 4.1.4.2-3, for yaw commands of 0.0, 1.5 and 2.0 degrees. Vehicle drift was not arrested unless a commanded yaw of at least 1.5° was used. In summary, with scatter parameters acting in the most adverse direction, and with a 95 percentile wind plus gust existing at launch time, a pre-programmed yaw command of at least 1.5 degrees is required to insure a collision-free launch of the INT-20 baseline vehicle.

4.1.4.3 Flexible Body Controls

A flexible body, point-time analysis of control frequency response was made for the INT-20 baseline configuration. The purpose was to determine the task involved in designing compensator networks for the INT-20 vehicle compared with that experienced with a typical Saturn V (Reference 4.1.4.3-1). No severe compensator design problems were evident and it appears that compensator network design for the INT-20 will be less difficult than it was for the Saturn V. The results are presented in the form of Nyquist and Bode plots. Since the INT-20 frequency responses are similar to those of the Saturn V, no root locus plots were generated. Any great difference in responses between the two vehicles would have necessitated use of root-locus plots for determination of the required number of encirclements of the point at $-1 + j0$ on the Nyquist plots.

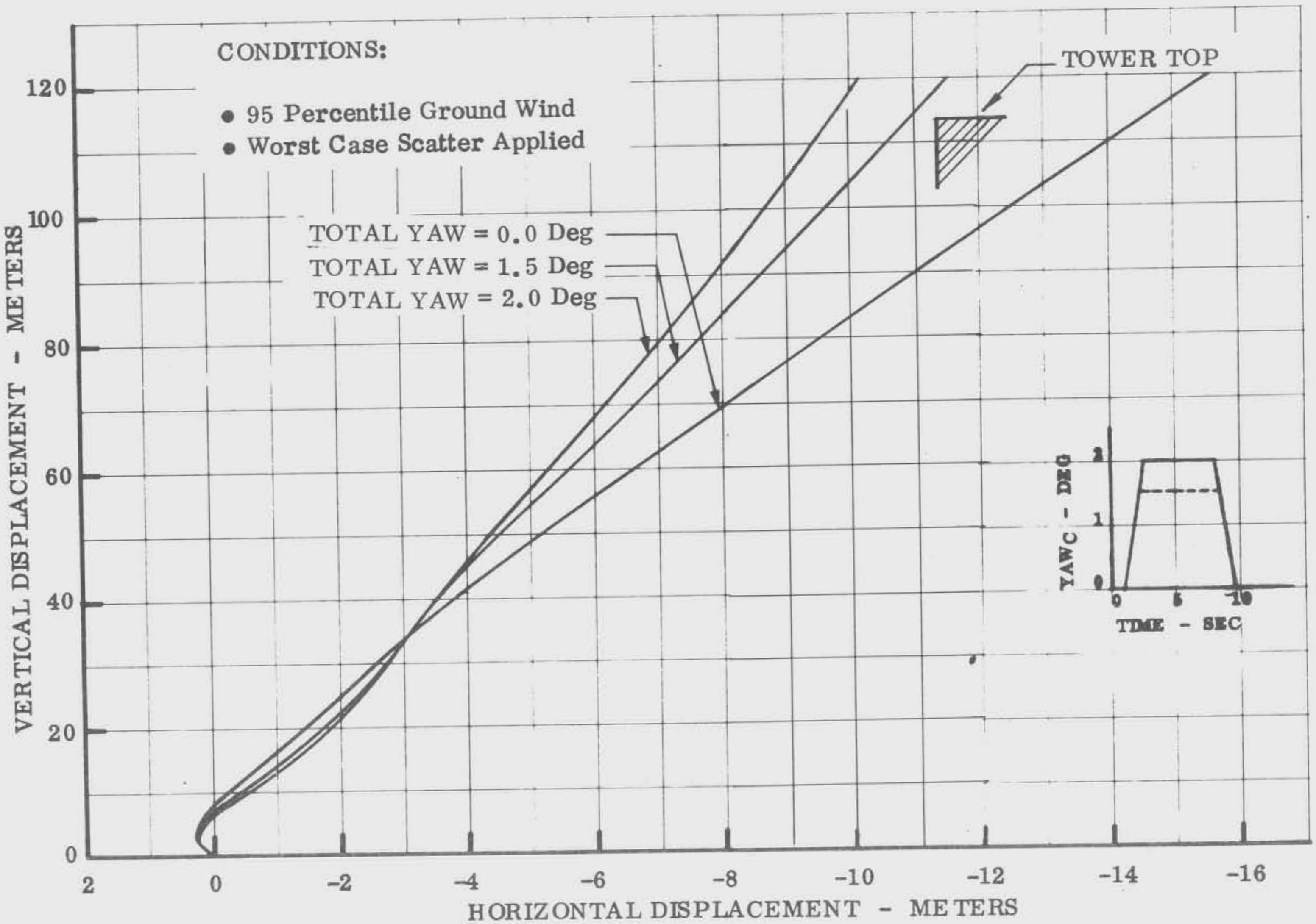


FIGURE 4.1.4.2-2 TOWER CLEARANCE TRAJECTORY FOR FIN TIP A FOR INT-20

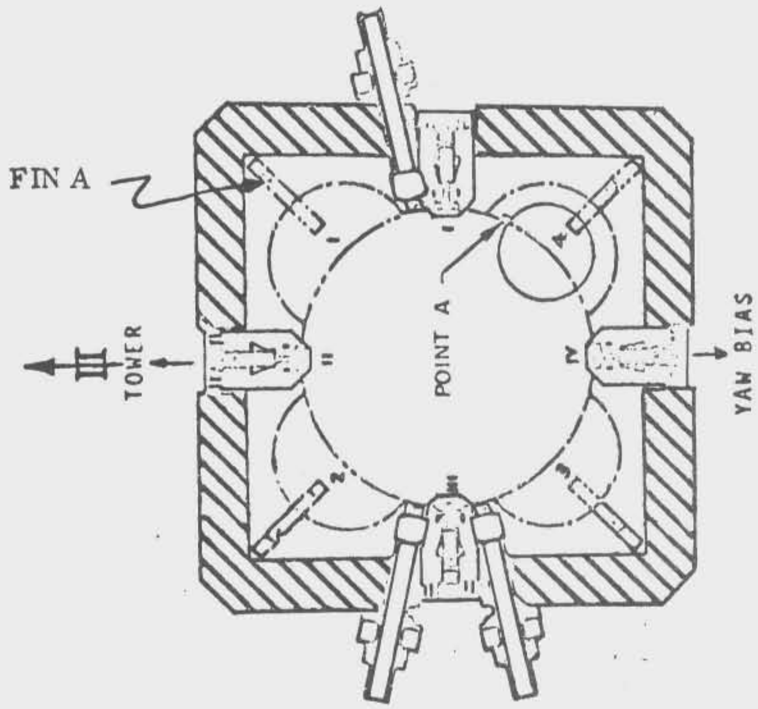


FIGURE 4.1.4.2-3 PLAN VIEW OF LUT SHOWING RELATIVE LOCATION OF FIN A TO TOWER

4.1.4.3 (Continued)

Due to elastic deformation of the vehicle, control feedback elements cannot distinguish the difference between actual vehicle direction changes and body flexing in bending. The bending modes generally have different frequencies from those of the control system itself so that phase and attenuation compensation may be used to allow for the bending effects in the control system. Nyquist and Bode plots were used in this analysis to show attitude error and attitude rate feedback. and combined attitude error - attitude rate feedback. The bode plots show variation of feedback magnitude and phase with frequency. The Nyquist plots show phase angles and frequency in polar form and are used to indicate stability margin magnitudes.

a. Saturn V Design Criteria

The design criteria for the INT-20 vehicle are not the same as those for the Saturn V, but the Saturn V criteria form a good basis for comparison. Typical Saturn V design criteria are shown for the q max condition in Table 4.1.4.3-I.

TABLE 4.1.4.3-I TYPICAL SATURN V DESIGN CRITERIA

PARAMETER	NOMINAL VALUE	3σ
Minimum Aerodynamic Gain Margin	6 db	3 db
Rigid Body Gain Margin	-6 db	3 db
Rigid Body Phase Margin	30 Deg	15 deg
Slosh Peak	0 db	0 db
Phase Stabilized Bending Phase Margins	45 Deg	20 deg
Gain Stabilized Bending Gain Margins	-12 db	3 db

4.1.4.3 (Continued)

Typical uncompensated and compensated combined feedback Nyquist plots for a Saturn V vehicle are shown in Figures 4.1.4.3-1 and 4.1.4.3-2. Compensator design is based upon the characteristics of the uncompensated system. For example, refering to Figure 4.1.4.3-1, some of the compensated stability margins shown in Table 4.1.4.3-I, are found as follows:

1. Aerodynamic Gain Margin

   This margin is the distance from the unit circle to the trace at the point where the trace crosses the 180 degree ray at the lowest frequency, measured along the 180 degree ray.

2. Rigid Body Phase Margin

   This consists of the angle between the 180 degree ray and the point at which the trace first crosses the unit circle from outside to inside. Measurement is made counterclockwise from the 180 degree ray.

3. First Bending Phase Margin

   This consists of the angle between the 180 degree ray and the point at which the trace first crosses the unit circle from inside to outside, measurement being made clockwise from the 180 degree ray. This description and those above are applicable to compensated and, generally, to uncompensated systems.

   Comparison between nominal responses (Table 4.1.4.3-I) and those in Figure 4.1.4.3-1 shows that the aerodynamic gain margin is acceptable since it exceeds the minimum of 6 db (two units on the plot).

   The rigid body phase margin is acceptable; however, the first (lead) bending phase margin must be rotated clockwise approximately 110 degrees, thereby altering the orientation of the plot. This rotation requires a compensating network to satisfy the stability constraints. The compensated response is shown in Figure 4.1.4.3-2.

b. Comparison of Frequency Spectrums

The frequency spectrums for the INT-20 and a typical Saturn V are shown in Figure 4.1.4.3-3. Note that the slosh mode frequencies are approximately the same, but that the bending mode frequencies for INT-20 are successively higher than for Saturn V. The higher bending mode frequencies reduce the difficulty in designing compensators for the INT-20 since the separation between

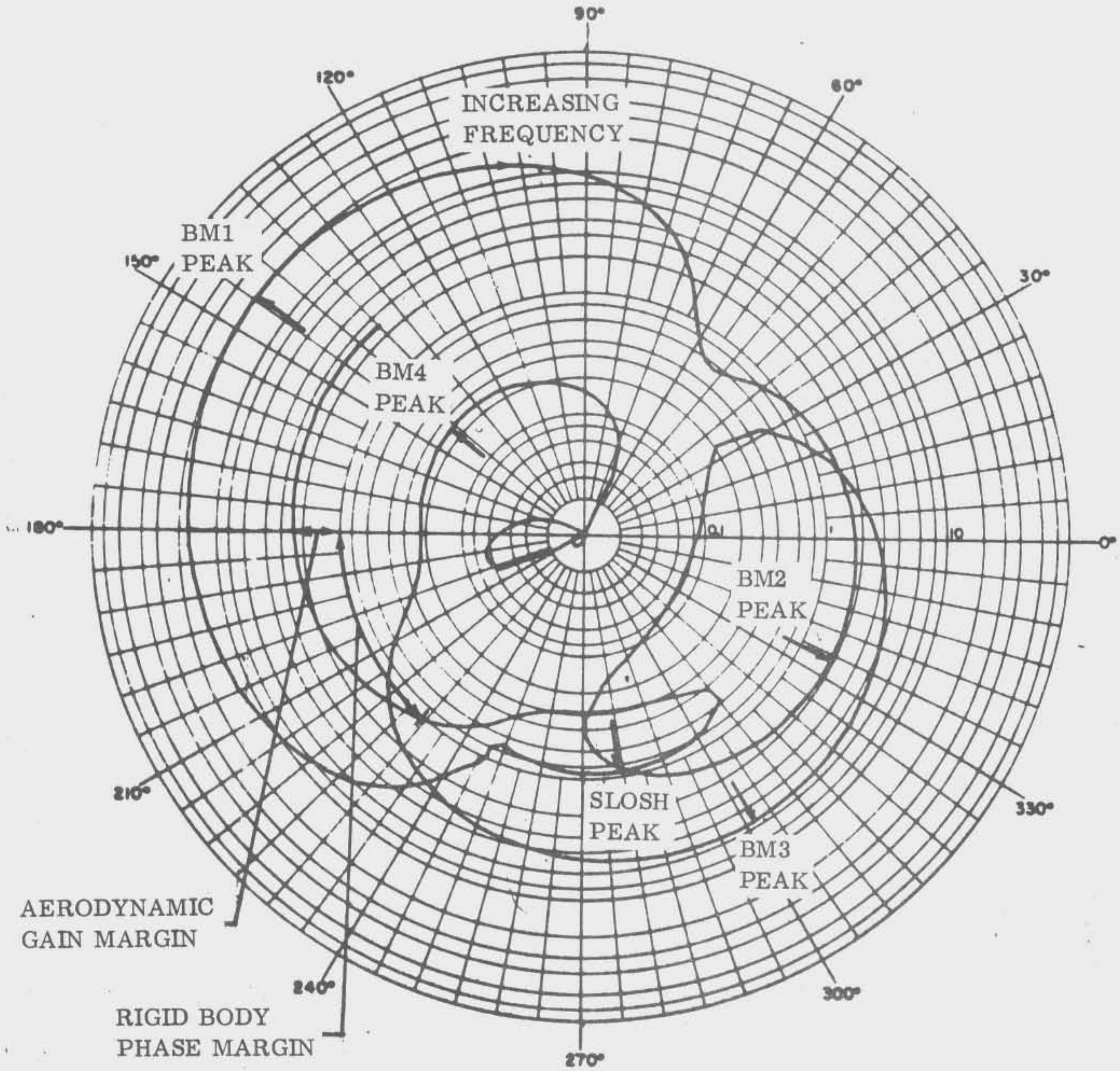


FIGURE 4.1.4.3-1 UNCOMPENSATED COMBINED FEEDBACK NYQUIST PLOT FOR A TYPICAL SATURN V AT q_{MAX}

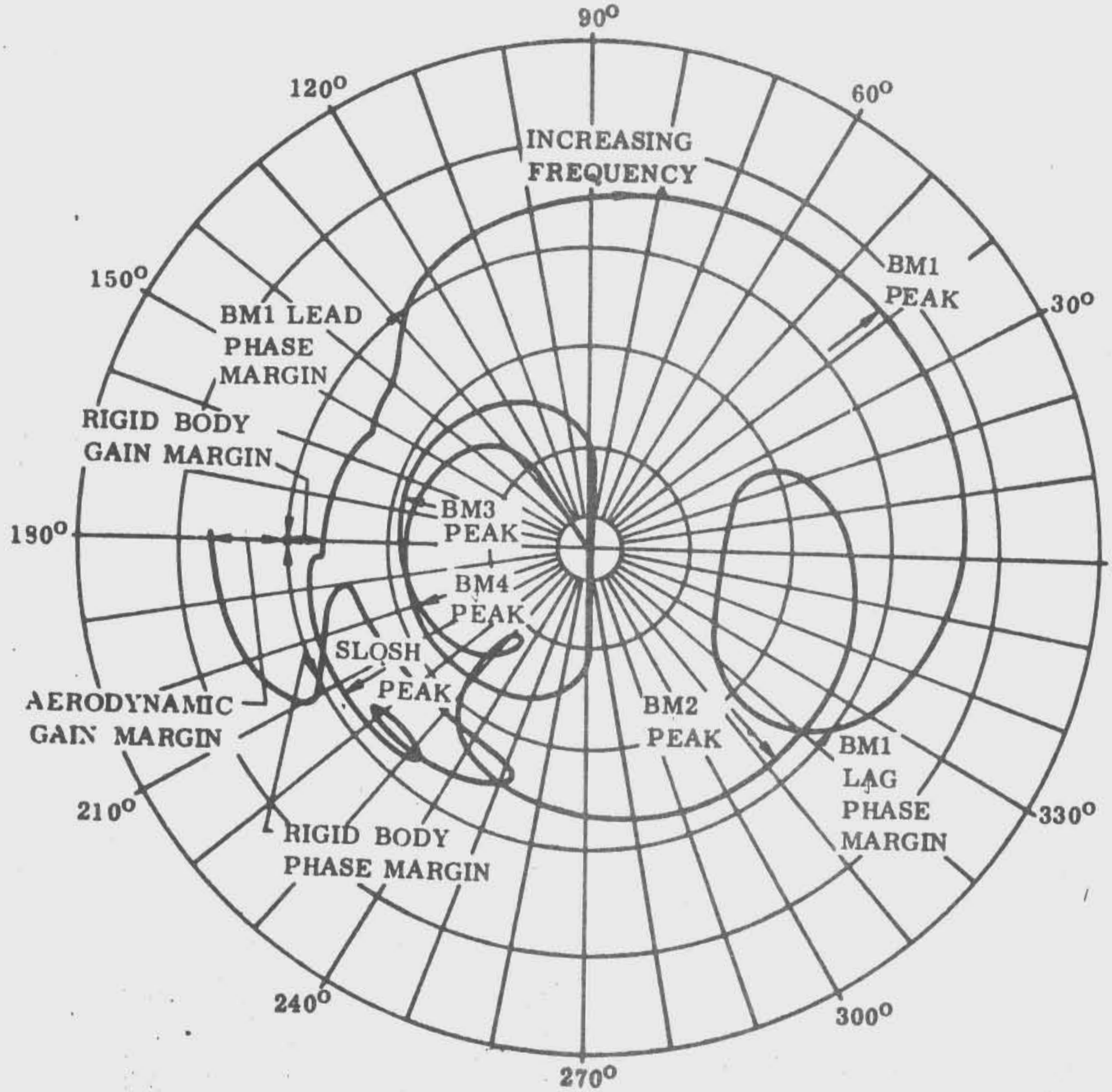
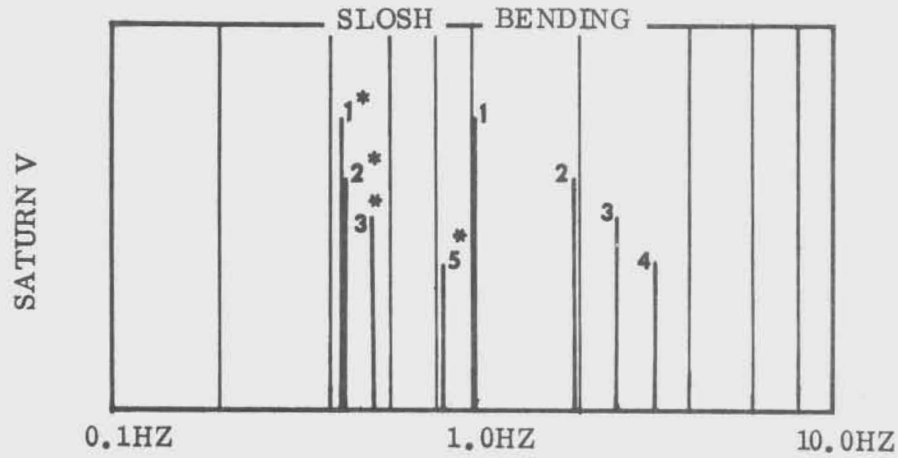


FIGURE 4.1.4.3-2 COMPENSATED COMBINED FEEDBACK NYQUIST PLOT FOR A TYPICAL SATURN V AT q_{MAX}



* Tank numbers, in ascending order up the vehicle

** First Four bending modes.

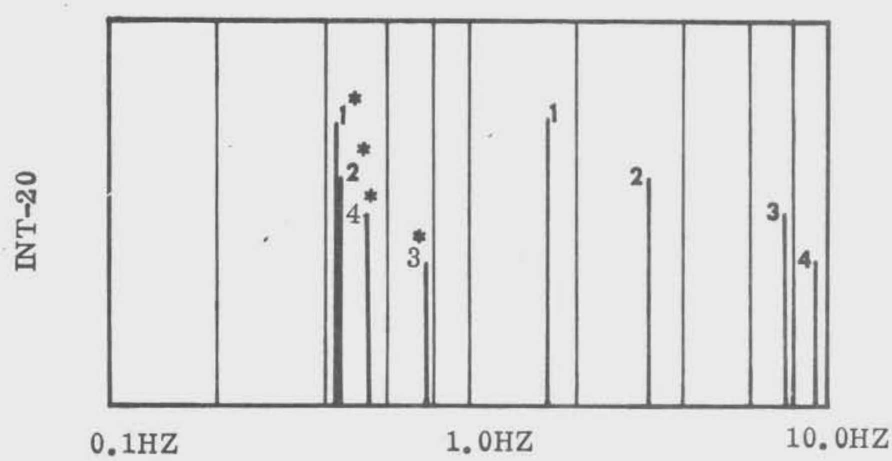


FIGURE 4.1.4.3-3 FREQUENCY SPECTRUM FOR INT-20 AND A TYPICAL SATURN V

4.1.4.3 (Continued)

the bending and slosh rigid body frequencies is larger than for Saturn V. Ideally, a separation in frequencies of one decade between any two incompatible factors is desirable.

c. Control Gains and Frequency

A control frequency of 0.2 Hertz and a damping ratio of 0.7 was chosen for the INT-20. The control frequency is well below the factor of one-fifth the first body bending mode and high enough so that the system is not sluggish and does not couple directly with the slosh disturbances. The damping ratio is sufficient to yield a fast response without excessive overshoot.

The rigid body flight control system gains (paragraph 4.1.4.1) were used to obtain the uncompensated results. These values at q max are: attitude error gain (A_0) = 0.839 Deg/Deg; attitude rate gain (A_1) = 0.764 Deg/Deg/Sec. These gains were reduced to 0.526 Deg/Deg and 0.479 Deg/Deg/Sec, respectively for the partially compensated response of Figure 4.1.4.3-10.

d. Results

In this analysis, the basic attitude - attitude rate feedback control law was used. Sensor characteristics and flight control computer characteristics were not included. The actuator transfer function was taken from reference 4.1.4.3-1. Rigid body, four bending modes and the first slosh mode for each tank were included. All data used corresponds to the flight time point of maximum dynamic pressure (q max).

1. Uncompensated INT-20 System

Uncompensated Nyquist trajectories and Bode plots were obtained for the combined attitude error - attitude rate feedback, attitude error feedback only and attitude rate feedback only. The combined feedback response is compared to a typical Saturn V uncompensated combined feedback response to show that the degree of difficulty which will be encountered during compensator design for the INT-20 is similar to that which was encountered in designing compensators for the Saturn V-class of vehicles. Although the first bending mode frequency for the INT-20 is higher than for Saturn V, it appears that the first slosh mode for the S-IVB LOX tank will cause some problem in compensator design This same problem was present in Saturn V control system compensator design.

4.1.4.3 (Continued)

Nyquist and Bode plots for uncompensated combined feedback are shown in Figures 4.1.4.3-4 and 4.1.4.3-5 for the INT-20 vehicle. A comparison between those design parameters for the INT-20 and the Saturn V is shown in Table 4.1.4.3-II.

TABLE 4.1.4.3-II

CONTROLS PARAMETER COMPARISON

DESIGN PARAMETER	INT-20 (Fig. 4.1.4.3-4)	SAT V (Fig. 4.1.4.3-1)
Aerodynamic Gain Margin (DB)	14.53	13.4
Rigid Body Phase Margin (Deg)	53.34	50.0
Slosh Peak (DB)	9.15 @ 283 Deg.	-0.7 @ 279 Deg.
First Bending Peak (DB)	23.97@ 142.6 Deg.	26 @ 144 Deg.

All of these parameters compare closely except for the slosh peak. A more refined analysis may show that the slosh disturbance is not so severe as it appears from these results.

Nyquist and Bode plots for the uncompensated error feedback are shown in Figures 4.1.4.3-6 and 4.1.4.3-7 and uncompensated attitude rate feedback is shown in Figures 4.1.4.3-8 and 4.1.4.3-9. Note the similarity between the attitude error and combined results plots at low frequencies and the attitude rate and combined feedback plots at higher frequencies. These facts are useful in that compensation of the attitude error feedback compensates the low frequency portion of combined feedback and compensation of the attitude rate feedback compensates the higher frequency part of the combined feedback. Obviously, there is interaction between the two, but generally this procedure will yield successful results in compensating the combined feedback system.

This is shown to be true by the partially compensated combined feedback system shown in Figure 4.1.4.3-10.

2. Partially Compensated INT-20 System

Based on the uncompensated results (Figure 4.1.4.3-4) the following transfer function was used as a trial compensator in the rate feedback loop:

4.1.4.3 (Continued)

$$F_{\dot{\phi}} = \left(\frac{1}{1 + 0.15923S}\right)^2$$

where $S = j\omega$

The results of using this function are shown in Figure 4.1.4.3-10. Comparison of the response shown with that in Figure 4.1.4.3-2 and the design criteria in Table 4.1.4.3-I shows that all of the design criteria were met except for the slosh peak, which although improved by the partial compensation, should be less than 0.0 db. Note that the response shown in Figure 4.1.4.3-10 is similar to that in Figure C.1-3, Appendix C, although the latter represents the digital control system and compensator function described in paragraph 4.3.4.

The filter transfer function shown above will have more terms after the sensor and onboard computer characteristics have been included and the slosh peak criterion has been satisfied. Also, an attitude error filter (F_{ϕ_ϵ}) probably will be required. However, it appears that the compensator for the INT-20 will be less complicated than that presently used for the Saturn V.

e. Conclusions

No difficult compensator design problems were evident; in fact, it appears that compensator design for the INT-20 should be less complicated than it was for the Saturn V class of vehicles.

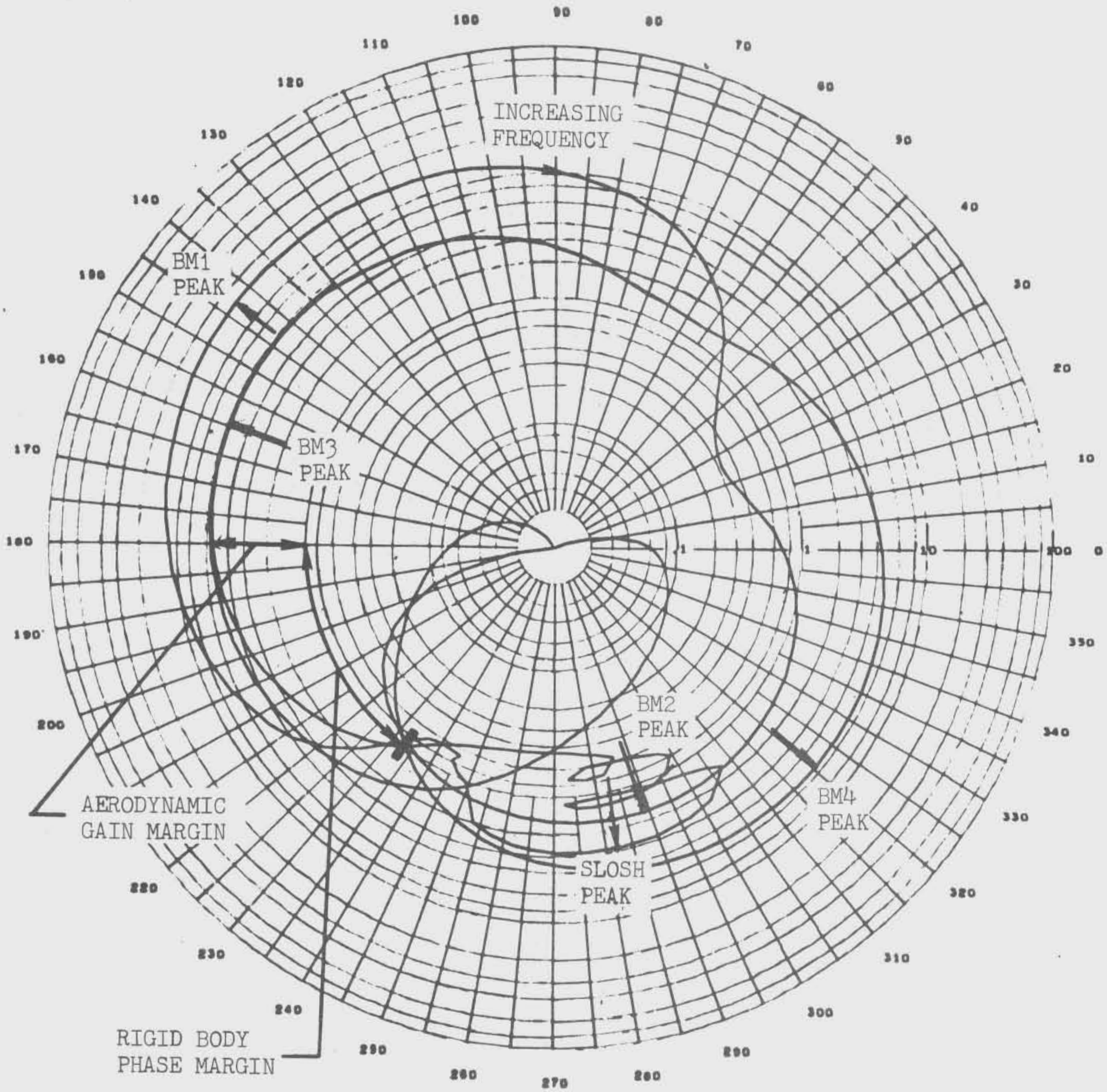


FIGURE 4.1.4.3-4 UNCOMPENSATED COMBINED FEEDBACK NYQUIST PLOT FOR INT-20 AT q_{MAX}

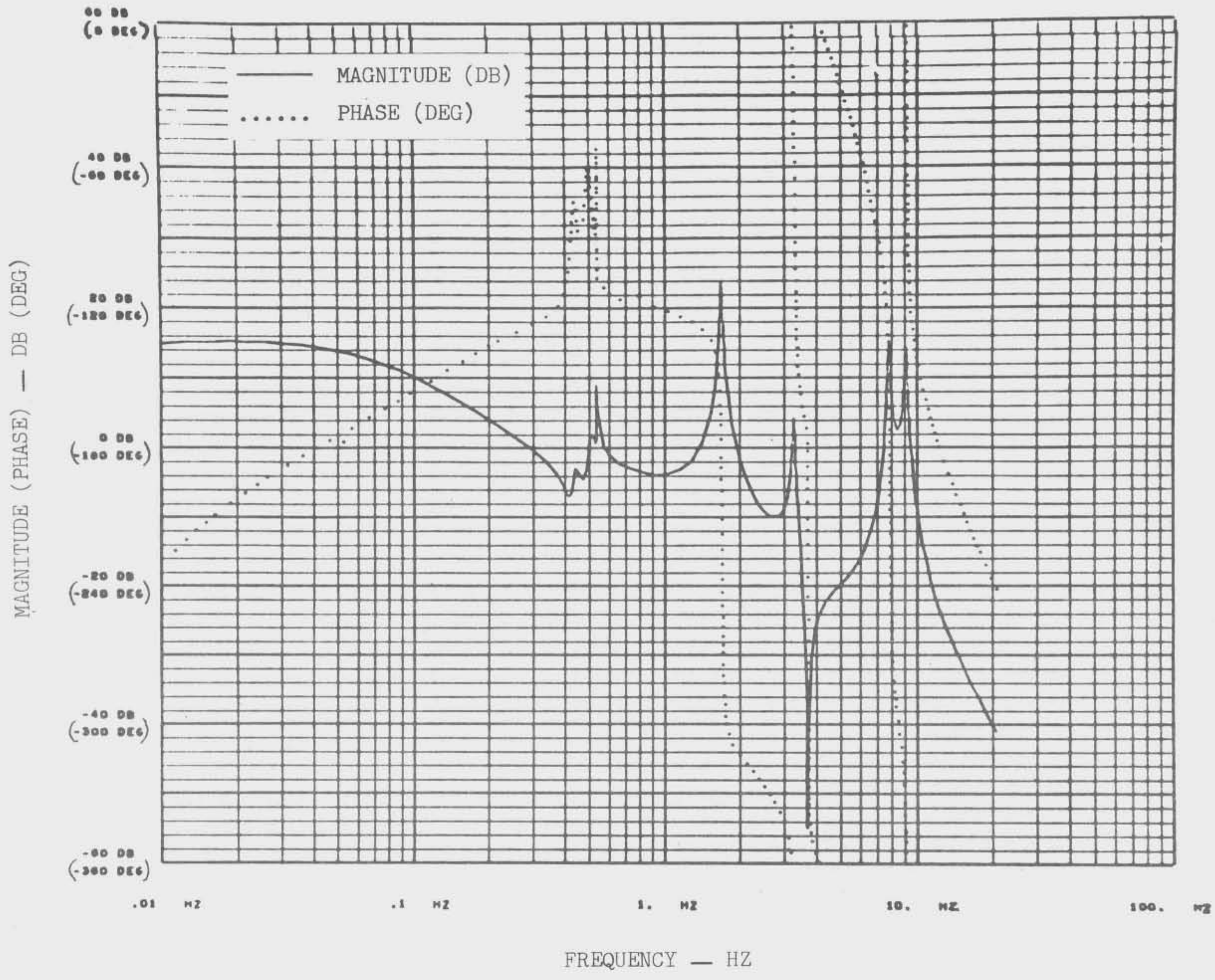


FIGURE 4.1.4.3-5 UNCOMPENSATED COMBINED FEEDBACK BODE PLOT FOR INT-20 AT q_{MAX}

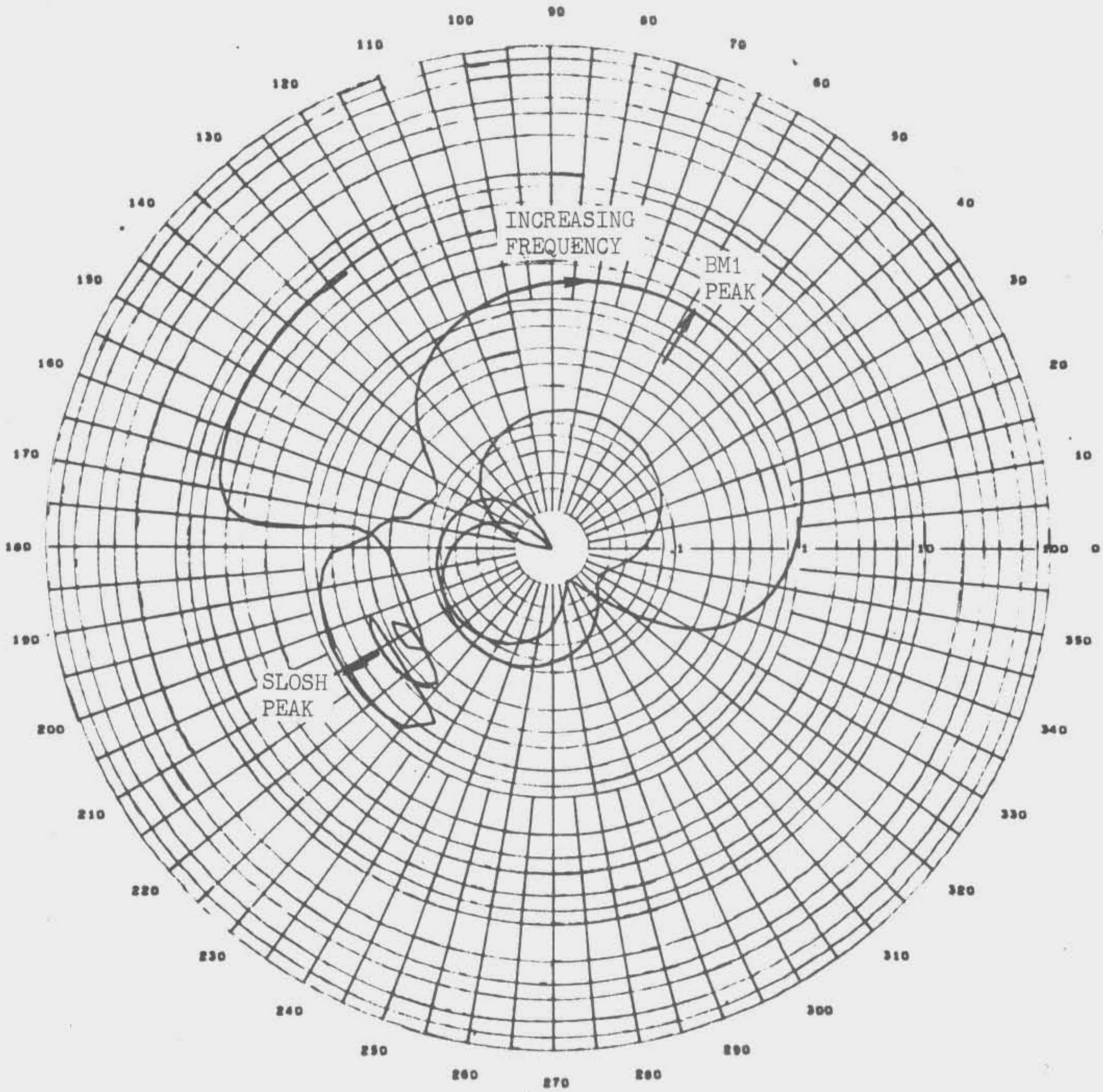


FIGURE 4.1.4.3-6 UNCOMPENSATED ATTITUDE ERROR FEEDBACK NYQUIST PLOT FOR INT-20 AT q_{MAX}

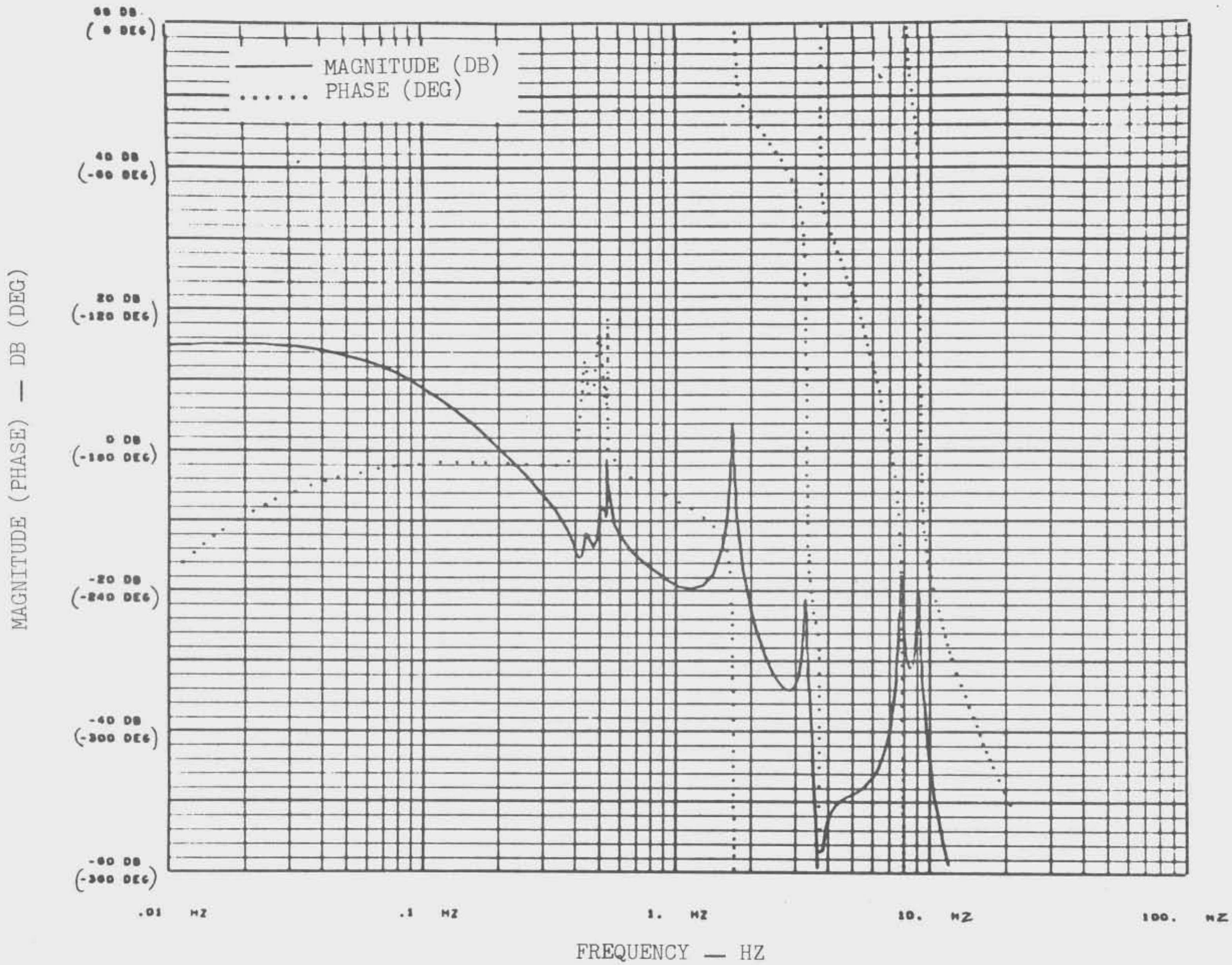


FIGURE 4.1.4.3-7 UNCOMPENSATED ATTITUDE ERROR FEEDBACK BODE PLOT FOR INT-20 AT q_{MAX}

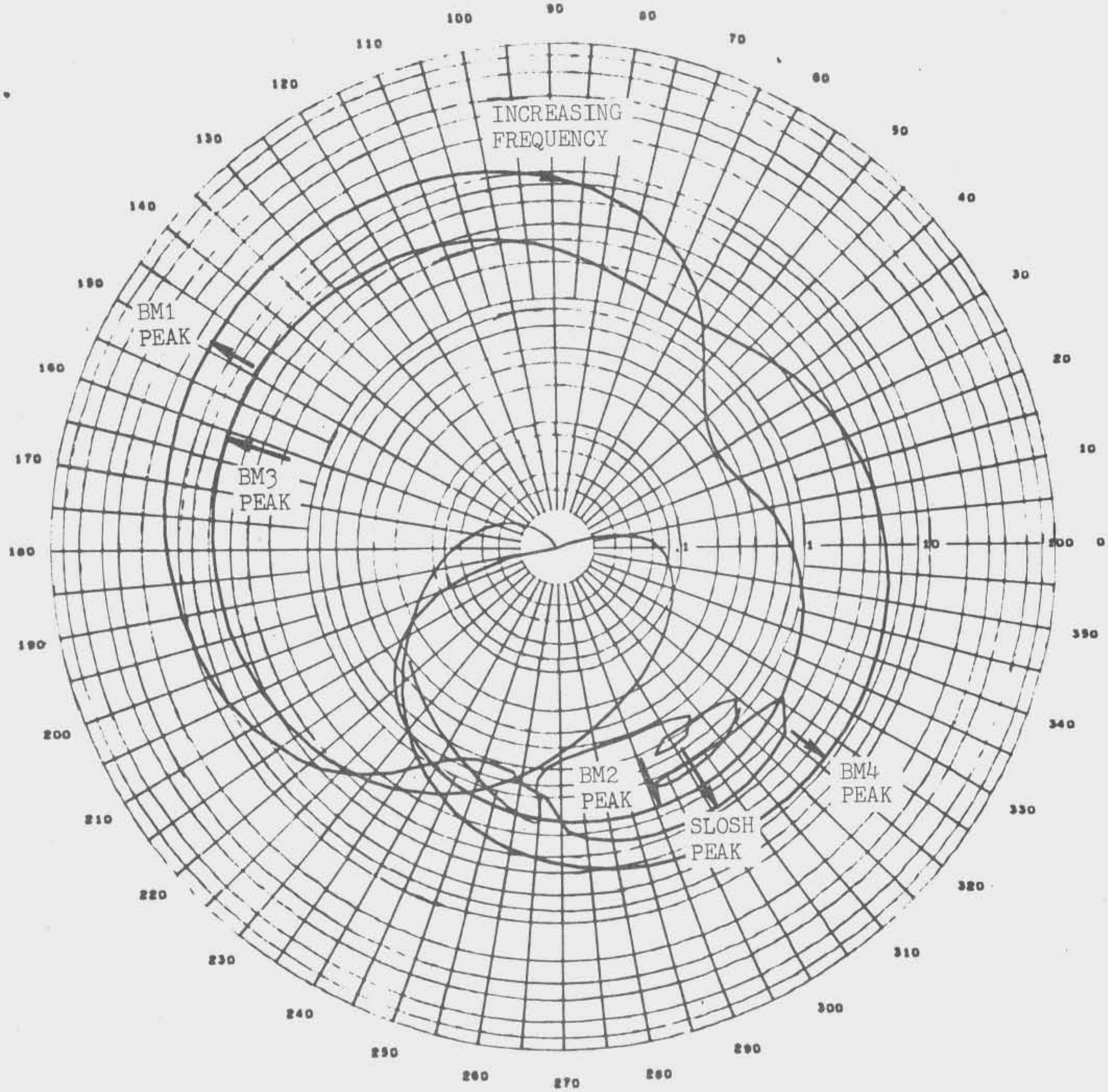


FIGURE 4.1.4.3-8 UNCOMPENSATED ATTITUDE RATE FEEDBACK NYQUIST PLOT FOR INT-20 AT q_{MAX}

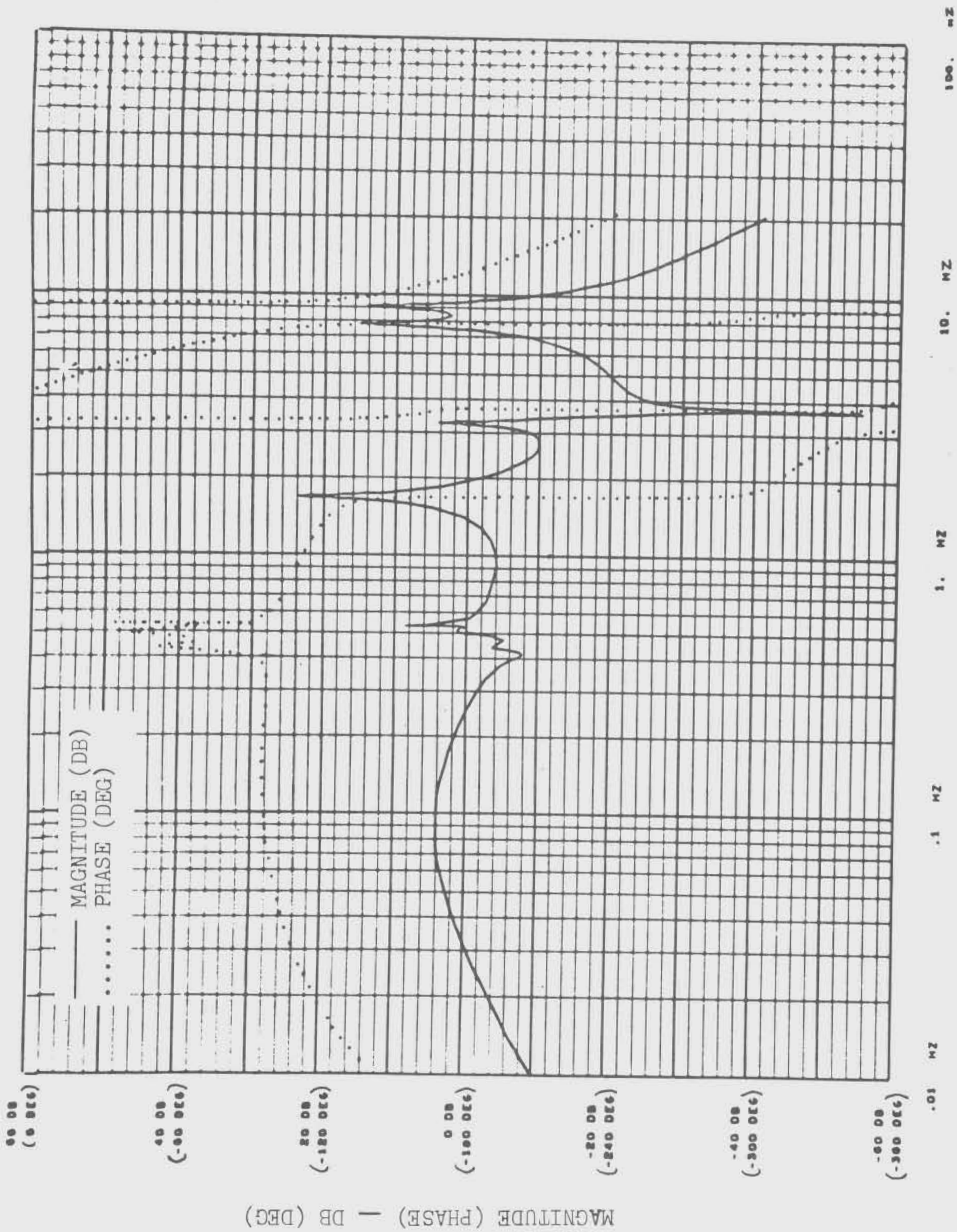


FIGURE 4.1.4.3-9 UNCOMPENSATED ATTITUDE RATE FEEDBACK BODE PLOT FOR INT-20 AT q_{MAX}

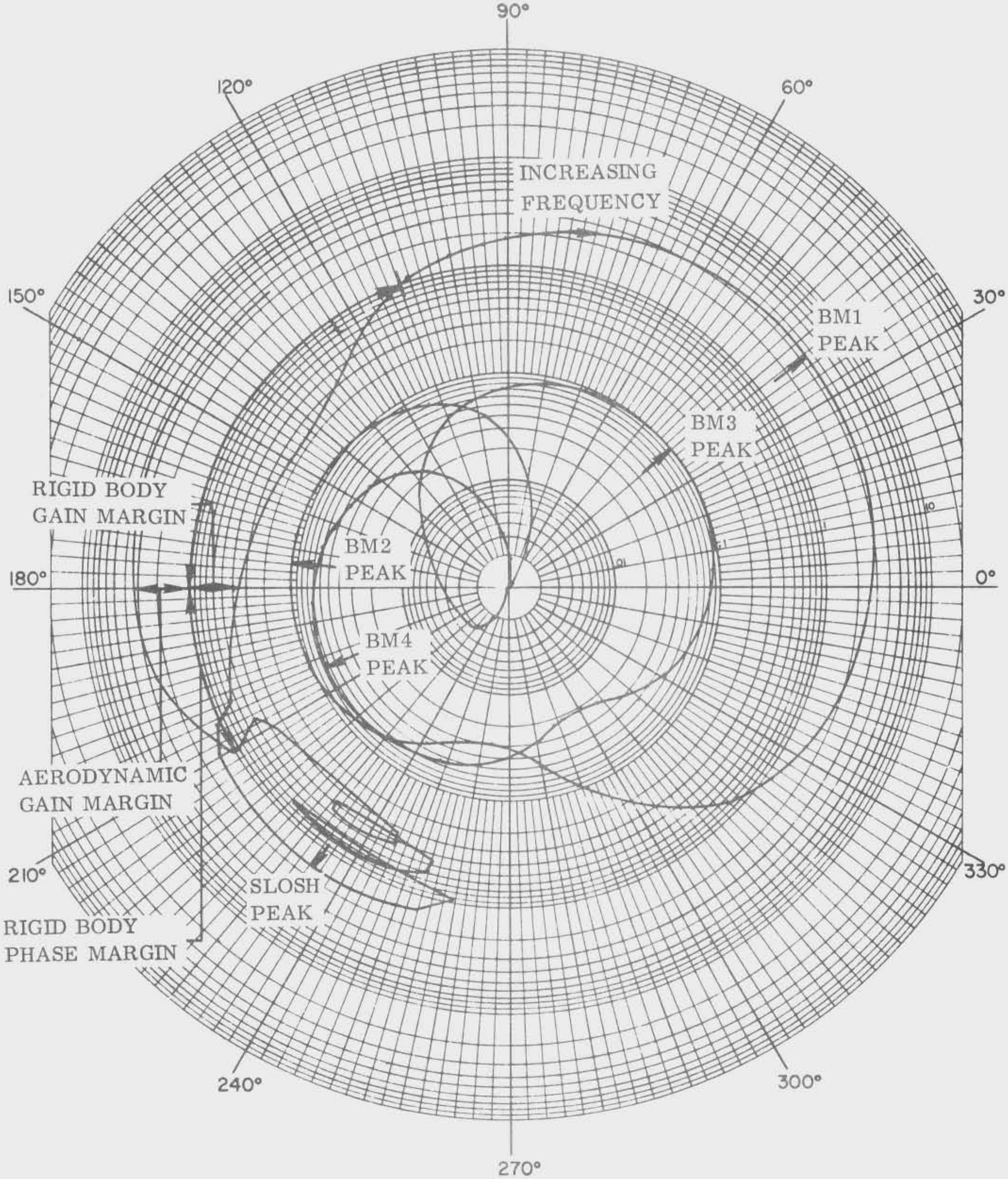


FIGURE 4.1.4.3-10 PARTIALLY COMPENSATED COMBINED FEEDBACK NYQUIST PLOT FOR INT-20 AT q_{MAX}

4.1.4.4 S-IC/S-IVB Separation Clearance and Control

The subsequent paragraphs describe the results of a two dimensional (pitch plane, positions I-III) separation and controllability analysis performed to determine the feasibility of a safe S-IC/S-IVB Staging maneuver for the assumed post separation sequence of events. The parameters describing the vehicle's mass and thrust characteristics, in conjunction with the aerodynamic force coefficients, were statistically selected by the Monte Carlo random sampling method.

At the start of each simulation random parameters which follow their probability distribution were generated. These parameters were kept constant for one complete simulation, and then were regenerated. One hundred simulations were run per each case considered. The averages and standard deviations quoted are based on these hundred simulations.

The study concluded that both nominally and with statistical deviations in all parameters, a safe staging maneuver could be deployed with the assumed post separation sequence of events. The same conclusion was drawn for the case of one retrorocket inoperable. Details follow.

a. System Description

Eight retrorockets are used in staging the S-IC, located in pairs in each of the four engine fairings. They are deployed in order to nullify the tailoff thrust of the two sustaining F-1 booster engines and to provide a rapid axial separation. The position and orientation of a typical retrorocket is shown on Figure 4.1.4.4-1. The specification thrust time history of a single retro motor is given in Figure 4.1.4.4-2. Nominal retro thrust is assumed to correspond to a solid propellant grain temperature of 70 degrees Fahrenheit, with $\pm 3\sigma$ band ranging from 30 to 120 degrees Fahrenheit. Standard perturbation of the retrorocket cant angle during the motor's thrust mode was assumed as 0.1 degree.

The estimated F-1 engine nominal thrust decay at altitude and 3-sigma limits about the nominal are shown on Figure 4.1.4.4-3.

A pressure field caused by each retrorocket thrust plume is assumed to apply a resultant impingement force of a magnitude equal to five percent of the retro thrust. Vectorially, the force is assumed to be acting normal to the stage surface approximately two feet above the retro nozzle exit plane.

The assumed pertinent sequencing data following Time Base 3 (TB3 is initiated by booster engines 1 and 3 cutoff command) is presented on Table 4.1.4.4-I. This sequence of events is also illustrated on Figure 4.1.4.4-4. For this vehicle configuration a two second J-2 fuel lead was assumed.

Table 4.1.4.4-II gives the numerical values of the pertinent parameters used in the study. As indicated in the Table, all parameters, with the

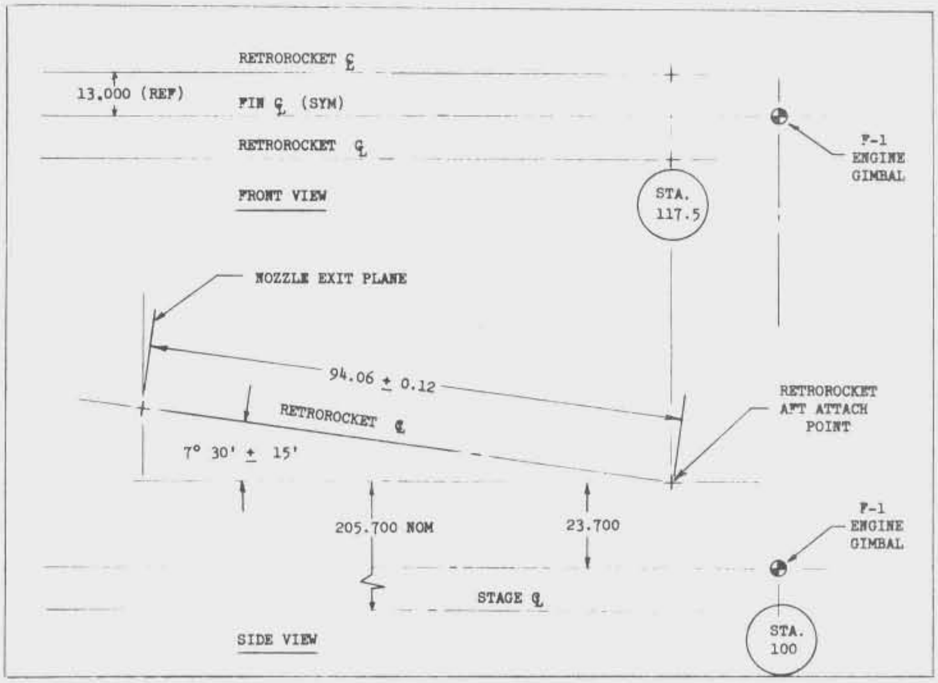


Figure 4.1.4.4-1. Critical Retrorocket Geometry

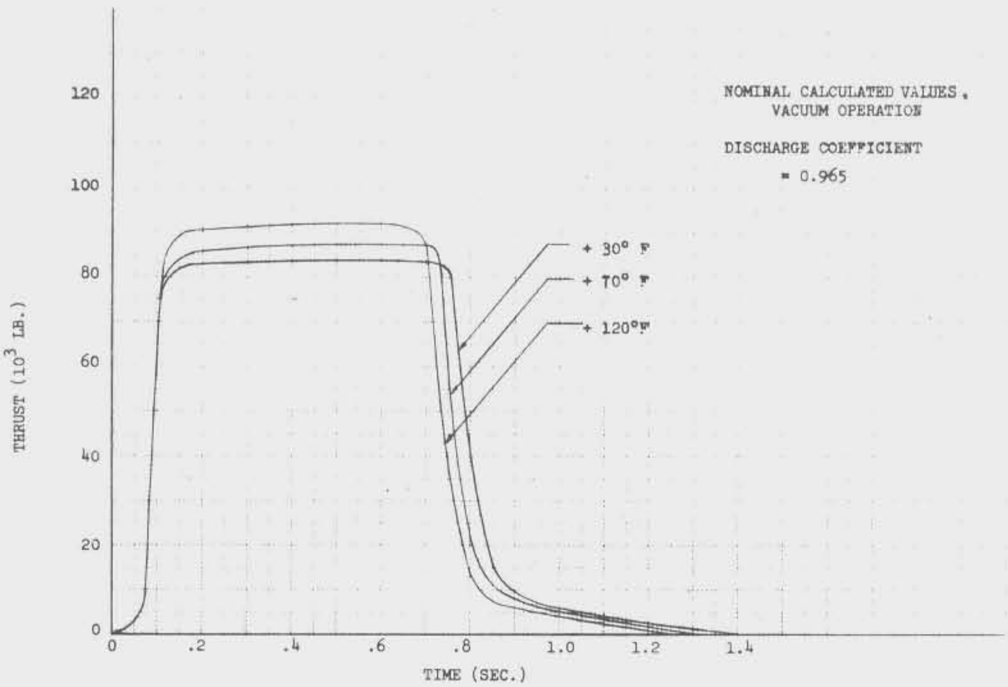


Figure 4.1.4.4-2. S-IC Retromotor (TE-424) Thrust History

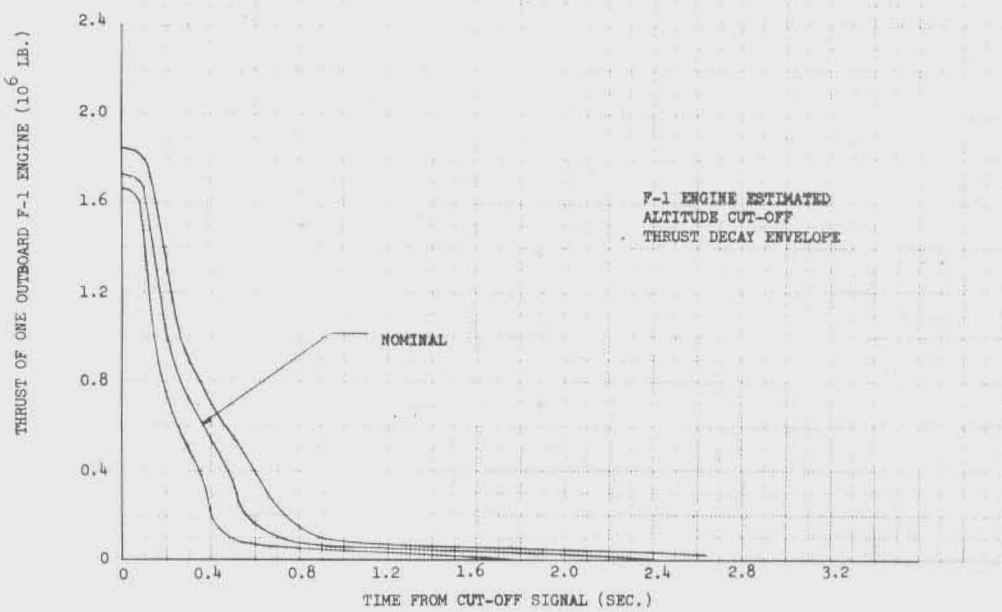


Figure 4.1.4.4-3. Estimated F-1 Engine Thrust Decay

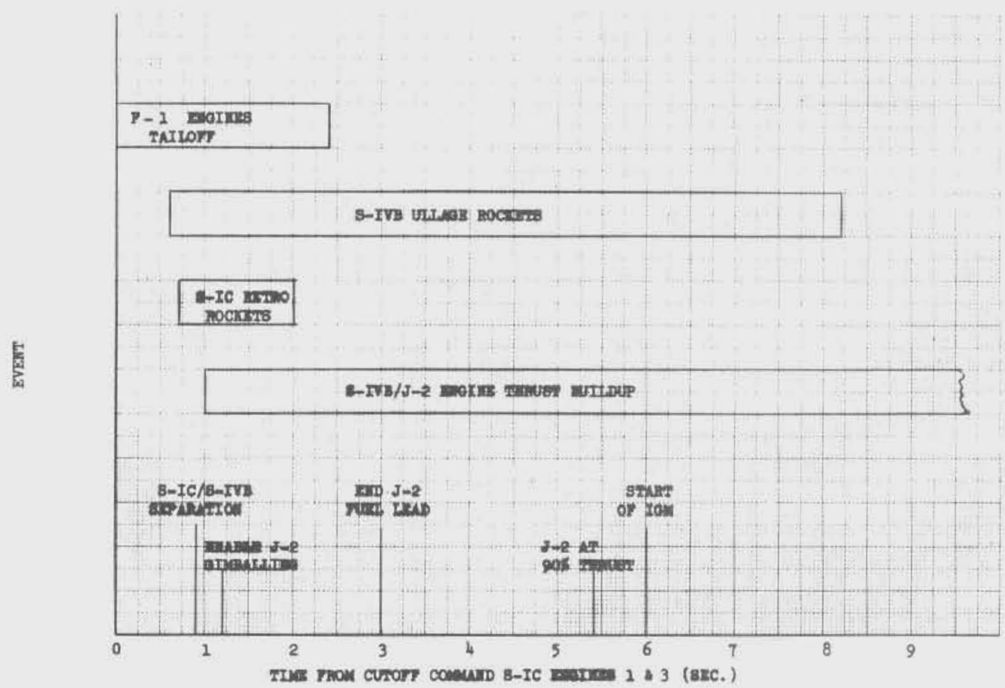


Figure 4.1.4.4-4. INT-20 (S-IC/S-IVB) Separation Sequence of Events

TABLE 4.1.4.4-I

INT-20 Flight Sequence of Events

Nominal Flight Time (sec)	Command	Time From Base (sec)
146.0	S-IC Engines 2 and 4 Cutoff - Start Time Base No. 2 (TB2)	TB2 + 0.0
211.0	S-IC Engines 1 and 3 Cutoff - Start Time Base No. 3 (TB3)	TB3 + 0.0
211.6	S-IVB Ullage Rockets Ignition Command	TB3 + 0.6
211.7	Signal to Separation Devices and S-IC Retrorockets	TB3 + 0.7
211.9	S-IC/S-IVB Separation	TB3 + 0.9
212.0	S-IVB Engine Start Sequence	TB3 + 1.0
212.2	Enable J-2 Engine Gimballing	TB3 + 1.2
214.0	End Fuel Lead	TB3 + 3.0
216.4	J-2 at 90 Percent Thrust	TB3 + 5.4
217.0	Start of IGM	TB3 + 6.0

TABLE 4.1.4.4-II (page 1 of 2)

PROBABILITY DISTRIBUTION OF S-IC/S-IVB SEPARATION PARAMETERS

Item	Average Value	Standard Deviation
Normal Distribution		
S-IVB Center of Gravity	24.08 ft	0.5 ft
S-IC Center of Gravity	110.85	0.5 ft
S-IVB Weight	393,712 lbm	1%
S-IC Weight	354,867 lbm	1%
S-IVB Moment of Inertia	9,141,482 slug-ft^2	1.66%
S-IC Moment of Inertia	17,893,300 slug-ft^2	1.66%
S-IVB Drag Coefficient, C_D	0.18	6.66%
S-IC Drag Coefficient, C_D	1.84	6.66%
S-IVB Normal Force Coefficient, C_N	0.04	0.024
S-IC Normal Force Coefficient, C_N	0.0272	0.001632
Initial Rotation Rate (each stage considered independent of the other)	0.15 deg/sec	1 deg/sec
Initial Angle of Attack	0.1305	1.33 deg
Booster Engine Misalignment (each engine is considered independent of the other)	0.00	0.5 deg
Retro-Rocket Misalignment (each rocket is considered independent of the other)	0.00	0.10 deg

TABLE 4.1.4.4-II (page 2 of 2)

Item	Average Value	C Factor
Beta Distribution		
Booster Engine Multiplication Factor (each engine is considered independent of the other	1.0	0.5
Retro Rocket Multiplication Factor (each engine is considered independent of the other)	1.0	0.263758

exception of the booster engine and retrorocket thrust multipliers, have a normal probability frequency distribution. The latter two have a beta distribution detailed by Figure 4.1.4.4-5.

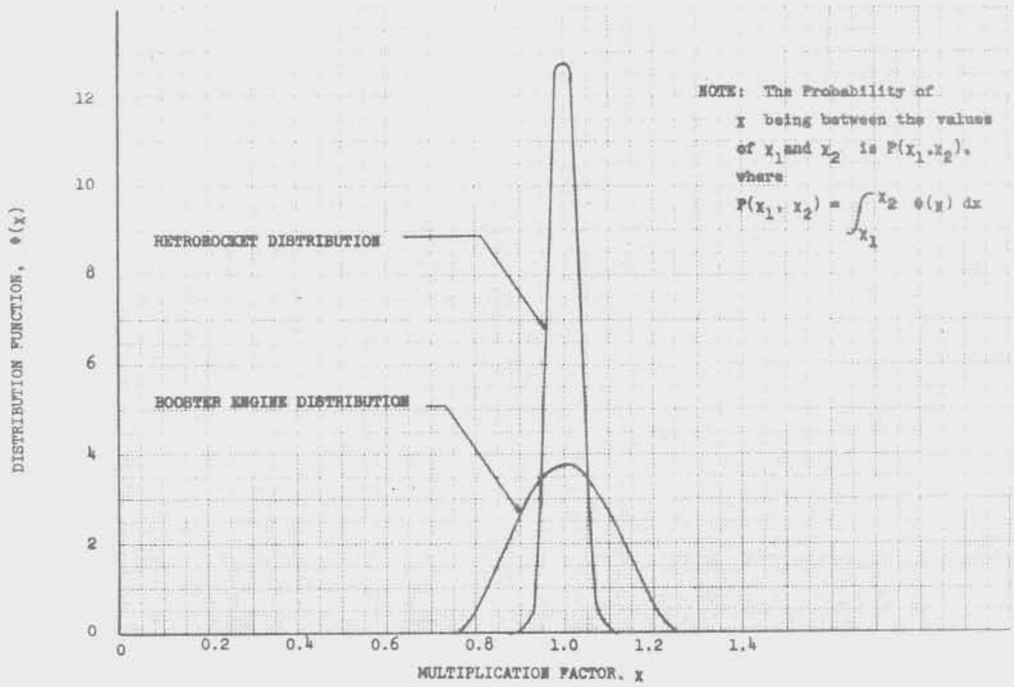


Figure 4.1.4.4-5. Retrorocket and Booster Engine Multiplication Factor Distribution

b. Separation Clearance

Figure 4.1.4.4-6 shows the displacement history profile of the J-2 engine bell for a nominally operating system, along with a standard deviation ($\pm\sigma$) envelope. In all simulations the engine bell surpassed the interstage upper periphery in an average time of 0.940 seconds following separation with a standard perturbation equal to 0.023 seconds. At time of separation the radial distance between the interstage centerline and the bell's aft periphery is 3.33 feet. After completing the simulation with randomly picked parameters the data showed that the average maximal radial distance between the J-2 aft periphery (point T_1) and the interstage centerline was 3.761 feet with a standard deviation of 0.315 feet. Figure 4.1.4.4-7 indicates that the maximum radial displacement traversed by the engine bell prior to clearing the interstage was not in excess of approximately 1.5 feet, corresponding to a 99.9 percent probability of no re-contact.

The case of one retrorocket inoperable represents a more critical condition. This is illustrated on Figure 4.1.4.4-8, which presents the increased lateral translation of the J-2 engine bell caused by the unbalanced retrorocket forces. In all simulations performed the engine bell crossed the interstage upper periphery in an average time of 1.060 seconds after separation with a standard

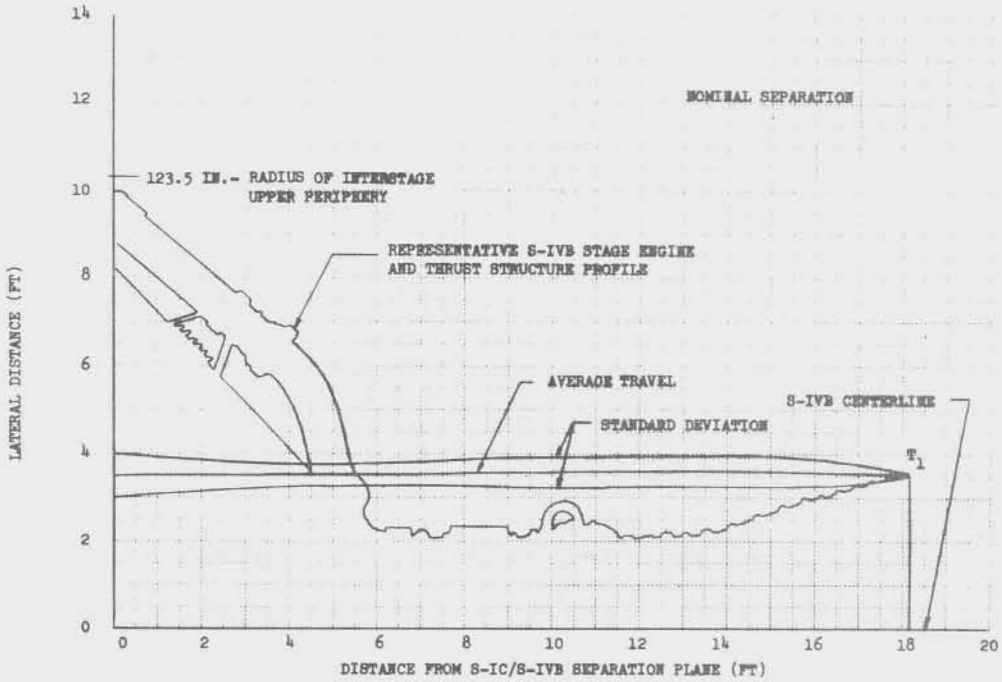


Figure 4.1.4.4-6. INT-20 (S-IC/S-IVB) Separation – Nominal Condition

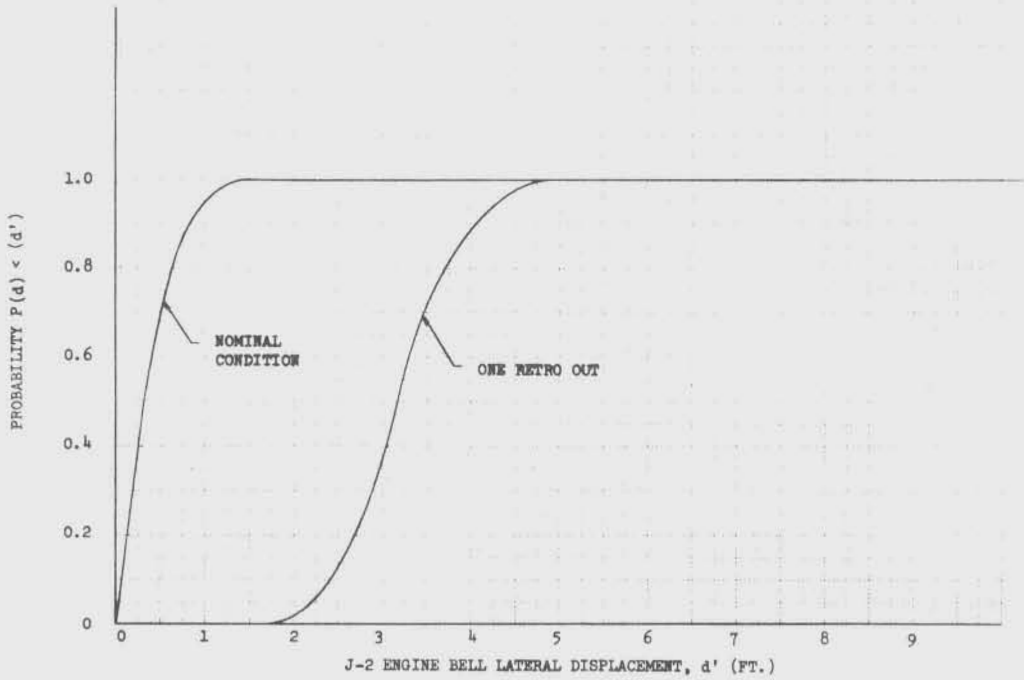


Figure 4.1.4.4-7. Probability of Lateral Displacement not Exceeding a Specified Value

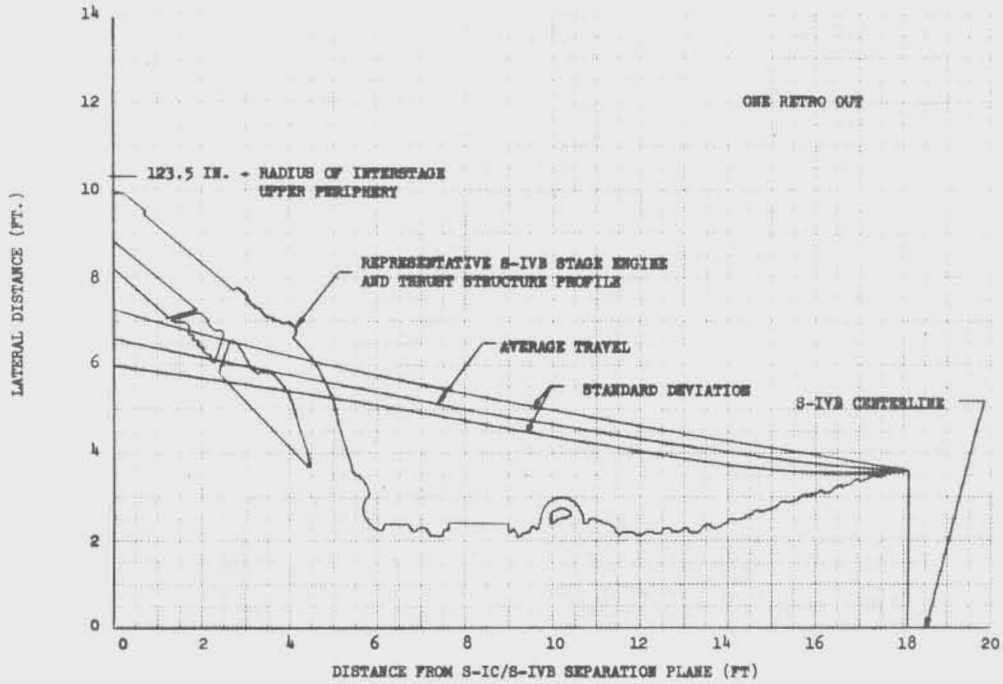


Figure 4.1.4.4-8. INT-20 (S-IC/S-IVB) Separation – One Retro Out

deviation of 0.018 seconds for the 100 simulations carried out. The average maximal radial distance between the J-2 bell aft periphery (point T_1) and the interstage centerline following separation was 6.551 feet with a standard deviation of 0.618 feet. As shown in Figure 4.1.4.4-7, the maximum lateral displacement of the engine bell was not in excess of 4.8 feet, approximately 1.7 feet radially from the interstage upper lip, corresponding to a 99.9 percent probability of no recontact.

c. S-IVB Controllability

An initial attitude error of 1.9 degrees along with an attitude rate of -0.15 degrees per second were assumed for the time of S-IC/S-IVB staging. Damping of the transients is achieved during J-2 engine thrust vector control mode. Figures 4.1.4.4-9 and 4.1.4.4-10 present the time history of the J-2 engine average deflection for the nominal separation case and the case of separation with one retro out, respectively. The curves show that the engine excursions are well within the actuator stop position of 7 degrees.

Despite a coast period of approximately two seconds between separation and mainstage thrust, S-IVB controllability is continuously maintained. The average attitude error and attitude rate histories for the two separation conditions are also shown on Figures 4.1.4.4-9 and 4.1.4.4-10. The probability of not exceeding a specified

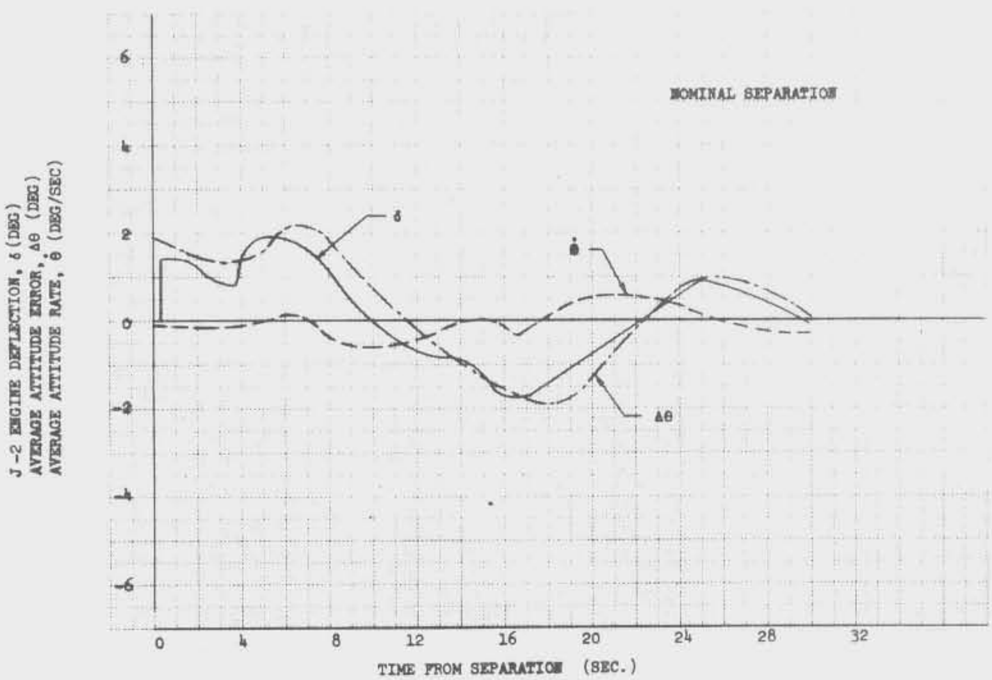


Figure 4.1.4.4-9. S-IVB Transients Following Separation – Nominal Condition

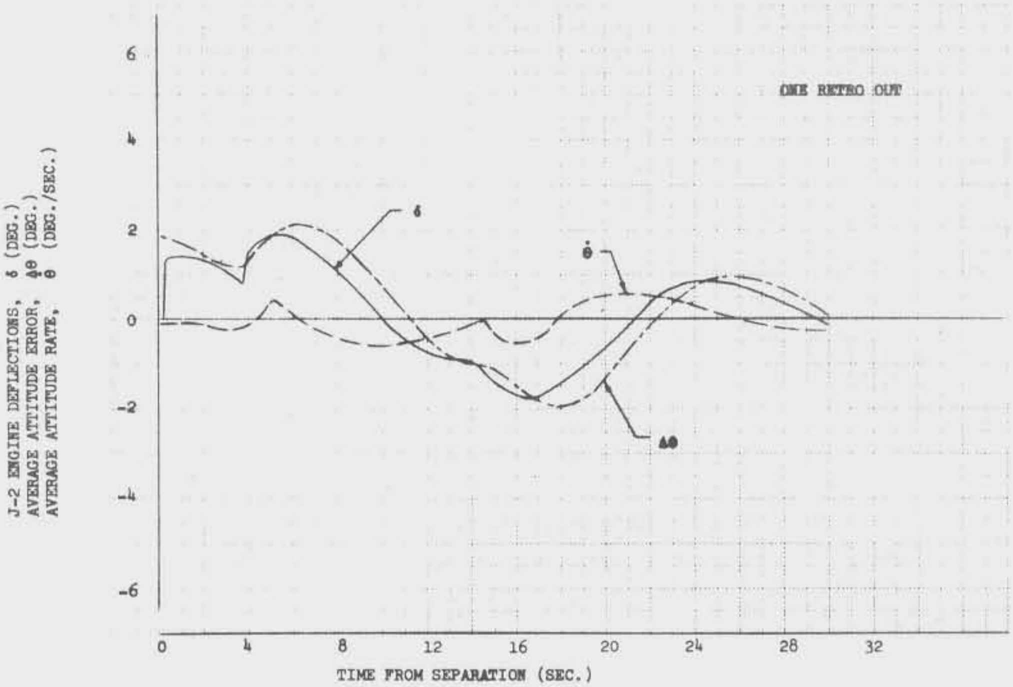


Figure 4.1.4.4-10. S-IVB Transients Following Separation – One Retro Out

maximum attitude error is presented on Figure 4.1.4.4-11. This probability curve applies to both separation conditions and includes a plus three sigma (+3σ) probability of not exceeding a maximum attitude error of 12 degrees.

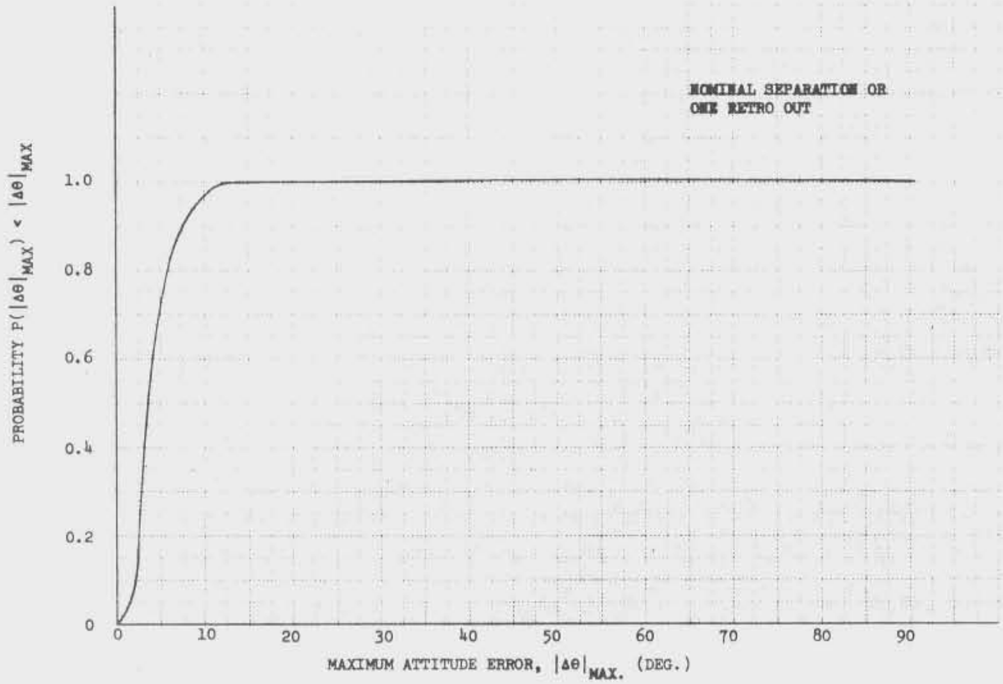


Figure 4.1.4.4-11. Probability of Maximum Attitude Error not Exceeding a Specified Value

4.1.5 Structural Dynamics

Vibration characteristics of the elastic vehicle structure and the dynamic properties associated with propellants sloshing in the tanks must be known in order to develop vehicle control responses and loads. Lateral vibration modes and sloshing propellant masses and frequencies were determined for the INT-20 vehicle, based upon the vehicle mass properties shown in Section 4.1.7.2.

4.1.5.1 Vehicle Vibration Properties

Characteristics of the first four lateral vibration modes were determined for various times of flight as follow: liftoff, maximum ($q\,\alpha$), and cutoff. These data were obtained from a digital computer program, with the vehicle being respresented as a beam with lumped masses having variable flexural and shear stiffness. Modal deflections and slopes of the vibration modes are shown in Figures 4.1.5.1-1 through 4.1.5.1-8 for liftoff, Figures 4.1.5.1-9 through 4.1.5.1-16 for maximum ($q\,\alpha$), and Figures 4.1.5.1-17 through 4.1.5.1-24 for cutoff.

4.1.5.2 Slosh

Slosh parameters were approximated by equivalent mass-spring systems which simulate the forces and moments on the tank caused by the oscillating propellants. Such systems are characterized by one or more asymmetric modes having associated masses (M) and frequencies (ω). Equivalent sloshing masses and frequencies computed for the INT-20 vehicle, using the method of analysis described in Reference 4.1.2-4 are listed in Table 4.1.5.2-I. Responses are included for the first mode only, since the effects of higher modes are generally insignificant. The slosh parameters are shown at four times of flight during S-IC stage burn: liftoff, ($q\,\alpha$) max, (q) max and cutoff. For example, the S-IC LOX tank fluid slosh is represented, at (q) max, by a mass (M) of 1176 lb sec^2/inch with a vibration frequency (ω) of 2.71 radians/second, positioned at Station 946.

TABLE 4.1.5.2-I INT-20 BASELINE VEHICLE PROPELLANT SLOSH PARAMETERS

TANK	TIME SEC.	S1 rad/sec	X_{S1} Sta inches	M_{S1} $Lb.Sec^2/in$	S1 (1)
S-IC LOX	Liftoff (t= 0)	2.11	1186	1163	0.025-0.057
	Max (qα) (t=70.16)	2.58	971	1179	0.025-0.057
	Max (q) (t=79.0)	2.71	946	1176	0.025-0.057
	Cutoff (t=210.95)	-	-	0 (2)	-
S-IC RP-1	Liftoff (t=0)	2.13	618	351	0.023-0.058
	Max (qα) (t=70.16)	2.56	326	797	0.023-0.058
	Max (q) (t=79.0)	2.67	319	793	0.023-0.058
	Cutoff (t-210.95)	-	-	0 (2)	-
S-IVB LOX	Liftoff (t=0)	3.77	1930	36	0.04
	Max (qα) (t=70.16)	4.60	1930	36	0.04
	Max (q) (t=79.0)	4.84	1930	36	0.04
	Cutoff (t=210.95)	7.30	1930	36	0.04
S-IVB LH_2	Liftoff (t=0)	2.61	2080	20	0.001
	Max (qα) (t=70.16)	3.19	2080	20	0.001
	Max (q) (t=79.0)	3.35	2080	20	0.001
	Cutoff (t=210.95)	5.06	2080	20	0.001

NOTES:

(1) Damping ratio. Damping is assumed to be the same as Saturn V. See Reference 4.1.4.1-2

(2) Slosh mass zero at cutoff (Reference 4.1.5.2-3).

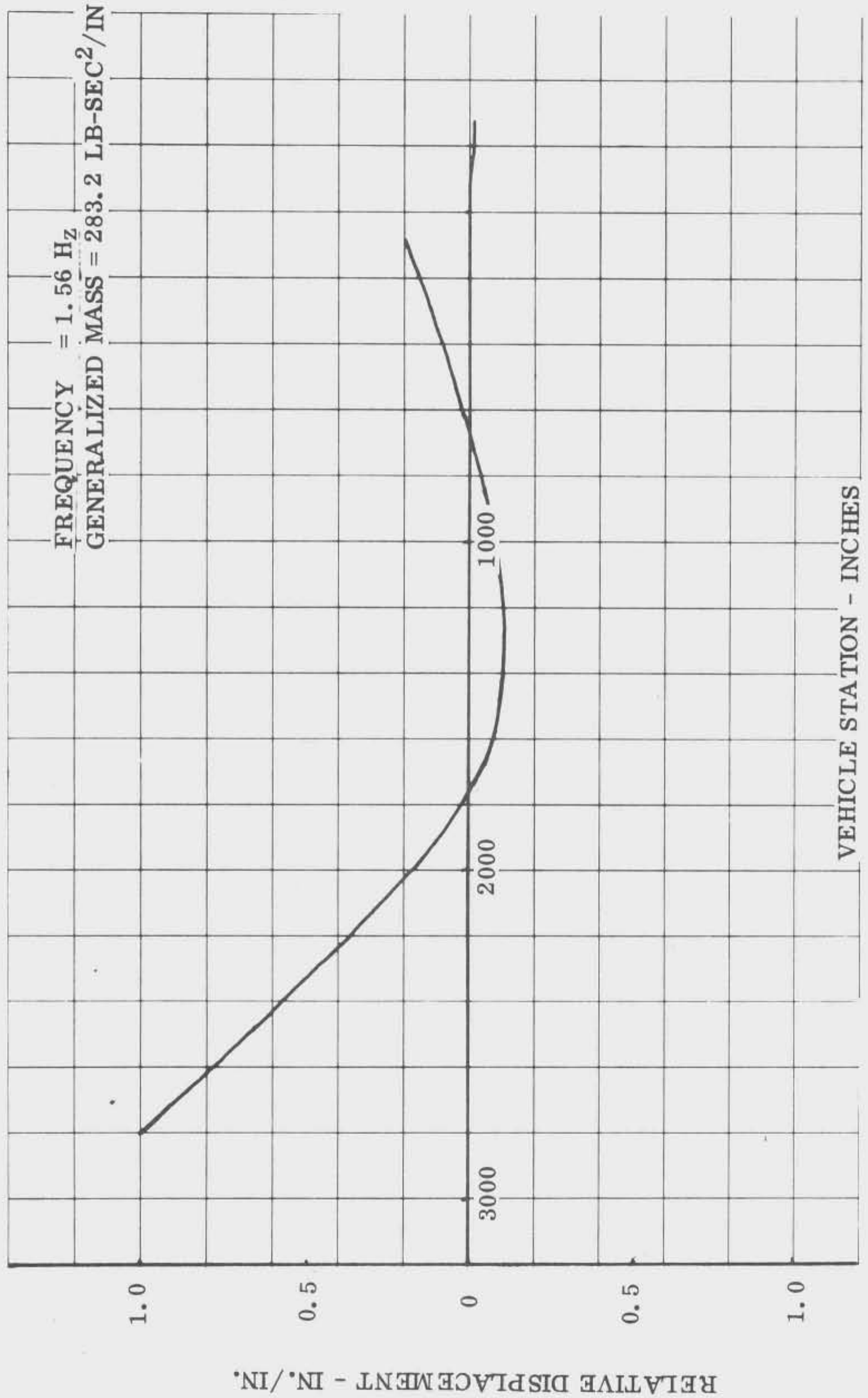


FIGURE 4.1.5.1-1 FIRST FREE-FREE BENDING MODE @ LIFT-OFF

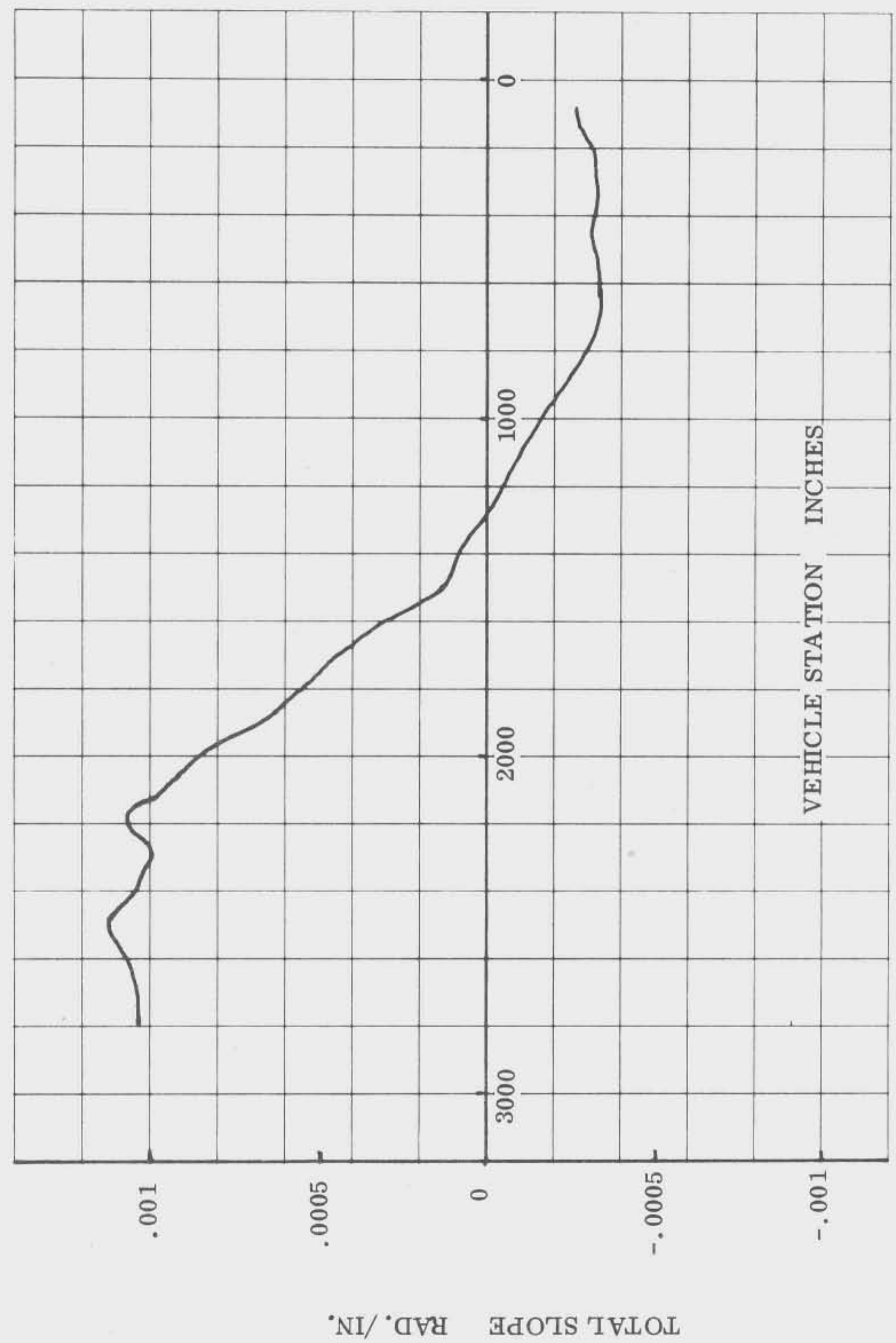


FIGURE 4.1.5.1-2 FIRST BENDING MODE TOTAL SLOPE @ LIFT-OFF

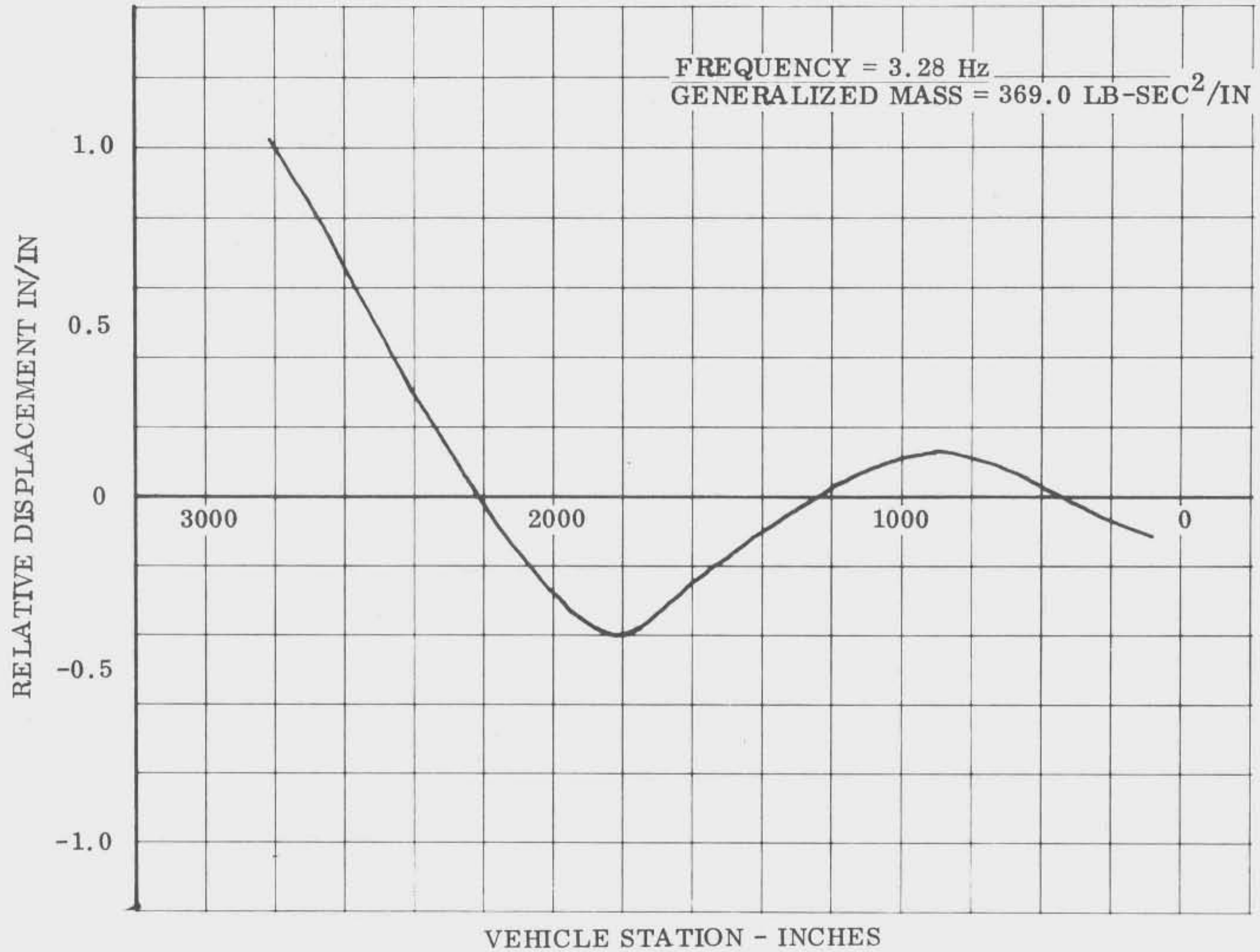


FIGURE 4.1.5.1-3 SECOND FREE-FREE BENDING MODE @ LIFT-OFF

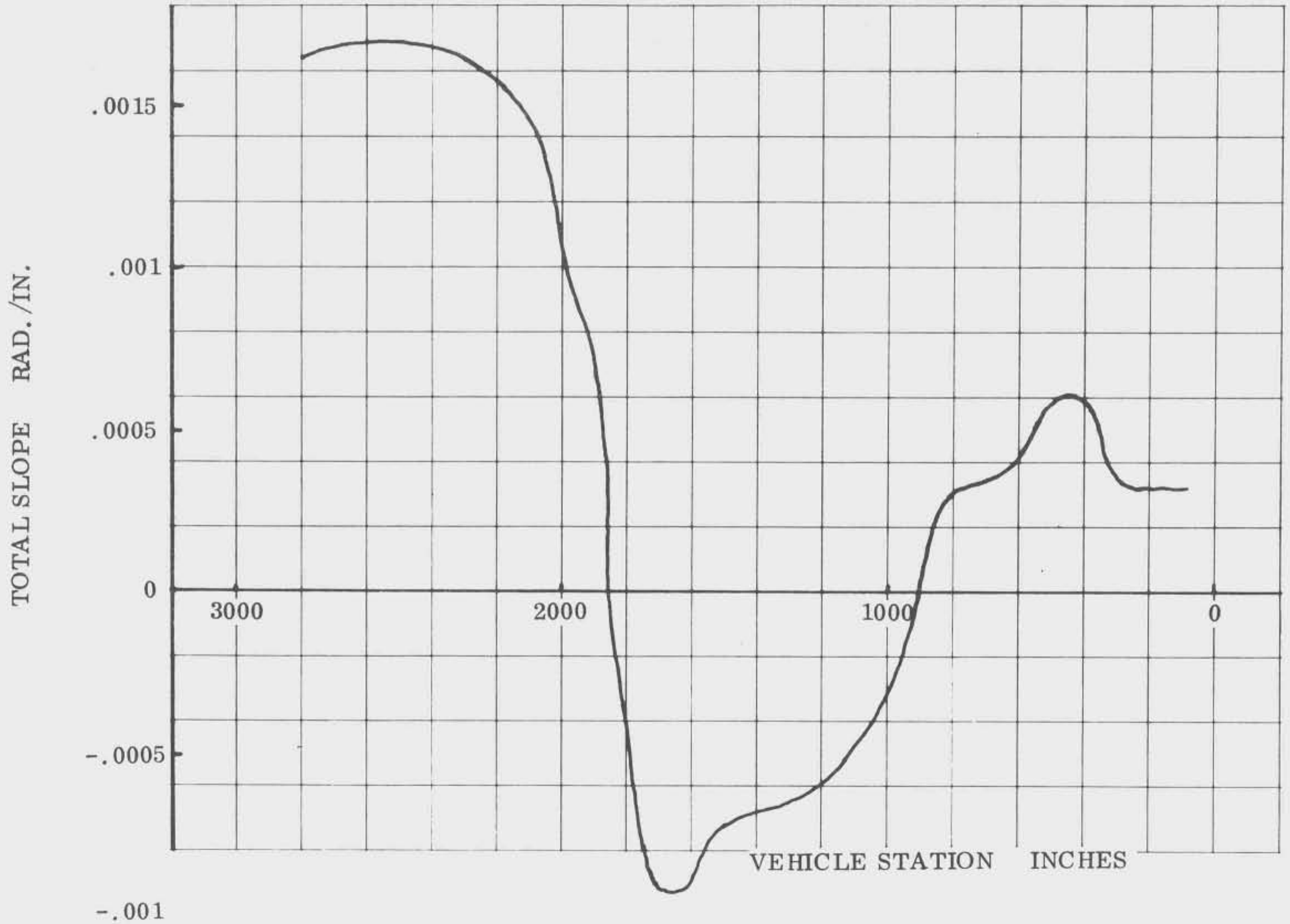


FIGURE 4.1.5.1-4 SECOND BENDING MODE TOTAL SLOPE @ LIFT-OFF

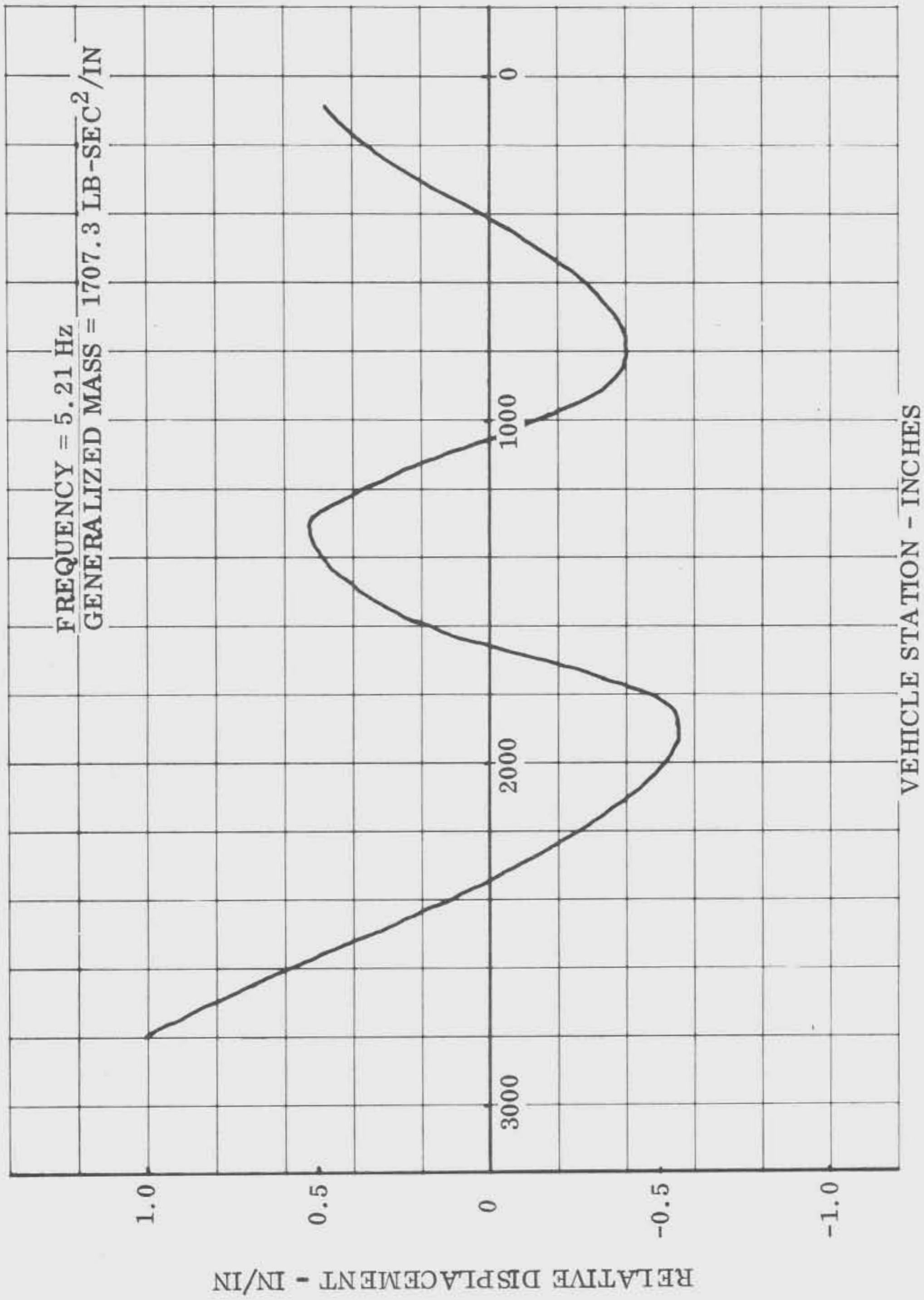


FIGURE 4.1.5.1-5 THIRD FREE-FREE BENDING MODE @ LIFT-OFF

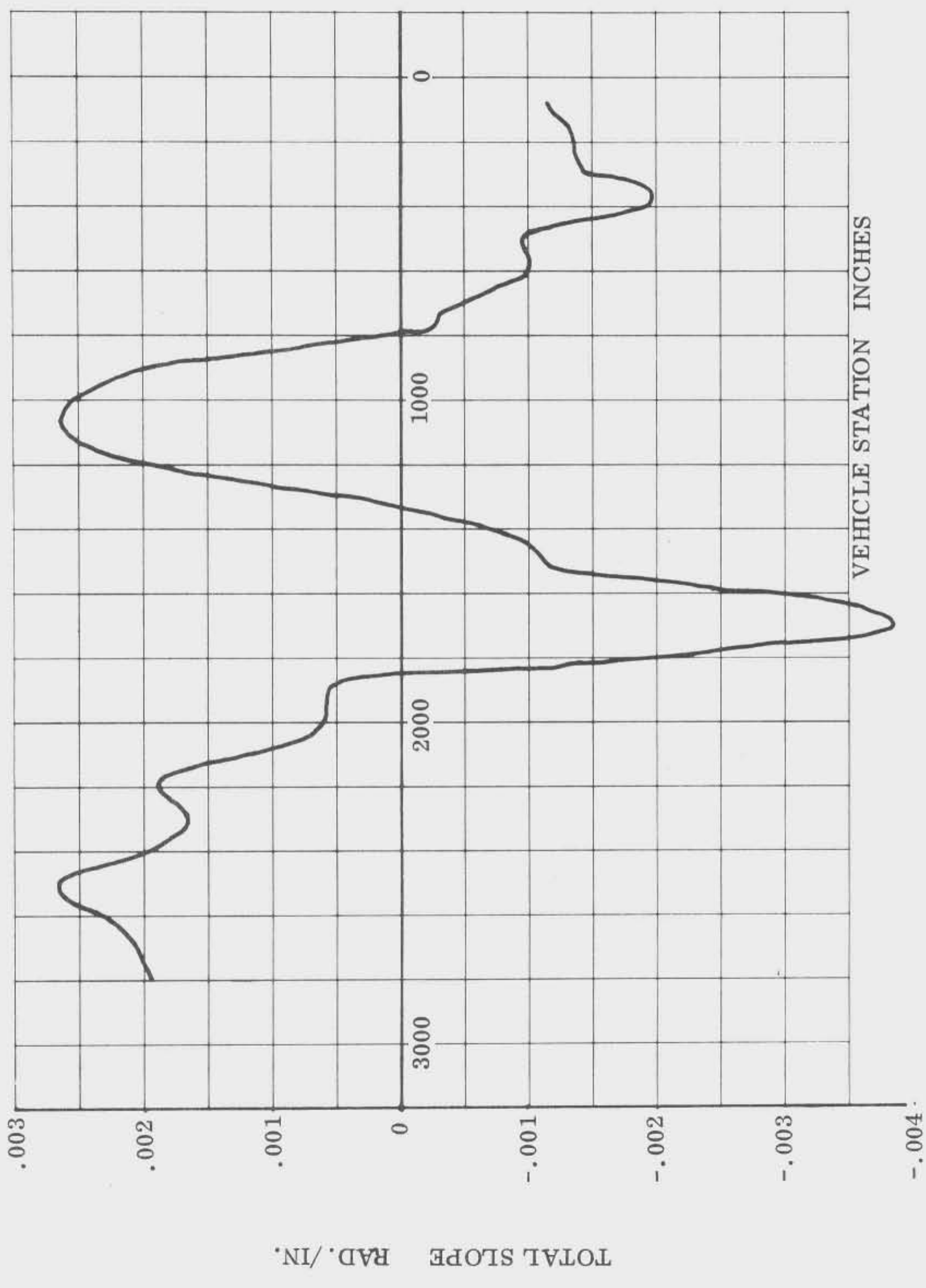


FIGURE 4.1.5.1-6 THIRD BENDING MODE TOTAL SLOPE @ LIFT-OFF

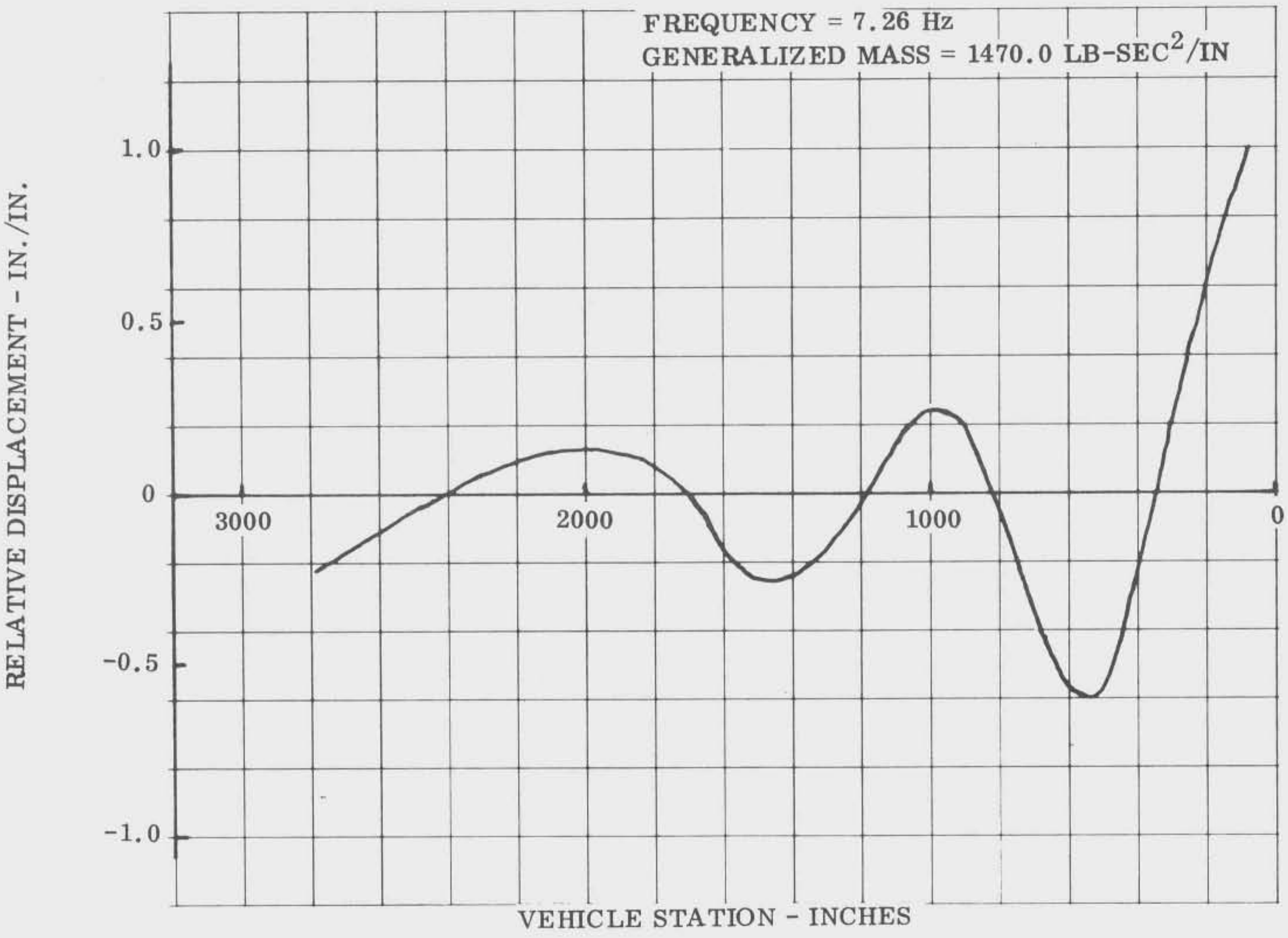


FIGURE 4.1.5.1-7 FOURTH FREE-FREE BENDING MODE @ LIFT-OFF

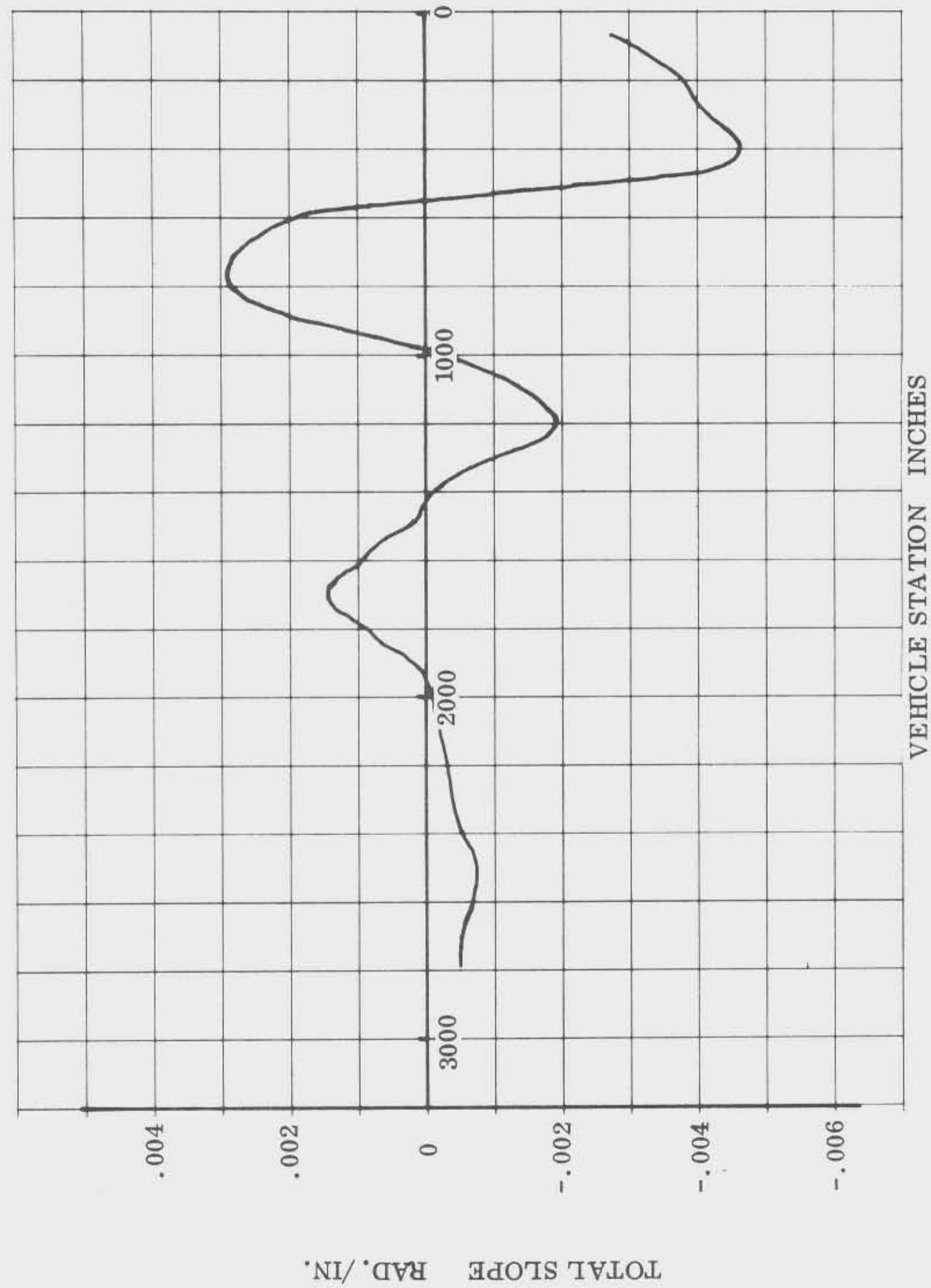


FIGURE 4.1.5.1-8 FOURTH BENDING MODE TOTAL SLOPE @ LIFT-OFF

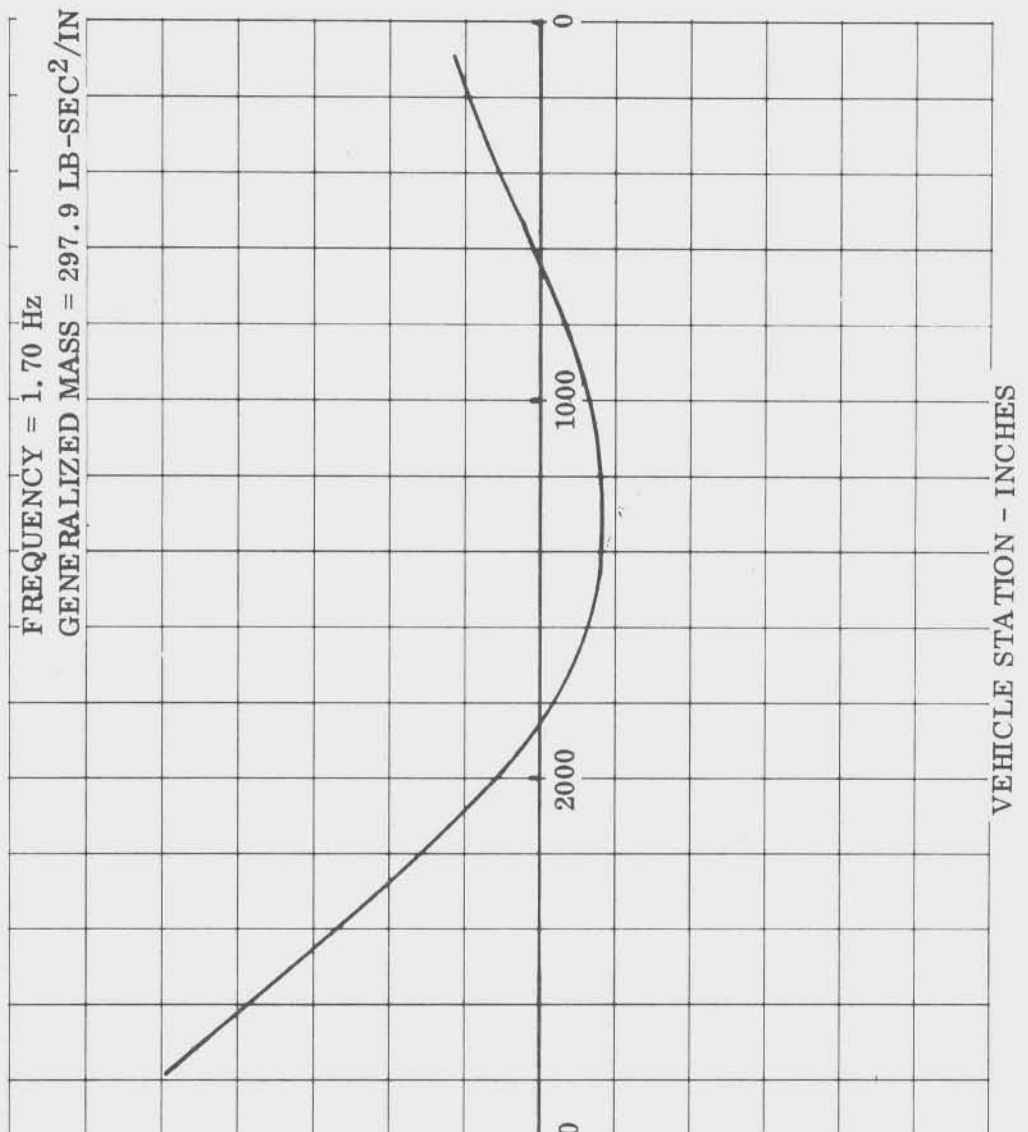


FIGURE 4.1.5.1-9 FIRST FREE-FREE BENDING MODE @ $Q\alpha$(MAX)

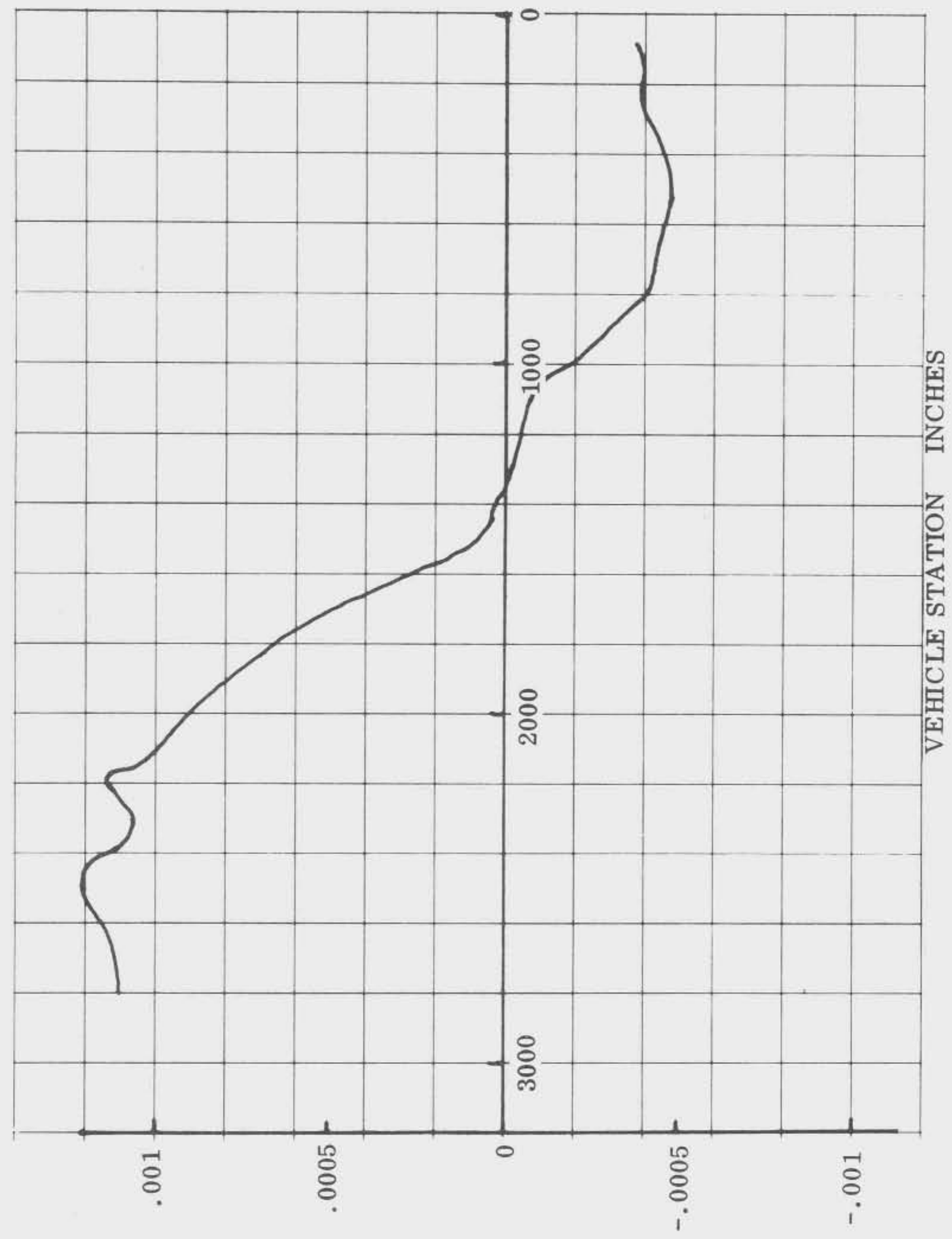


FIGURE 4.1.5.1-10 FIRST BENDING MODE TOTAL SLOPE @ Qα (MAX.)

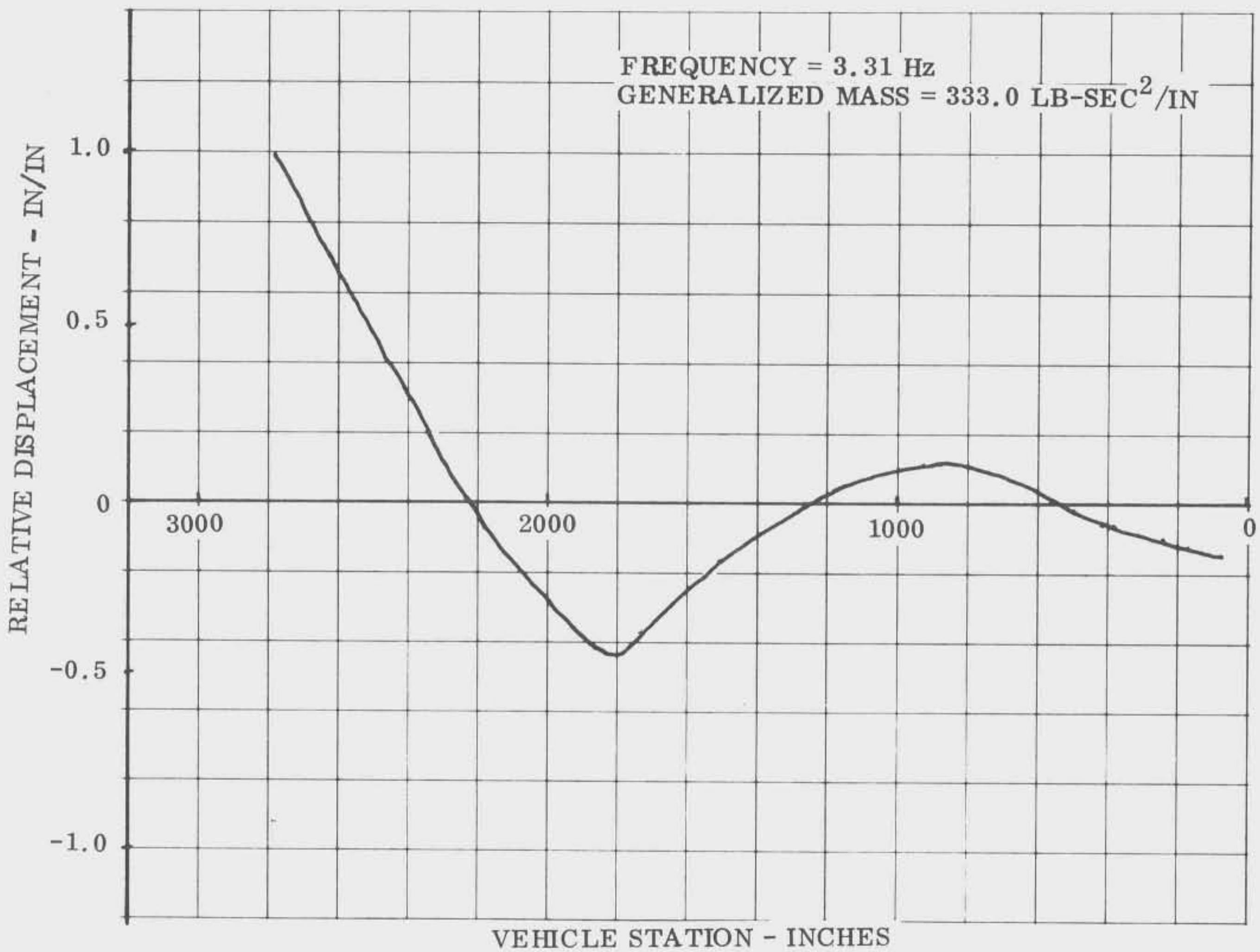


FIGURE 4.1.5.1-11 SECOND FREE-FREE BENDING MODE @ $Q\alpha$ (MAX.)

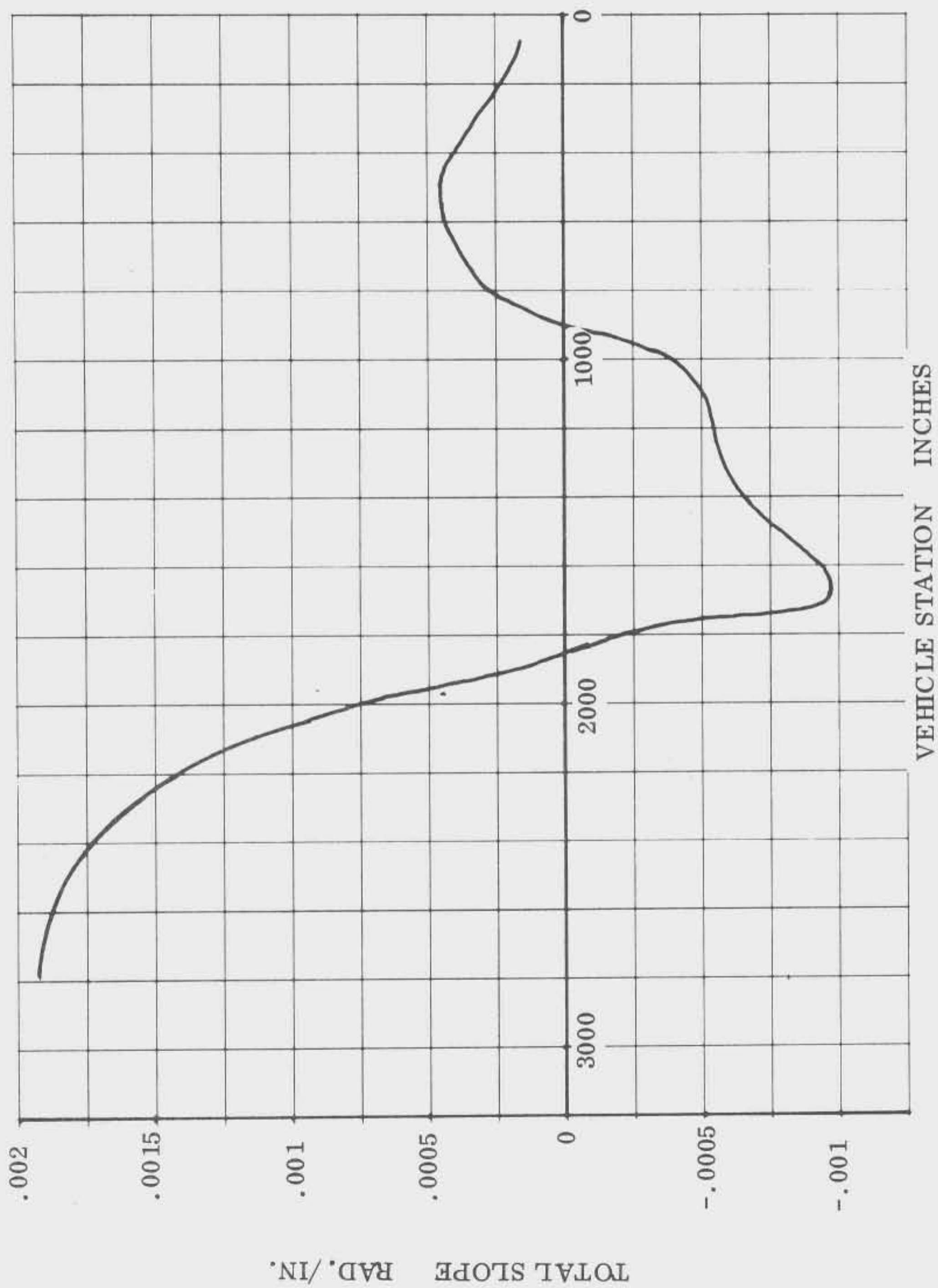


FIGURE 4.1.5.1-12 SECOND BENDING MODE TOTAL SLOPE @ Qα (MAX)

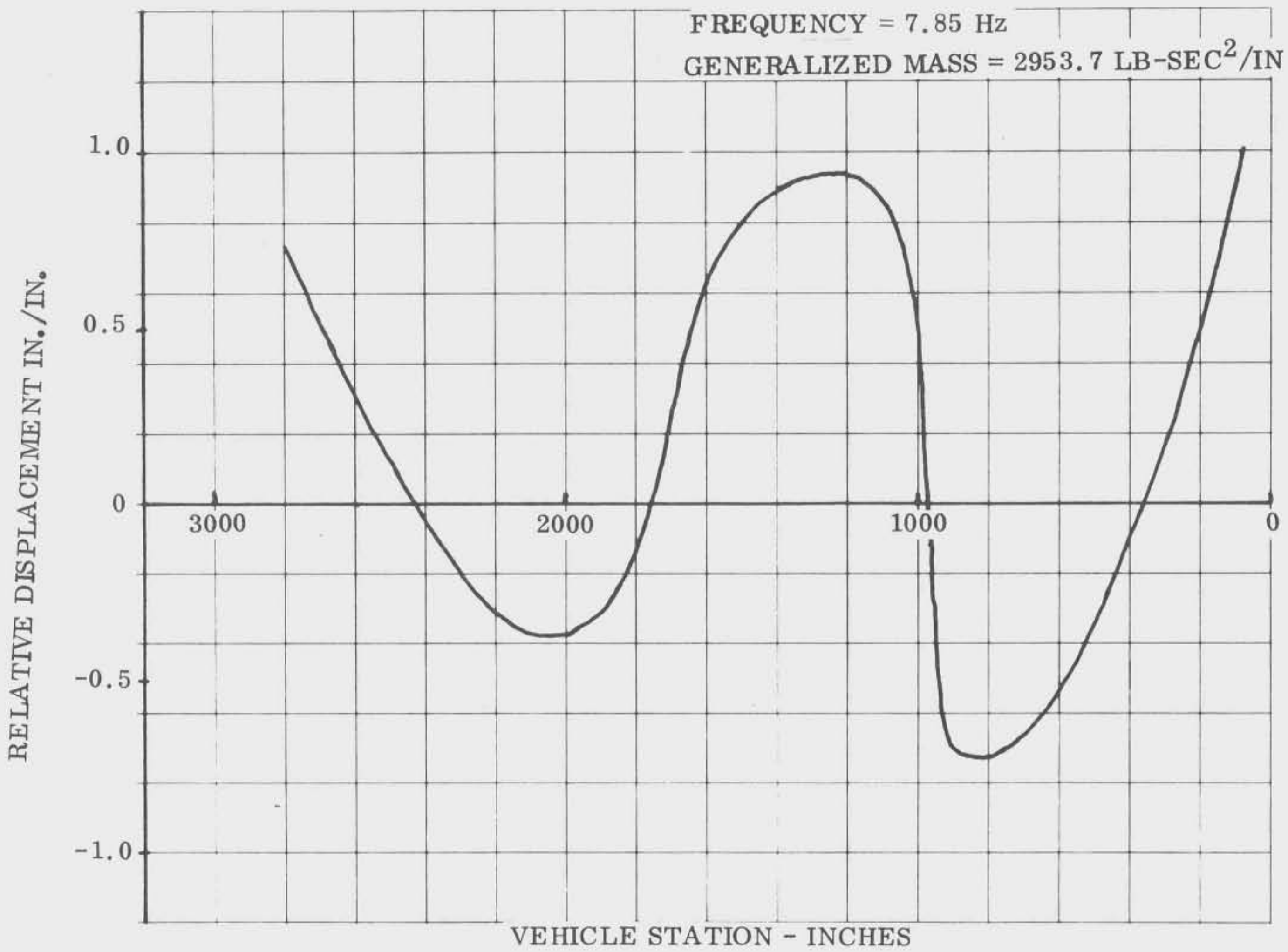


FIGURE 4.1.5.1-13 THIRD FREE-FREE BENDING MODE @ Qα (MAX.)

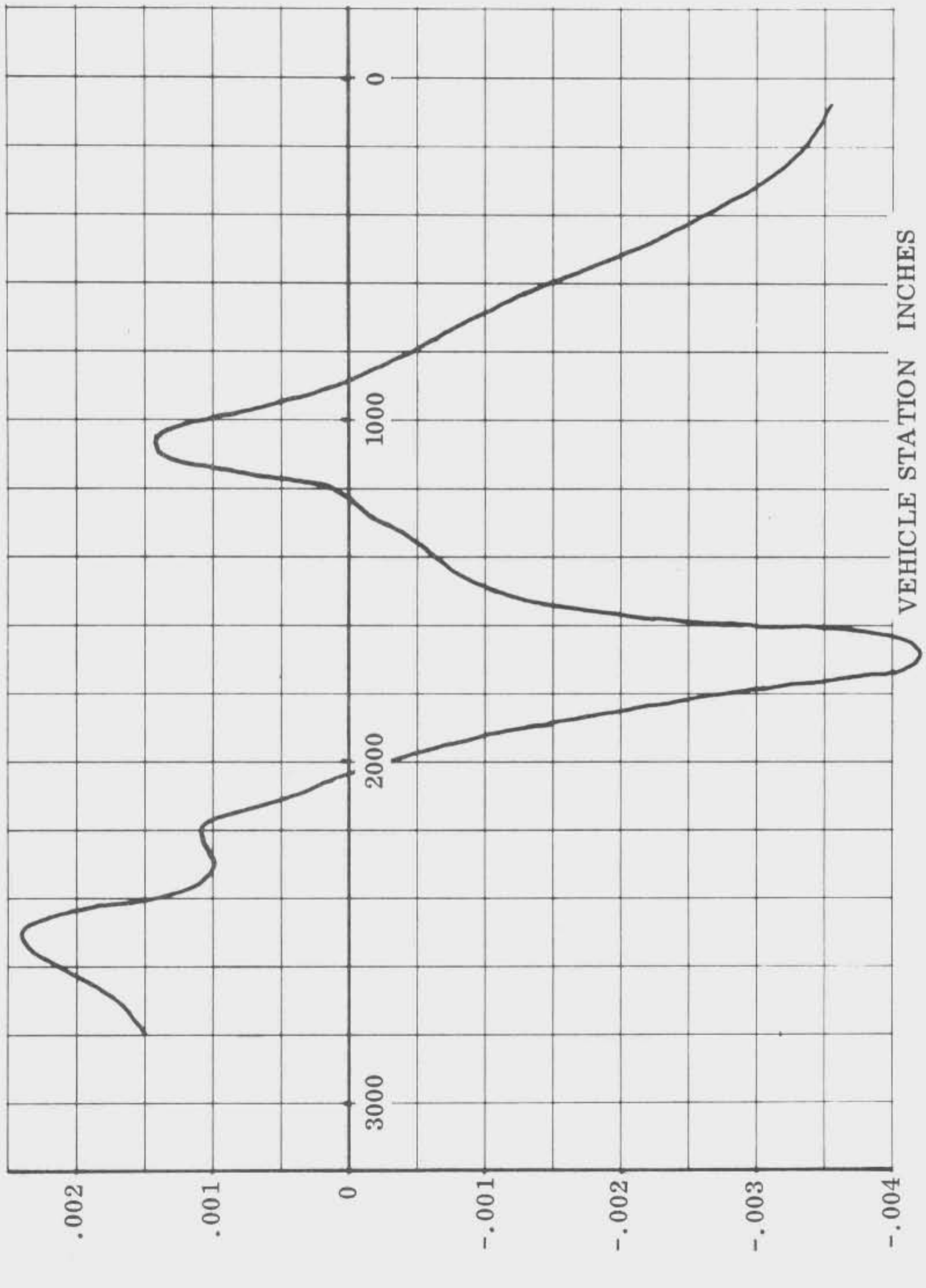


FIGURE 4.1.5.1-14 THIRD BENDING MODE TOTAL SLOPE @ $Q\alpha$ (MAX.)

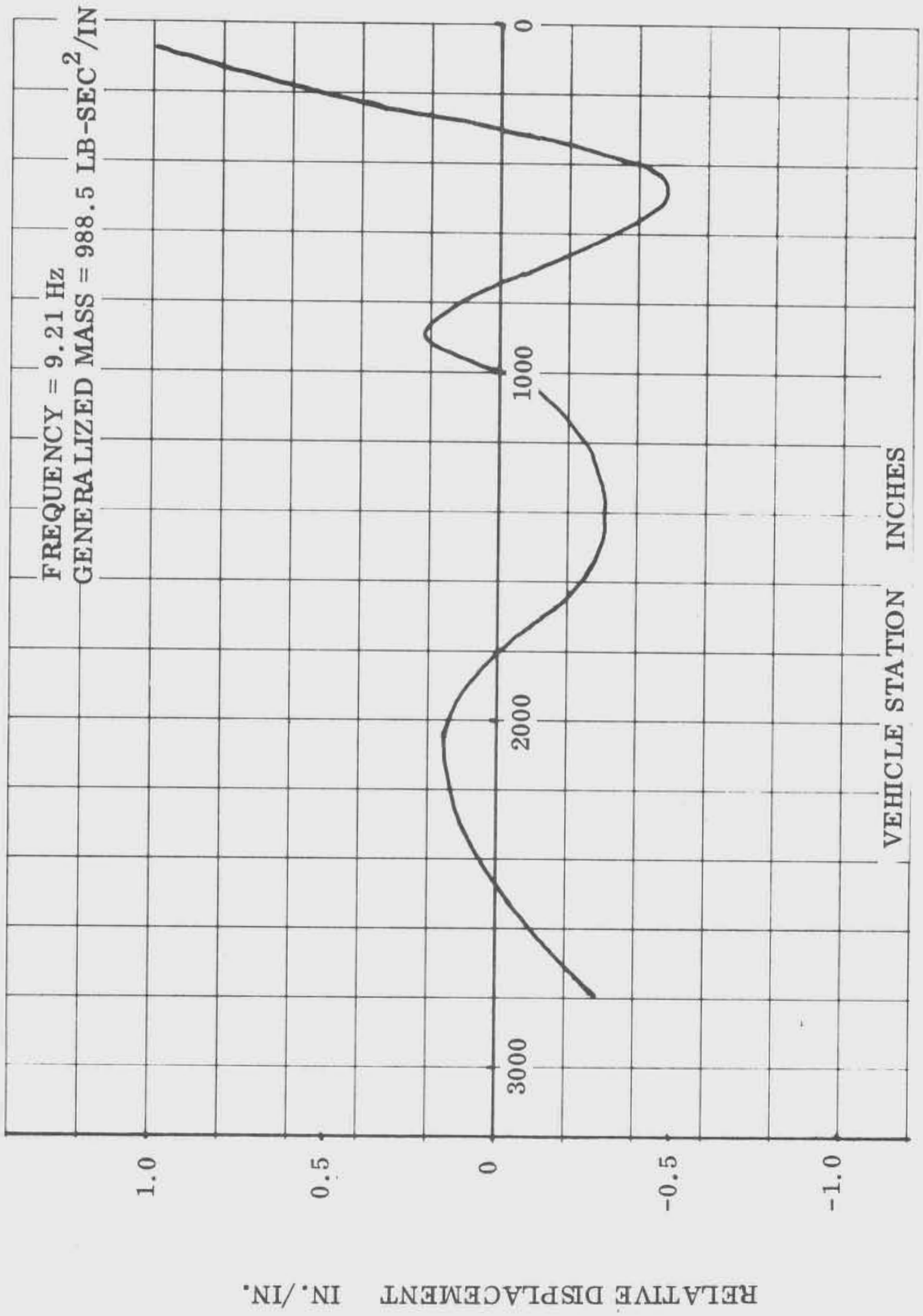


FIGURE 4.1.5.1-15 FOURTH FREE-FREE BENDING MODE @ Qα (MAX.)

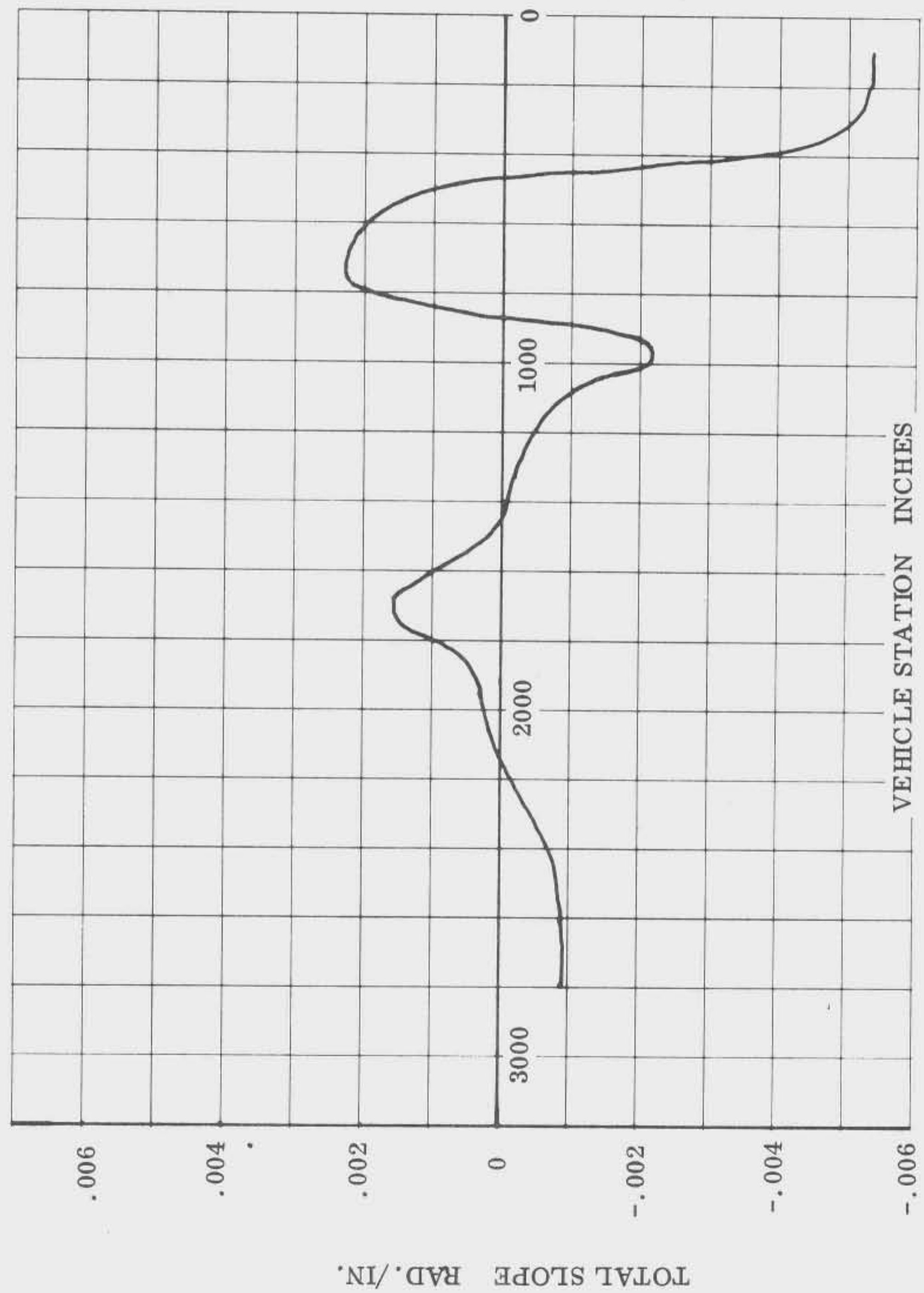


FIGURE 4.1.5.1-16 FOURTH BENDING MODE TOTAL SLOPE @ Qα (MAX.)

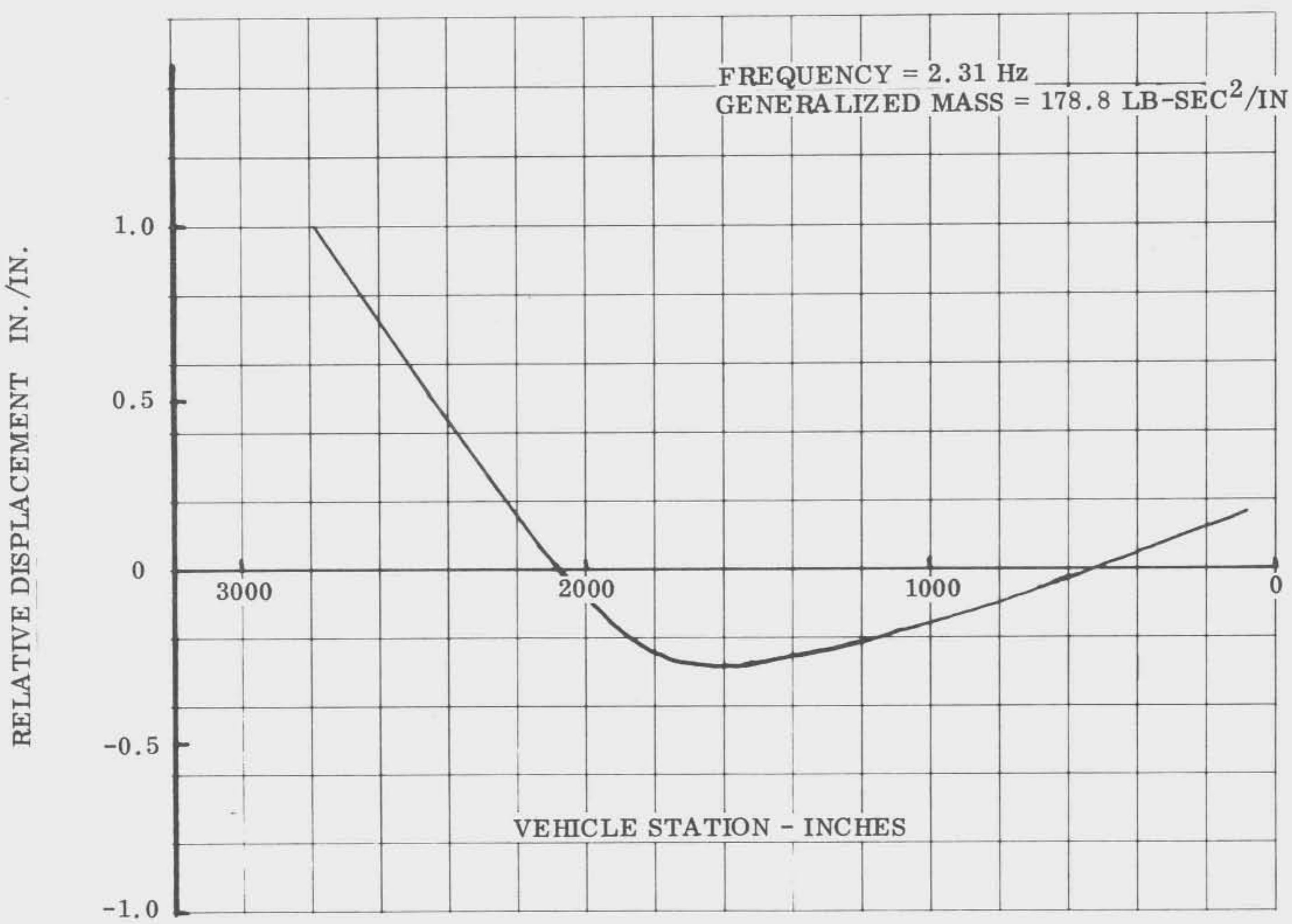


FIGURE 4.1.5.1-17 FIRST FREE-FREE BENDING MODE @ CUT-OFF

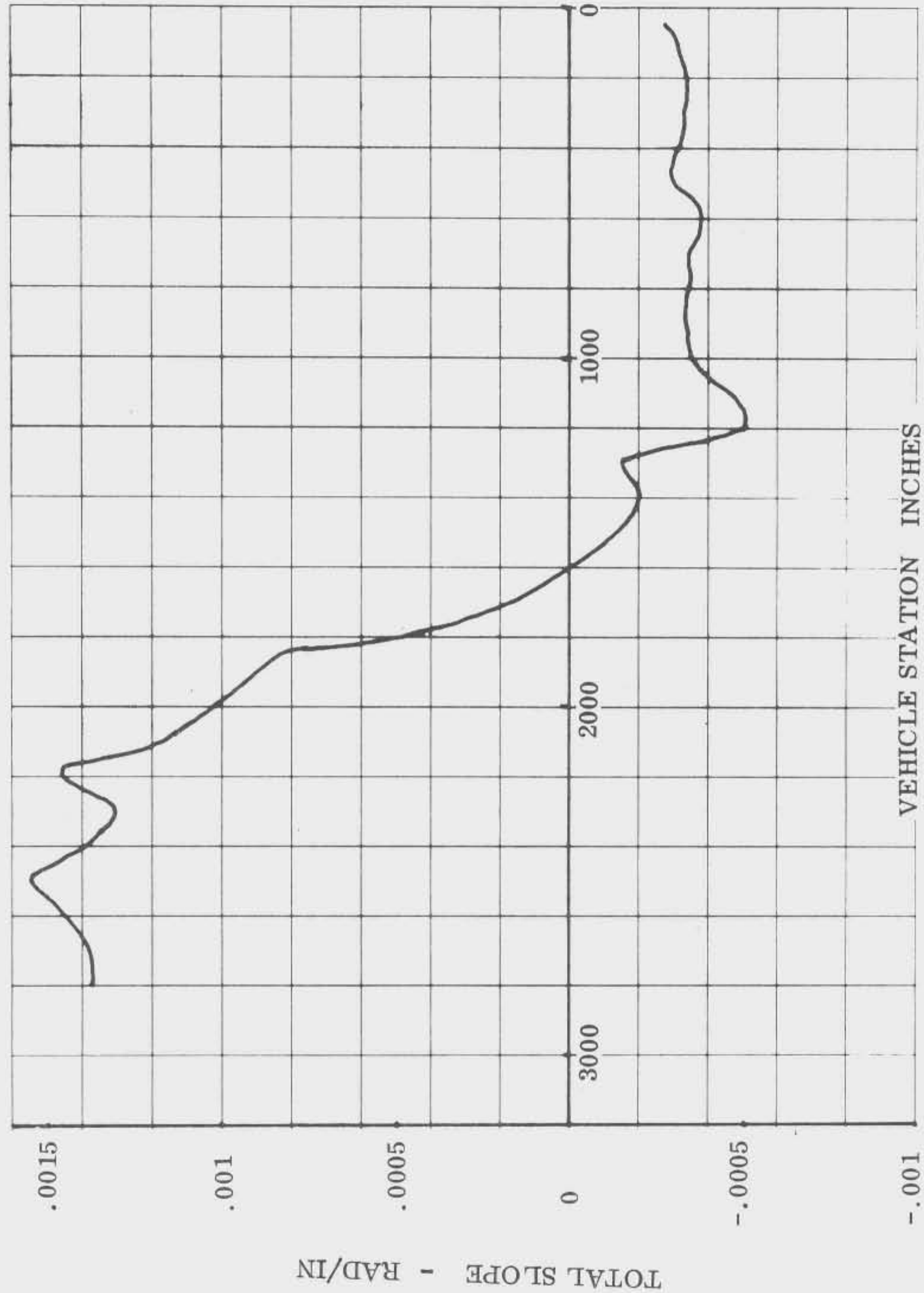


FIGURE 4.1.5.1-18 FIRST BENDING MODE TOTAL SLOPE @ CUT-OFF

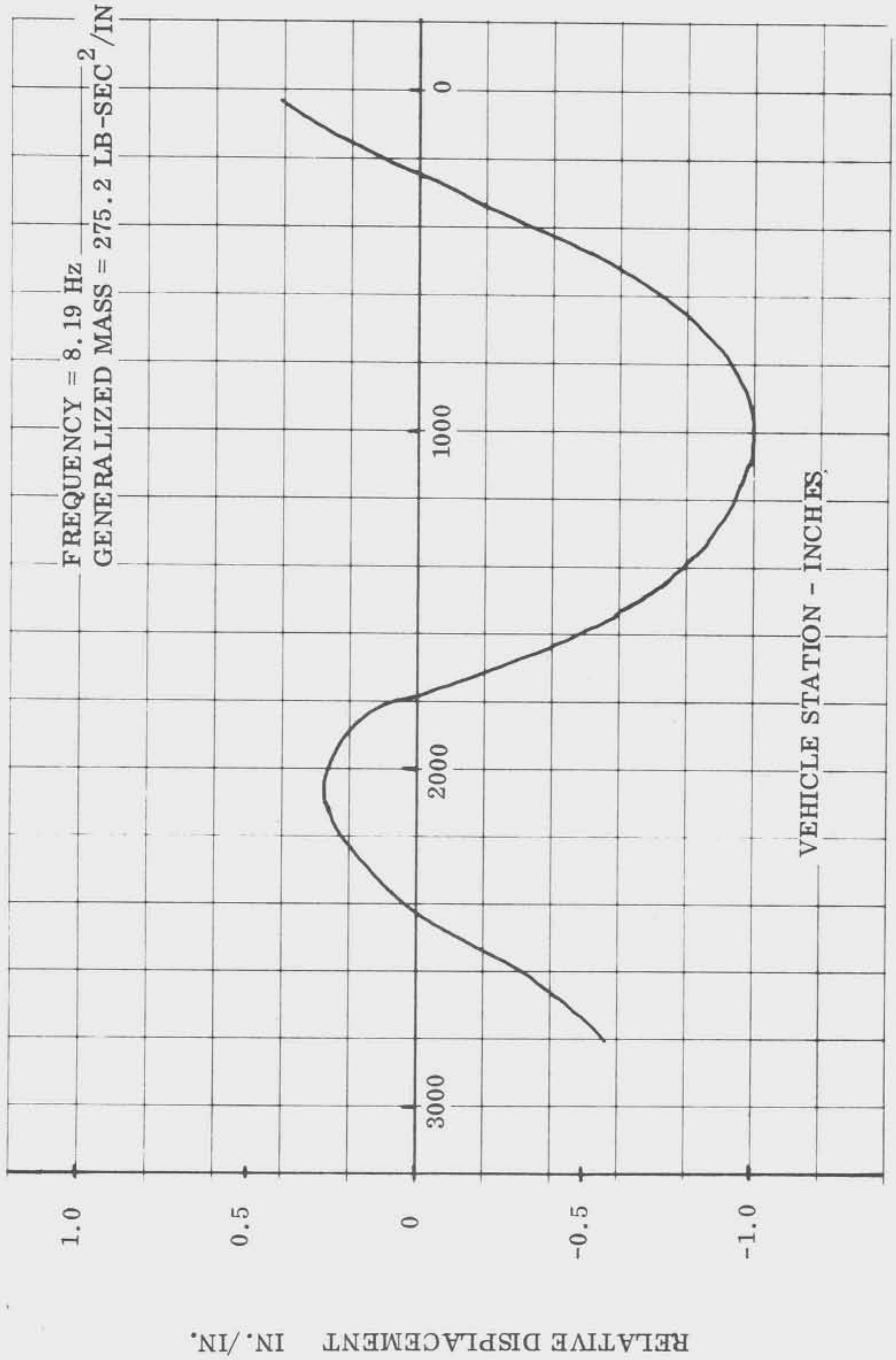


FIGURE 4.1.5.1-19 SECOND FREE-FREE BENDING MODE @ CUT-OFF

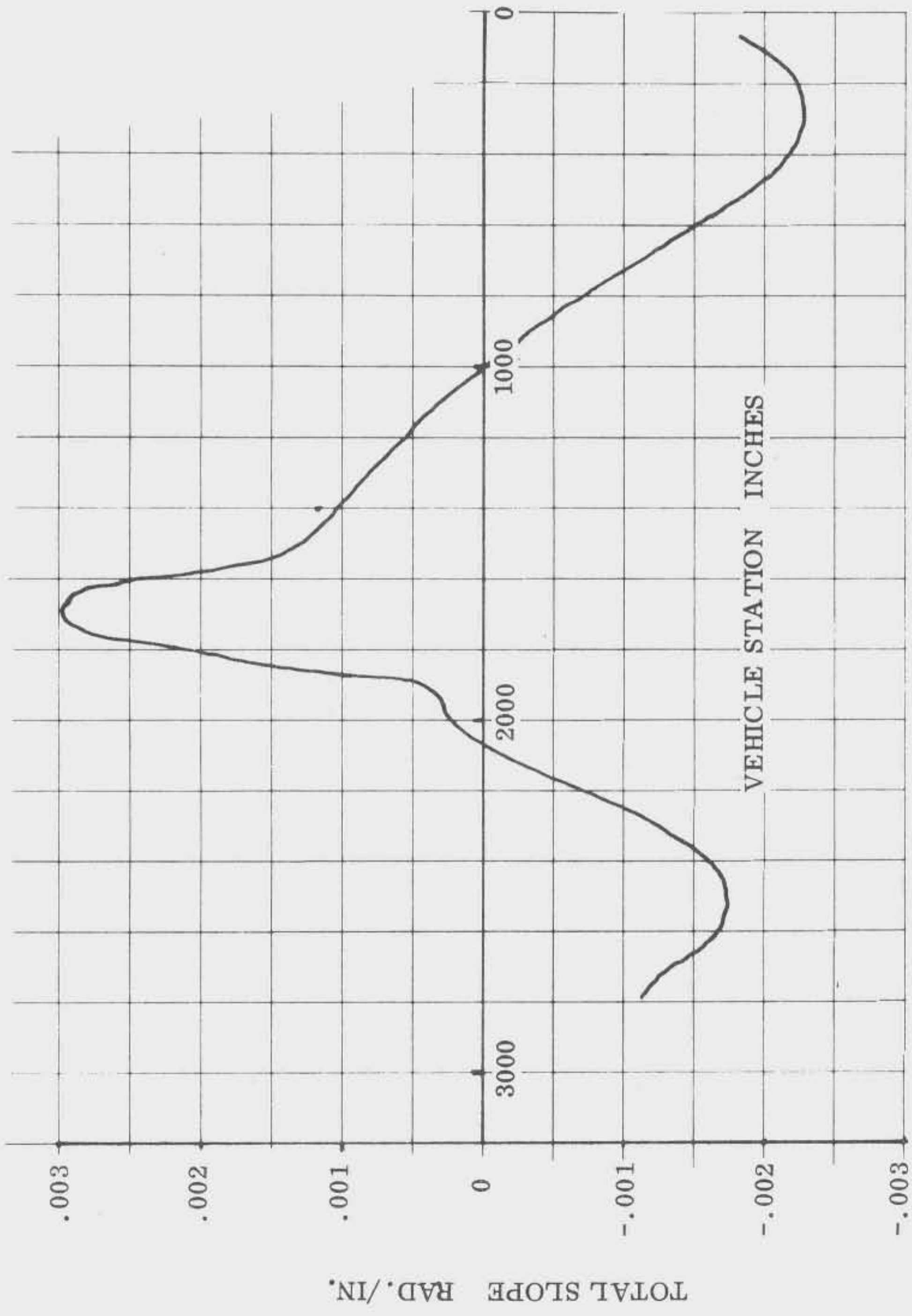


FIGURE 4.1.5.1-20 SECOND BENDING MODE TOTAL SLOPE @ CUT-OFF

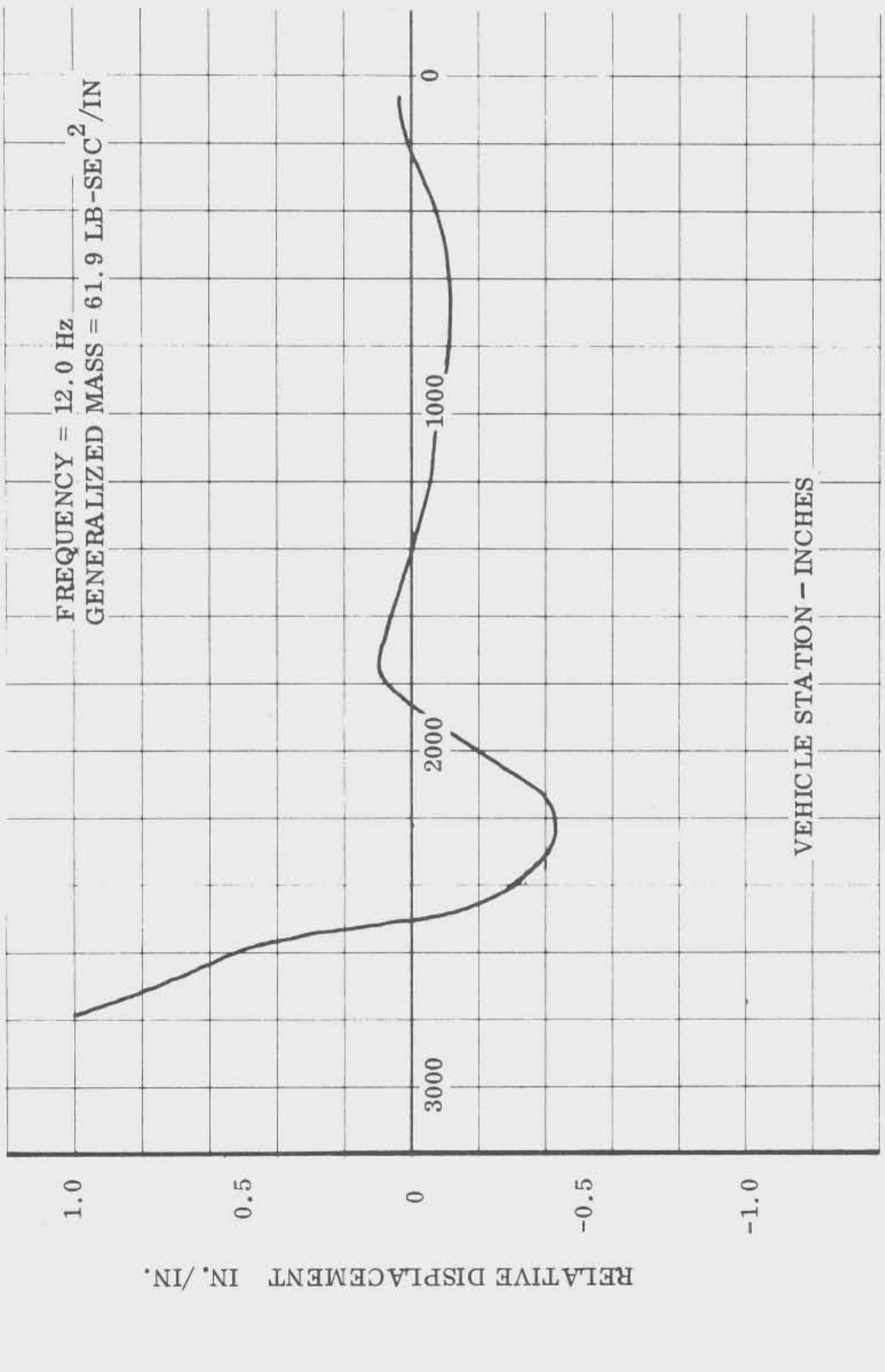


FIGURE 4.1.5.1-21 THIRD FREE-FREE BENDING MODE @ CUT-OFF

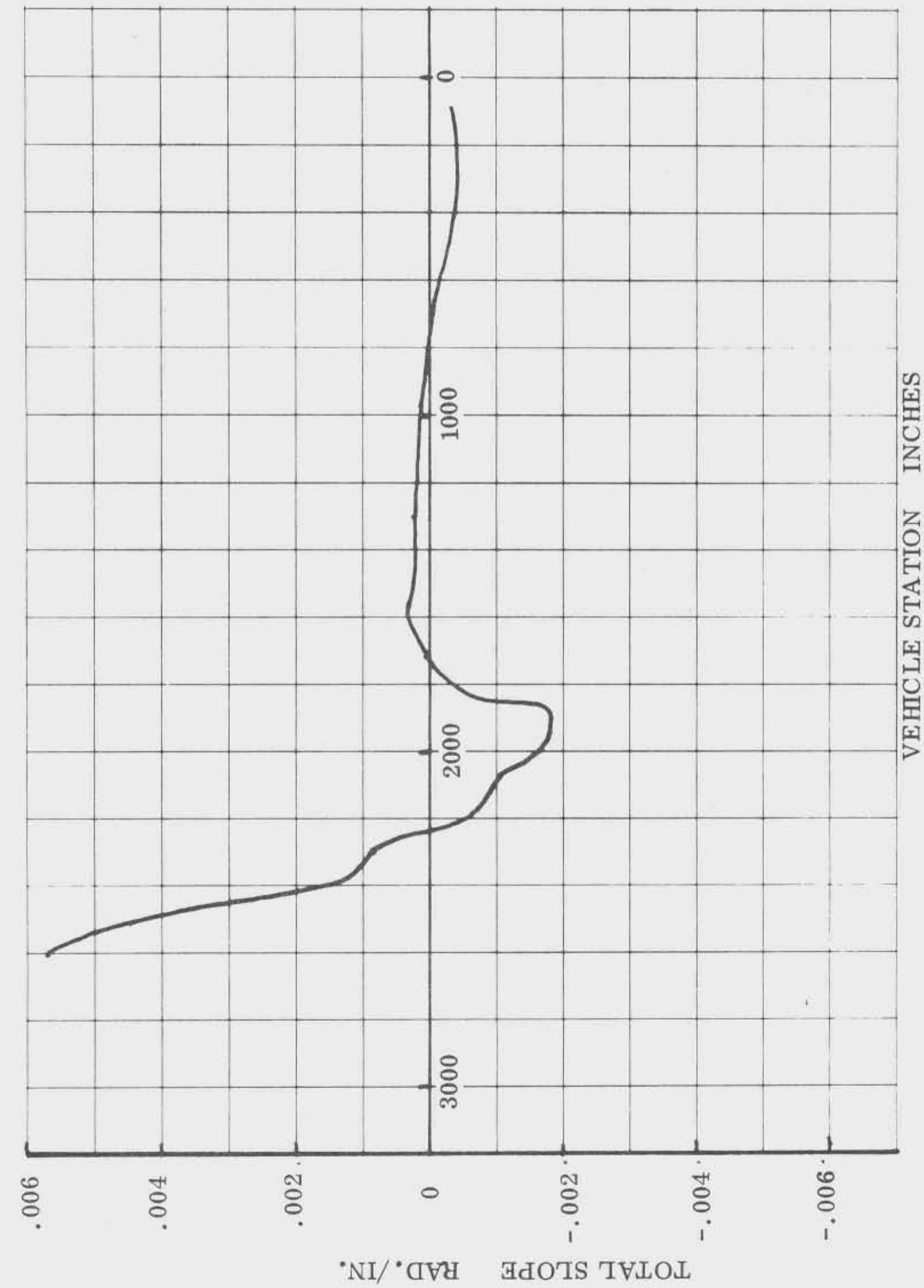


FIGURE 4.1.5.1-22 THIRD BENDING MODE TOTAL SLOPE @ CUT-OFF

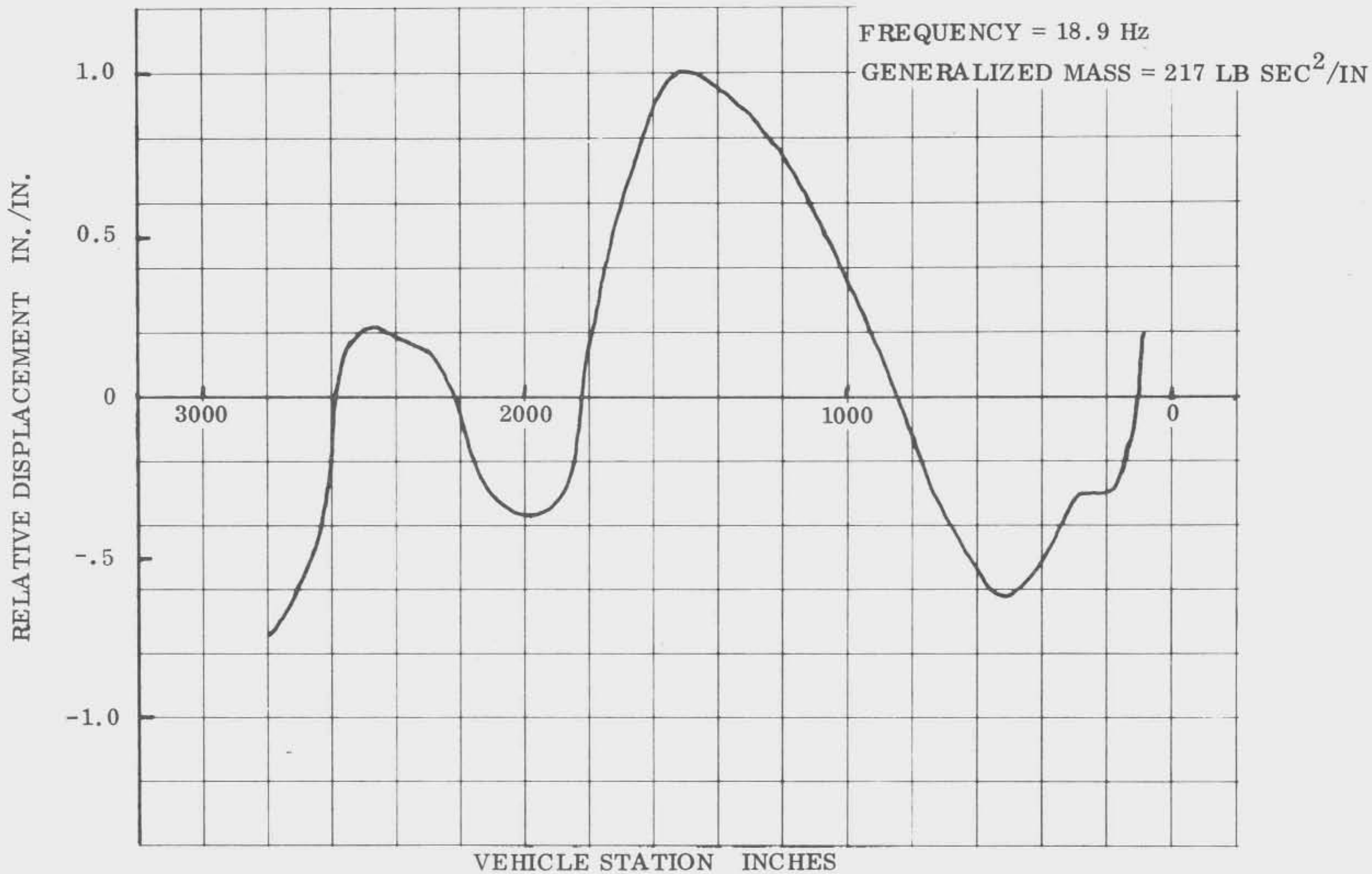


FIGURE 4.1.5.1-23 FOURTH FREE-FREE BENDING MODE @ CUT-OFF

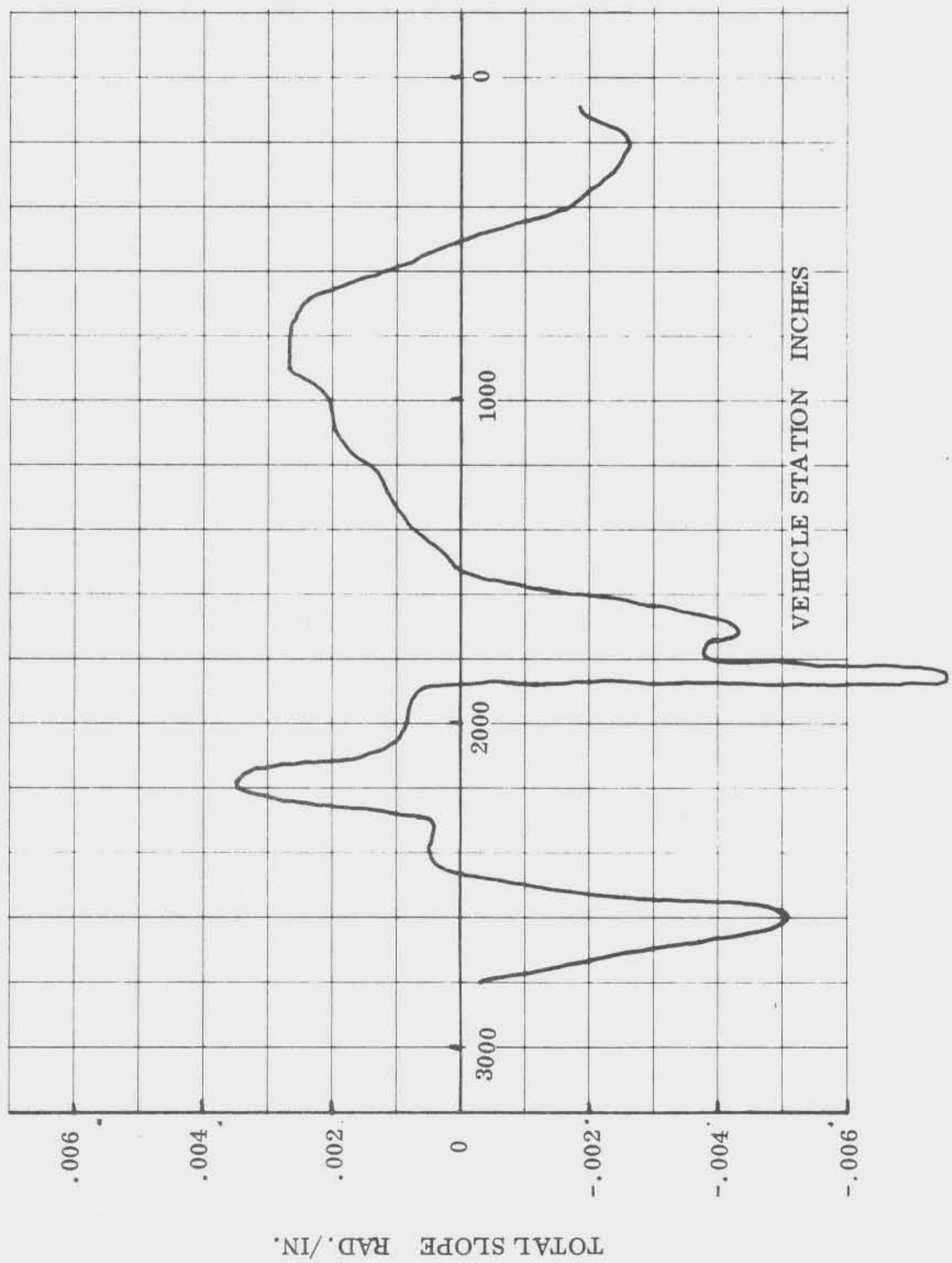


FIGURE 4.1.5.1-24 FOURTH BENDING MODE TOTAL SLOPE @ CUT-OFF

4.1.6 Structural Loads

The INT-20 initial baseline vehicle configuration, shown in Figure 4.1.6-1, was used for determining structural loads capability. Payload weight for this vehicle was 132,026 pounds (59886 kg) with a payload length of 43 feet (13.1m). The payload structural stiffness parameters (flexural and shear rigidity) assumed for this study are shown in Figure 4.1.6-2. These parameters were assumed because the payload had not yet been defined. Structural parameters, used in this analysis, for the S-IC stage, the S-IVB stage and the Instrument Unit are shown in Figures 4.1.6-3 through 4.1.6-7. Vehicle structural loads were determined for conditions on the launch pad, for lift-off and for flight. Propellant tank pressures and acceleration loads also were determined.

Structural loads were calculated as follows:

a. Longitudinal Force Distributions

Longitudinal force distributions for the critical design conditions were calculated as follows:

$$P(X) = \eta W(X) + D(X)$$

where

$P(X)$ = longitudinal force at any station
η = longitudinal acceleration load factor
$W(X)$ = vehicle weight forward of any vehicle station
$D(X)$ = aerodynamic axial drag force at any vehicle station

b. Combined Loads

The ultimate compressive combined loads are determined as follows:

$$N_C \text{ ULT} = \text{F.S.}\left[\frac{P(X)}{2\pi R(X)} + \frac{BM(X)}{\pi R^2(X)}\right] - P_U \text{ Min } \frac{R(X)}{2}$$

where: $P(X)$ = distributed longitudinal forces including aerodynamic drag

$BM(X)$ = distributed bending moment

$R(X)$ = distributed body radius

4.1.6 (Continued)

$P_{u_{min}}$ = minimum ullage pressure (applicable to tank shells only).

F.S. = ultimate factor of safety of 1.4 for manned flight.

Ultimate tension loads were determined as follows:

$$N_{T\ ULT} + F.S. \left[\frac{BM(X)}{\pi R^2\ (X)} - \frac{P(X)}{2\,\pi R(X)} + P_{u_{max}} \frac{R(X)}{2} \right]$$

Primary vehicle structural capability exceeds the requirements. However, the lower bulkhead of the RP-1 tank in the S-IC stage is overloaded in hoop-compression (see paragraph 4.1.6.4), and a manned factor of safety of 1.4 cannot be maintained under flight conditions with unrestricted longitudinal acceleration. This condition was eliminated, and a factor of safety of 1.4 was maintained, by restricting the maximum longitudinal acceleration of the vehicle to 3.68 g's and 4.68 g's for first two-engine cutoff and final cutoff, respectively, during S-IC stage flight.

4.1.6.1 Wind Profile

The ground and inflight wind environments which were used in calculation of the respective bending moment and shear distributions were obtained using the MSFC design wind criteria and methods in Reference 4.1.4.5-1. The inflight synthetic wind profile was constructed from a 99 percent shear build-up envelope, reduced by 15 percent, merged with the 95 percentile wind envelope at 10,000 meters altitude. An imbedded jet gust, reduced in magnitude by 15 percent, was superimposed upon the peak of the wind profile. The inflight wind profile is shown in Figure 4.1.6.1-1. Surface wind speed envelopes for 99.9 and 99 percentile winds, also from reference 4.1.4.5-1, were used for the pre-launch and launch winds.

4.1.6.2 On-Pad and Lift-Off Loads

Shear and bending moment distribution due to the 99.9 percentile pre-launch and the 99 percentile launch winds are shown in Figures 4.1.6.2-1 and 4.1.6.2-2. Longitudinal force distribution for the on-pad, fueled, unpressurized condition is shown in Figure 4.1.6.2-3, and for the emergency shutdown condition in Figure 4.1.6.2-4. Ultimate compressive and tensile combined loads for these conditions are given in Tables 4.1.6.2-I through 4.1.6.2-IV.

4.1.6.2 (Continued)

Also shown in Figures 4.1.6.1-1 and -2 are shear and bending moment distributions for a 95 percentile qualification wind for the Instrument Unit (IU) access door. The access door installation is load-carrying and, with door removed, must withstand ground wind loads (see section 4.2.5.3). Ultimate structural loads and capability for ground wind conditions are shown in Table 4.2.5.3-II.

4.1.6.3 Flight Loads

a. General

Combined compressive and tensile loads and vehicle capability are shown for the INT-20 vehicle stages in Figures 4.1.6.3-1 through 4.1.6.3-4. The load capability everywhere exceeds the applied loads.

b. Maximum ($q\alpha$)

Bending moment distribution at the most severe vehicle structural loading condition, at maximum ($q\alpha$), is plotted in Figure 4.1.6.3-5. Longitudinal force distribution at this condition are shown in Figure 4.1.6.3-6. Ultimate compressive and tensile combined loads at maximum ($q\alpha$) are given in Tables 4.1.6.3-I and 4.1.6.3-II.

c. Peak Acceleration

Longitudinal force distribution at peak acceleration is shown in Figure 4.1.6.3-7. Ultimate compressive and tensile combined loads at this condition are given in Tables 4.1.6.3-III and 4.1.6.3-IV.

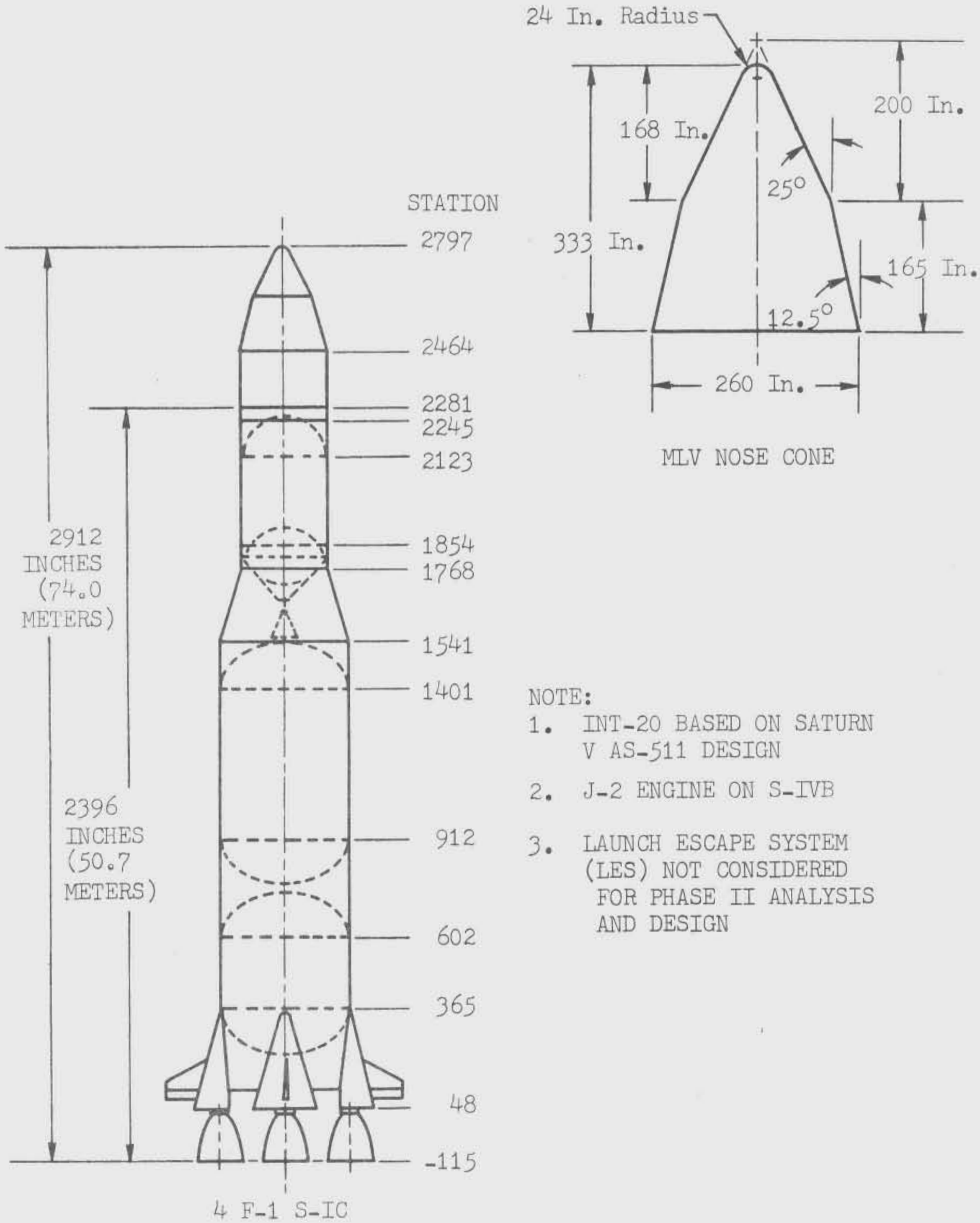


FIGURE 4.1.6-1 INT-20 BASELINE CONFIGURATION

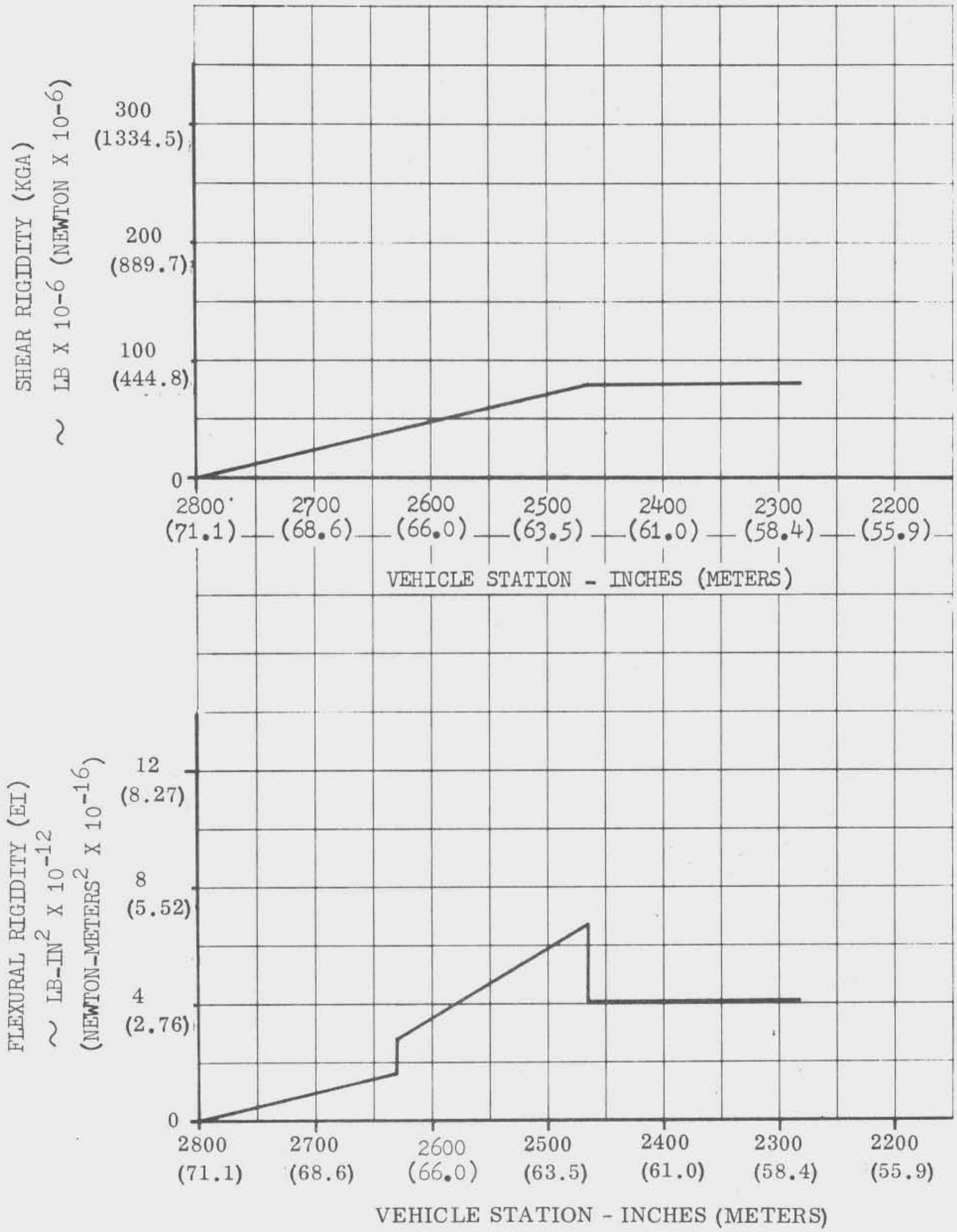


FIGURE 4.1.6-2 INT-20 BASELINE VEHICLE PAYLOAD STRUCTURAL STIFFNESS

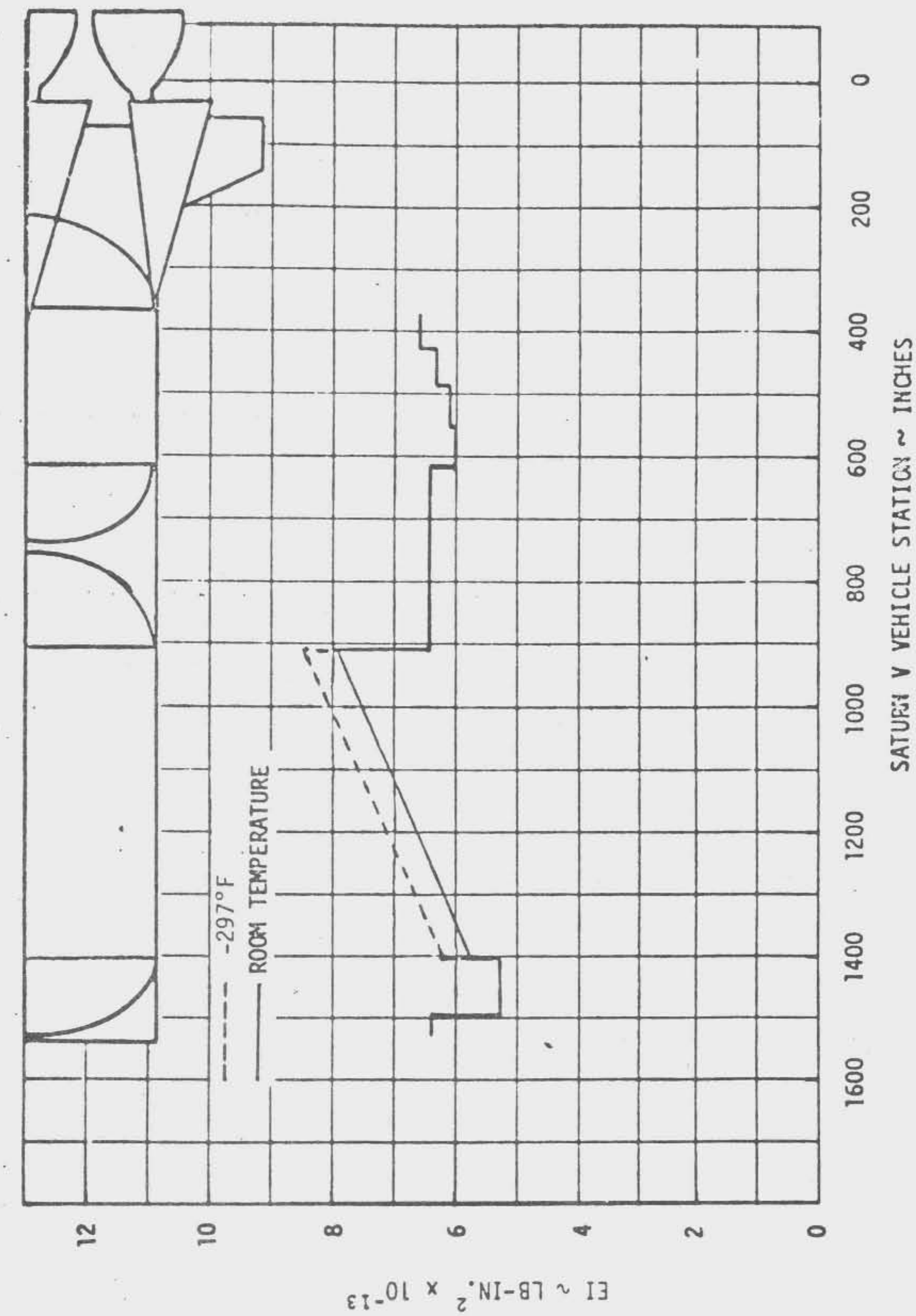


FIGURE 4.1.6-3 S-1C BENDING STIFFNESS

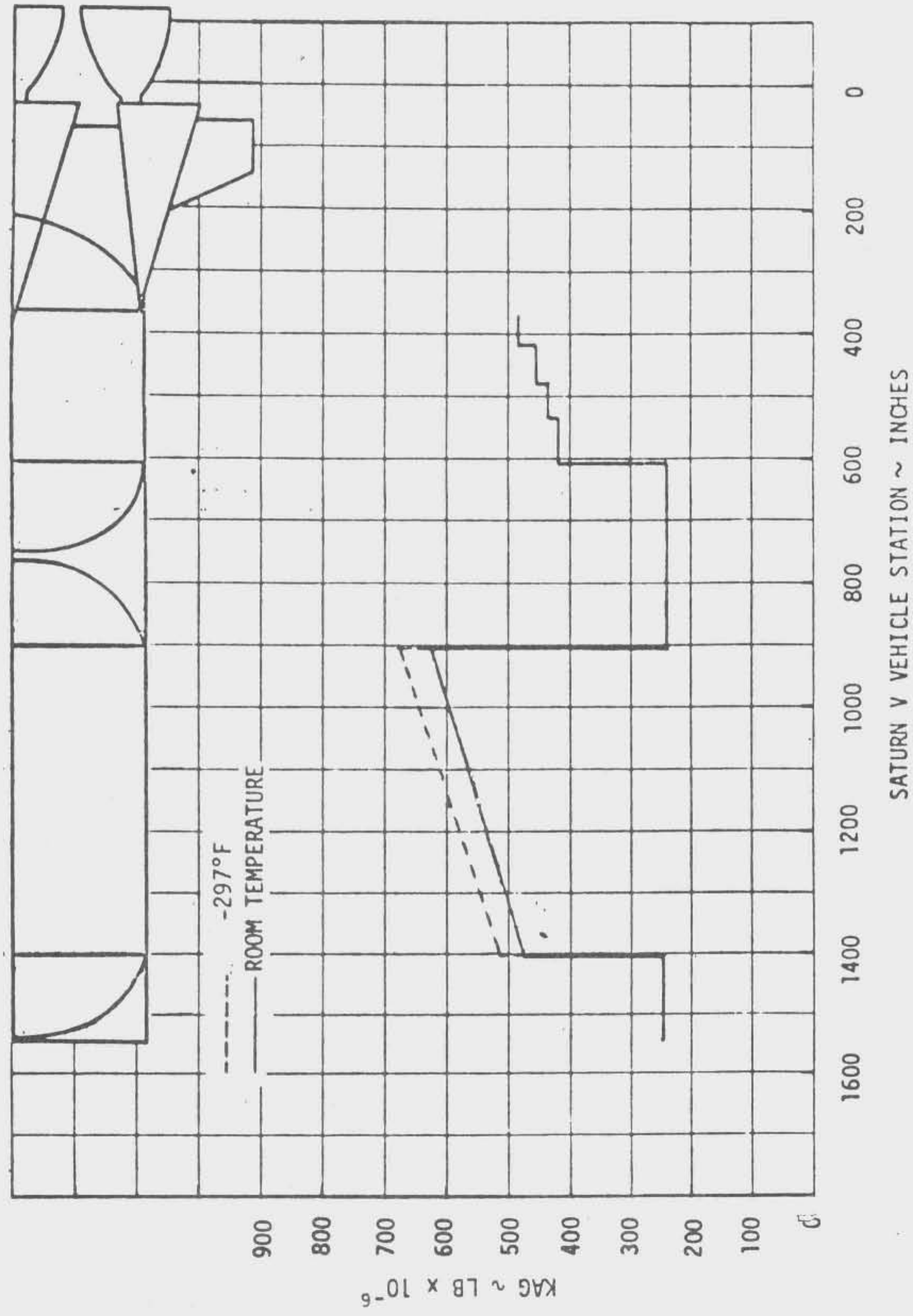


FIGURE 4.1.6-4 S-1C SHEAR STIFFNESS

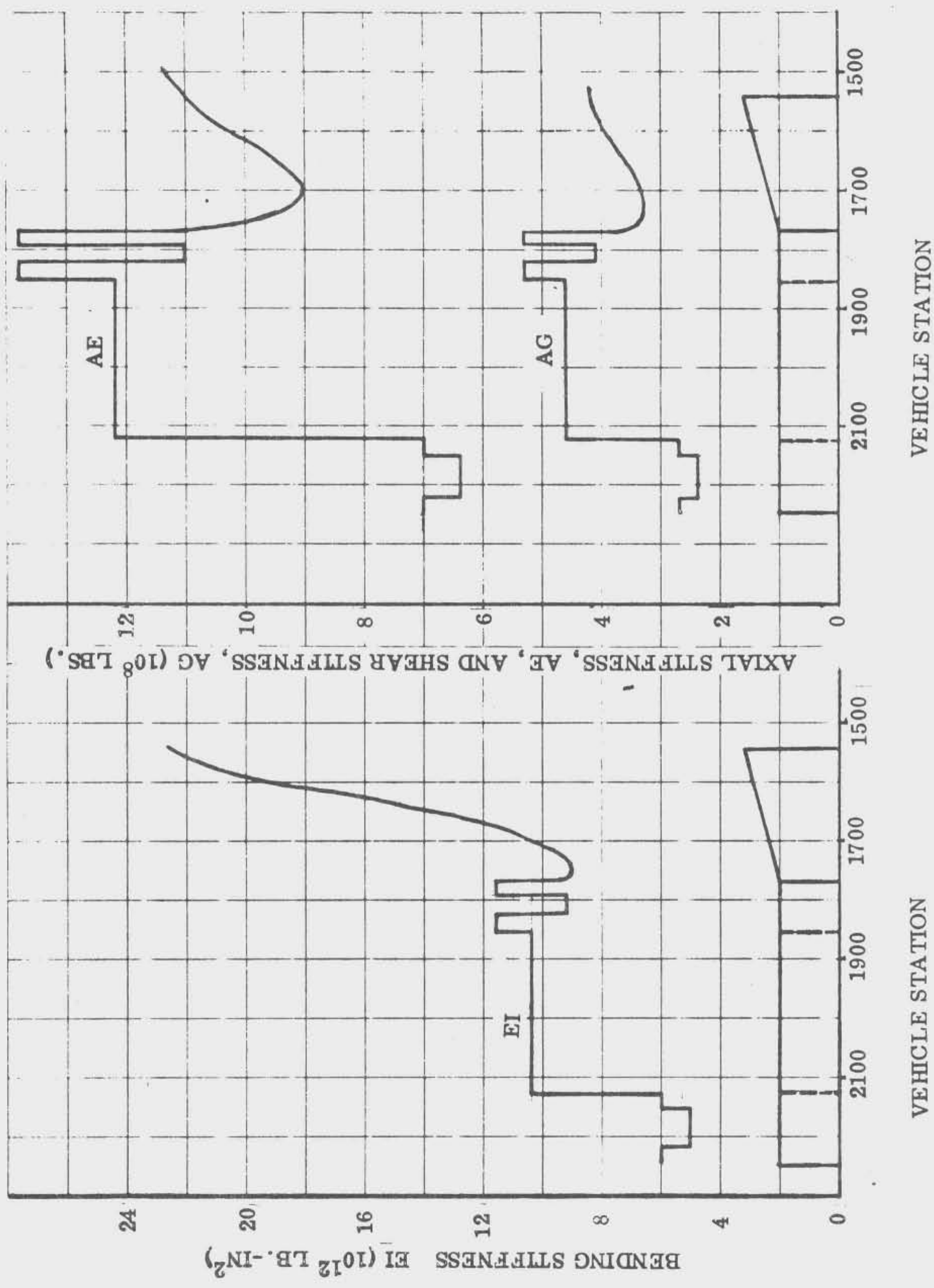


FIGURE 4.1.6-5 S-VB STAGE STIFFNESS

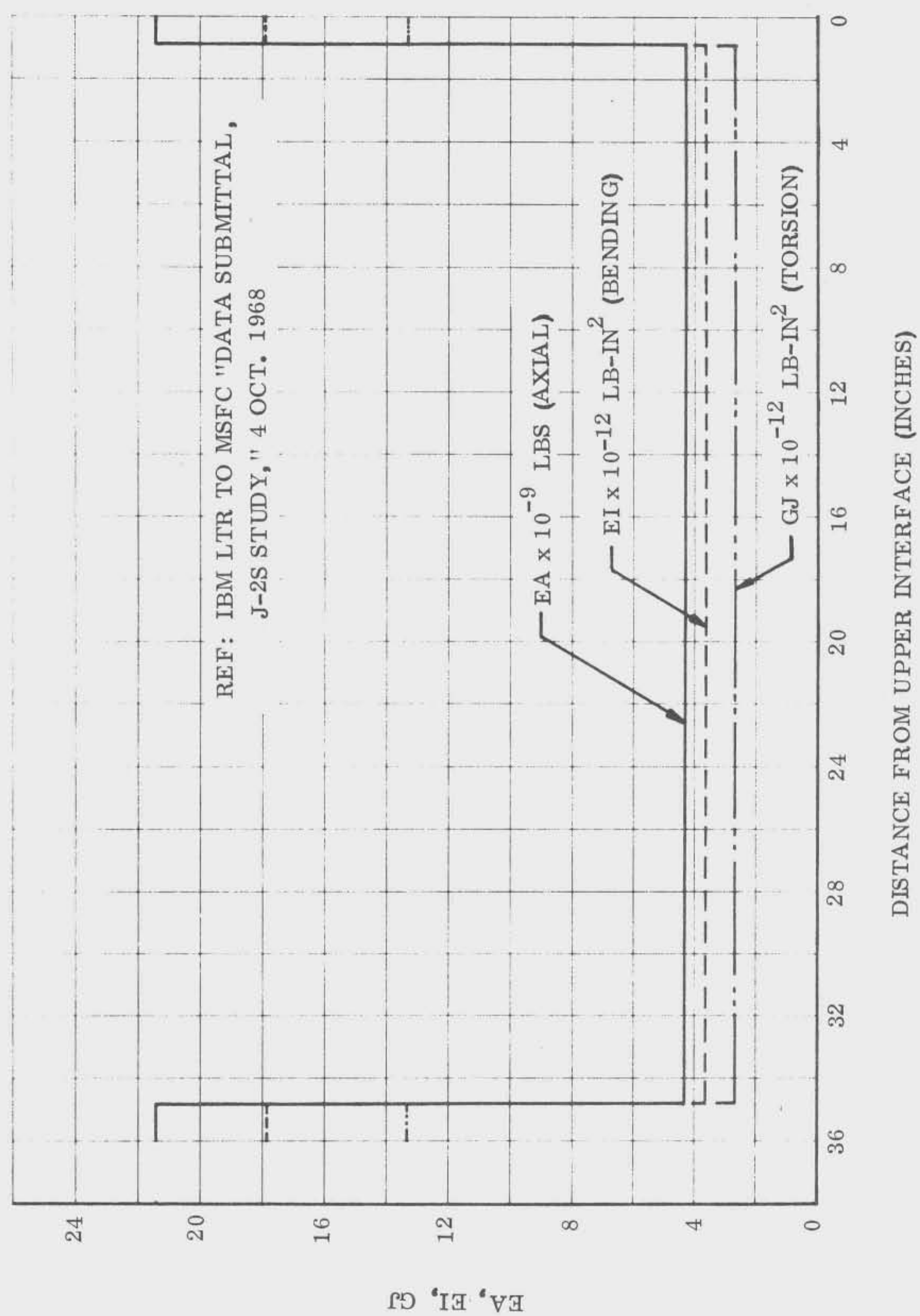


FIGURE 4.1.6-6 INSTRUMENT UNIT STIFFNESS

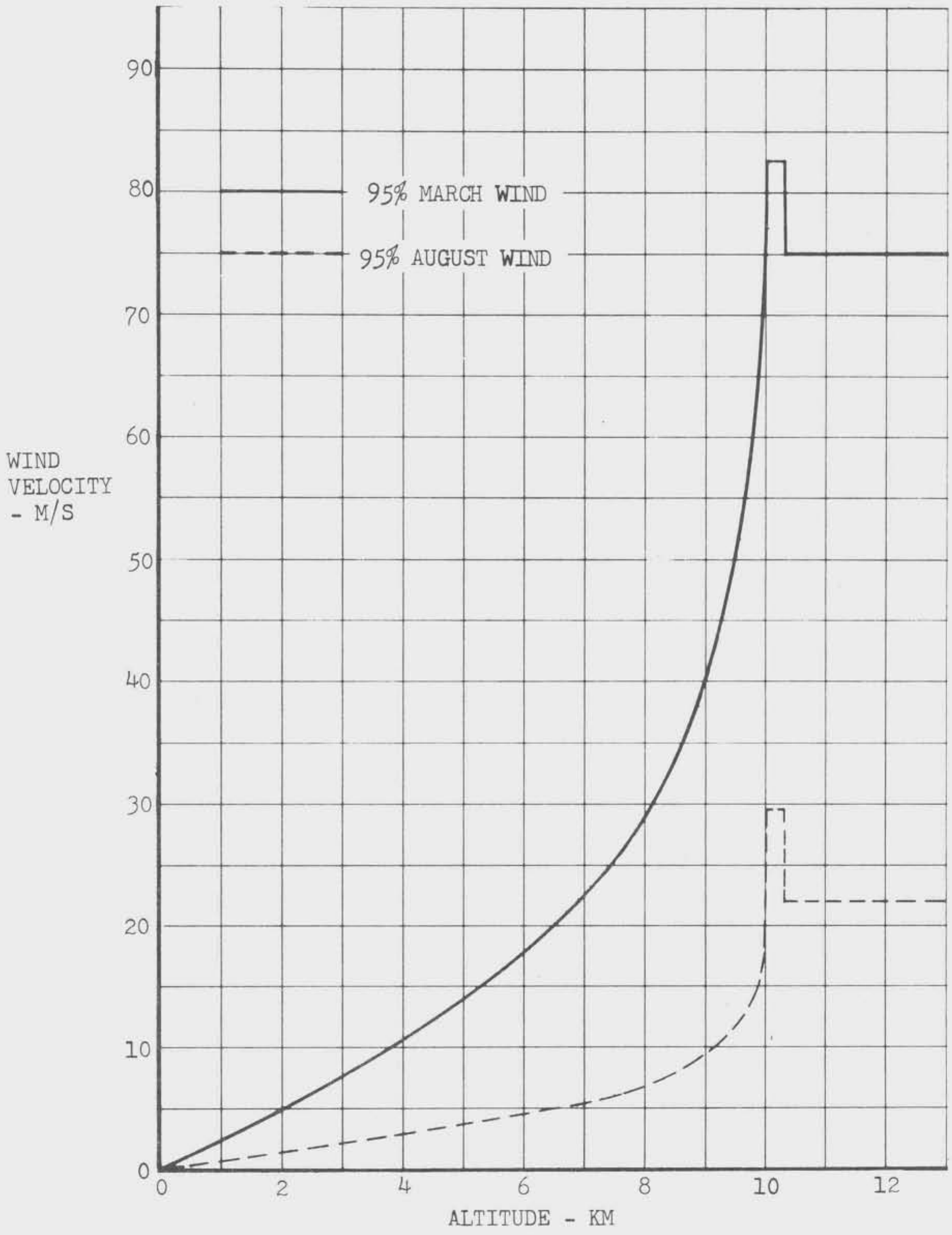


FIGURE 4.1.6.1-1 INFLIGHT WIND PROFILE FOR MARCH AND AUGUST

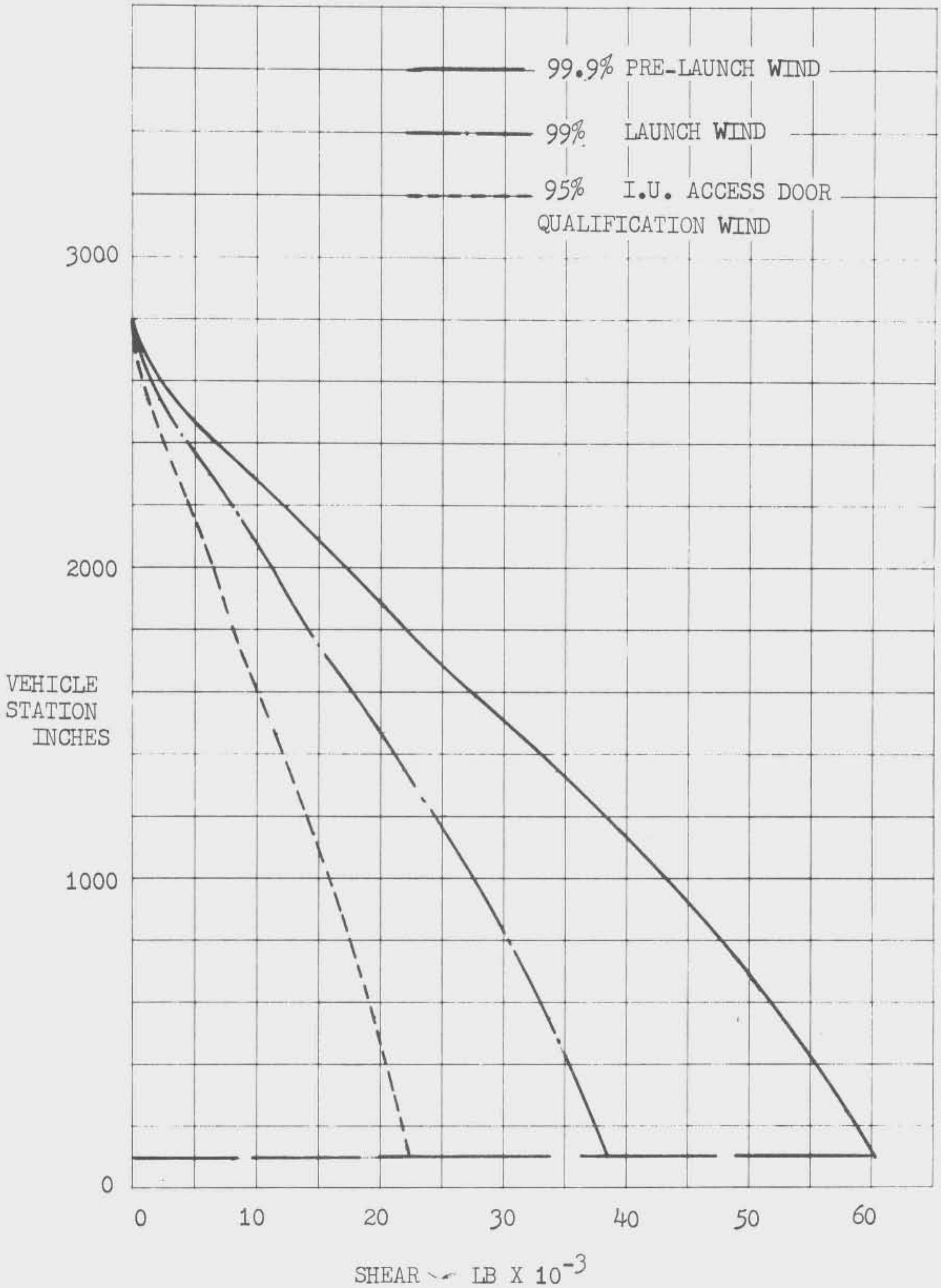


FIGURE 4.1.6.2-1 INT-20 BASELINE VEHICLE GROUND WIND SHEAR DISTRIBUTIONS

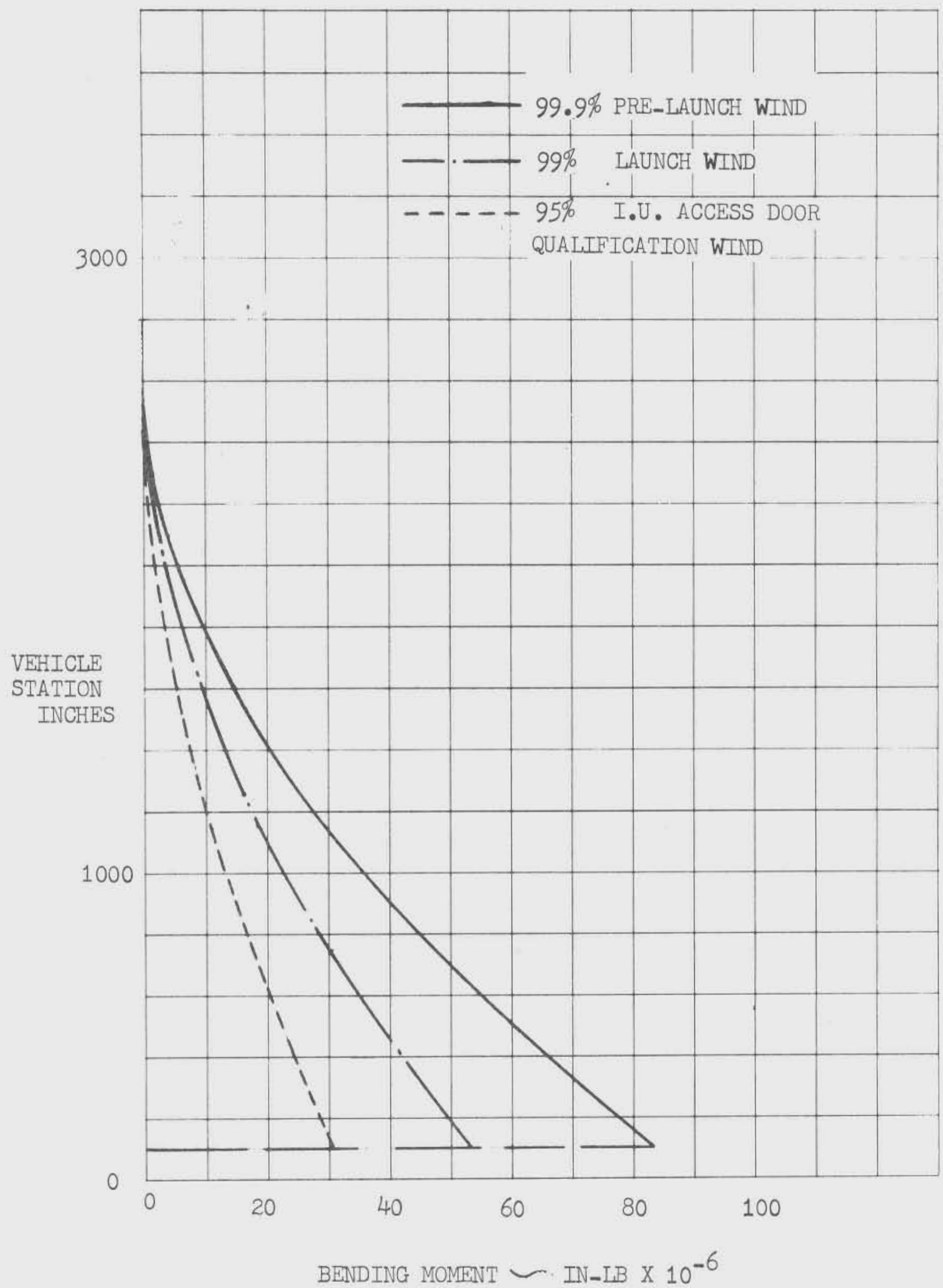


FIGURE 4.1.6.2-2 INT-20 BASELINE VEHICLE GROUND WIND BENDING MOMENT DISTRIBUTIONS

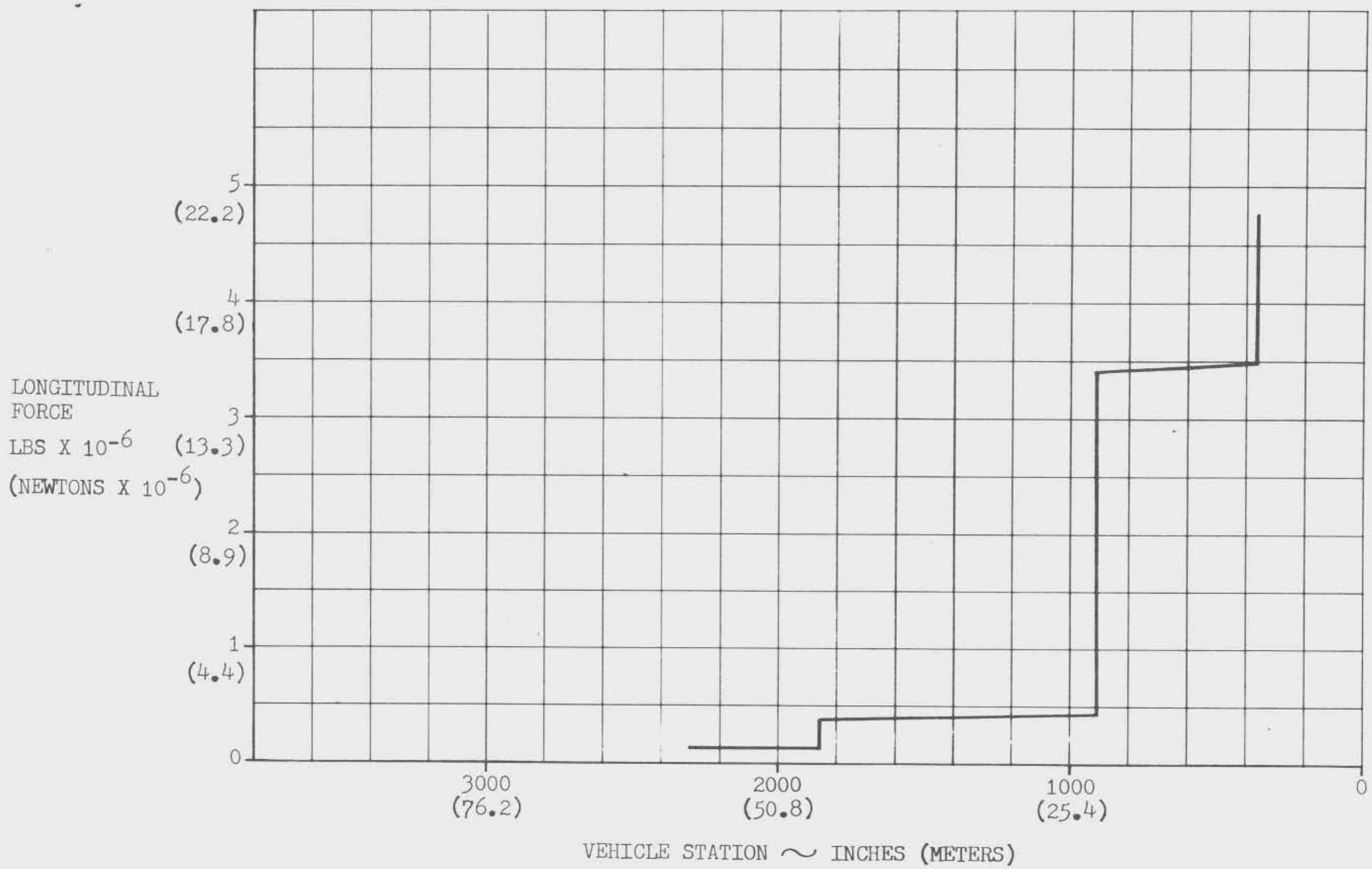


FIGURE 4.1.6.2-3 INT -20 BASELINE VEHICLE LONGITUDINAL FORCE DISTRIBUTION FOR ON-PAD, FUELED, UNPRESSURIZED CONDITION

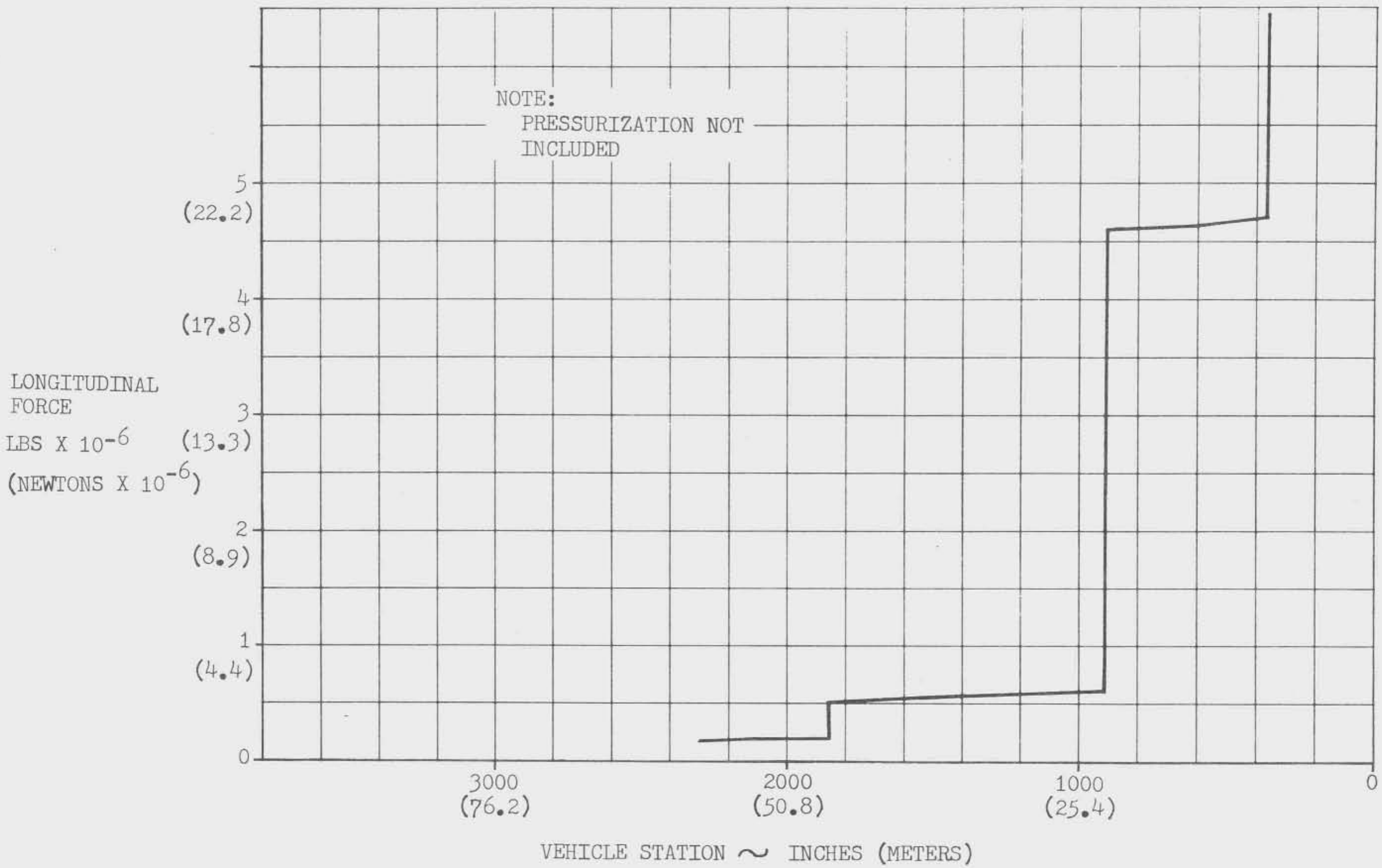


FIGURE 4.1.6.2-4 INT-20 BASELINE VEHICLE LONGITUDINAL FORCE DISTRIBUTION FOR REBOUND (EMERGENCY SHUTDOWN)

TABLE 4.1.6.2-I INT-20 BASELINE N_C LOADS FOR ON-PAD, FUELED, UNPRESSURIZED

		A		B	C				
STATION (IN)	M X 10^{-6} (IN-LB)	M/ R^2 (LB/IN)	PX10^{-6} (LB)	P/2 R (LB/IN)	A + B (LB/IN)	N_C ULT 1.4C (LB/IN)			
365F	68	552	3.4884	2805	3357	4700			
602A	55	447	3.4462	2771	3218	4505			
602F	55	447	3.4419	2768	3215	4501			
912A	40	325	3.4178	2749	3074	4304			
912F	40	325	.4440	357	682	955			
1401A	21	171	.4148	334	505	707			
1401F	21	171	.4100	330	501	701			
1541	16.8	136.4	.4012	323	459	643			
1768	10.6	200	.3900	477	677	948			
1854A	8.7	164	.3885	476	640	895			
1854F	8.7	164	.1466	180	344	481			
2123A	3.6	68	.1399	171	239	335			
2123F	3.6	68	.1391	170	238	333			
2245	2.2	41.4	.1362	167	208.4	291			
2281	2.0	37.7	.1320	162	199.7	279			

TABLE 4.1.6.2-II INT-20 BASELINE N_T LOADS FOR ON-PAD FUELED, UNPRESSURIZED

		A		B	C				
STATION (IN)	M X 10^{-6} (IN-LB)	M/ π R^2 (LB/IN)	P X 10^{-6} (LB)	P/2 π R (LB/IN)	A - B	N_T ULT 1.4 C (LB/IN)			
365F	68	552	3.4884	2805	-2253	-3154			
602A	55	447	3.4462	2771	-2324	-3254			
602F	55	447	3.4419	2768	-2321	-3249			
912A	40	325	3.4178	2749	-2424	-3394			
912F	40	325	.4440	357	- 32	- 45			
1401A	21	171	.4148	334	- 163	- 228			
1401F	21	171	.4100	330	- 159	- 223			
1541	16.8	136	.4012	323	- 187	- 262			
1768	10.6	200	.3900	477	- 277	- 288			
1854A	8.7	164	.3885	476	- 312	- 437			
1854F	8.7	164	.1466	180	- 16	- 22			
2123A	3.6	68	.1399	171	- 103	- 144			
2123F	3.6	68	.1391	170	- 102	- 143			
2245	2.2	41.4	.1362	167	- 125.6	- 176			
2281	2.0	37.7	.1320	162	- 124.3	- 174			

TABLE 4.1.6.2-III INT-20 BASELINE N_C LOADS FOR EMERGENCY SHUTDOWN

		A		B	C	D		E	
STATION ℄ (IN)	$M \times 10^{-6}$ (IN-LB)	M/R^2 (LB/IN)	$P \times 10^{-6}$ (LB)	$P \times 10^{-6}$ (LB/IN)	A + B	Pu MIN (PSIG)	Pu R/2 (LB/IN)	N_C ULT D-E (LB/IN)	
365F	43.5	353	4.710	3780	4033	5646	12.4	1228	4418
602A	35.3	287	4.643	3734	4021	5629	12.4	1228	4401
602F	35.3	287	4.638	3730	4017	5624	0	0	5624
912A	25.5	207	4.609	3703	3910	5474	0	0	5474
912F	25.5	207	.6105	484	691	967	9.3	921	46
1401A	13.5	110	.5703	459	569	797	9.3	921	-124
1401F	13.5	110	.5637	453	563	788	0	0	788
1541	11	90	.5516	444	534	748	0	0	748
1768	7	132	.5362	656	788	1103	0	0	1103
1854A	5.8	109	.5342	654	763	1068	0	0	1068
1854F	5.8	109	.2016	247	356	498	13.3	865	-367
2123A	2.5	47	.1924	236	283	396	13.3	865	-469
2123F	2.5	47	.1912	234	281	393	0	0	393
2245	1.5	28	.1872	229	257	360	0	0	360
2281	1.1	21	.1815	222	243	340	0	0	340

TABLE 4.1.6.2-IV INT-20 BASELINE N_T LOADS FOR EMERGENCY SHUTDOWN

		A		B		C	D		
STATION (IN)	M X 10^{-6} (IN-LB)	M/ R^2 (LB/IN)	P X 10^{-6} (LB)	P/2 R (LB/IN)	PU MAX (PSIG)	PuR/2 (LB/IN)	A+C-B	N_T ULT 1.4 D (LB/IN)	
365F	43.5	353	4.710	3780	16.8	1663	-1764	-2470	
602A	35.3	287	4.643	3734	16.8	1663	-1784	-2498	
602F	35.3	287	4.638	3730	0	0	-3443	-4820	
912A	25.5	207	4.609	3703	0	0	-3496	-4894	
912F	25.5	207	.6105	484	16.8	1663	1386	1940	
1401A	13.5	110	.5703	459	16.8	1663	1314	1840	
1401F	13.5	110	.5637	453	0	0	-343	-480	
1541	11	90	.5516	444	0	0	-354	-496	
1768	7	132	.5362	656	0	0	-524	-734	
1854A	5.8	109	.5342	654	0	0	-545	-763	
1854F	5.8	109	.2016	247	23.0	1515	1377	1928	
2123A	2.5	47	.1924	236	23.3	1515	1326	1856	
2123F	2.5	47	.1912	234	0	0	-187	-262	
2245	1.5	28	.1872	229	0	0	-201	-281	
2281	1.1	21	.1815	222	0	0	-201	-281	

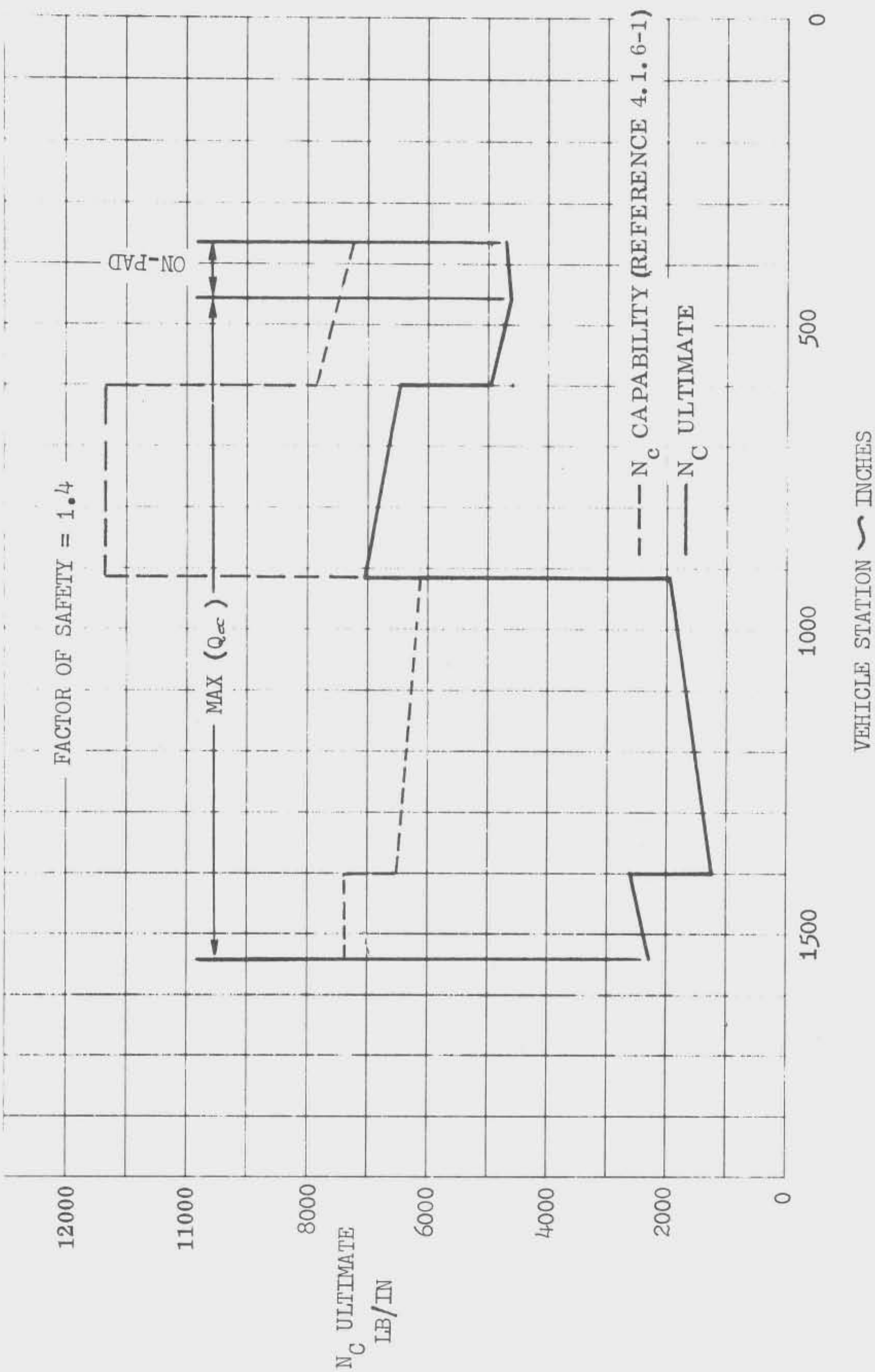


FIGURE 4.1.6.3-1 INT-20 BASELINE VEHICLE S-IC COMBINED COMPRESSIVE LOADS

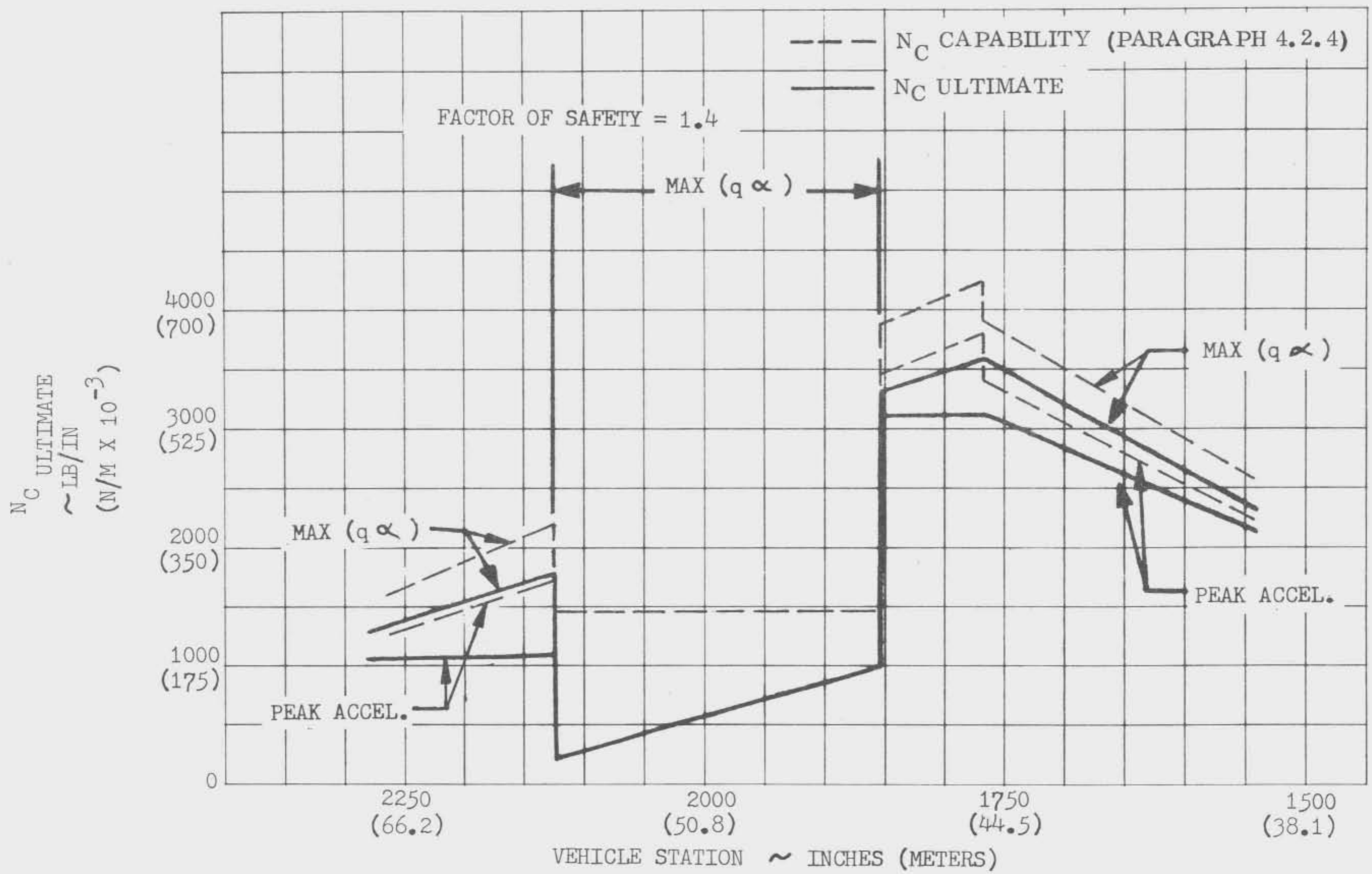


FIGURE 4.1.6.3-2 INT-20 BASELINE VEHICLE COMBINED COMPRESSIVE LOADS DISTRIBUTION FOR IU, S-IVB AND S-IVB/S-IC INTERSTAGE

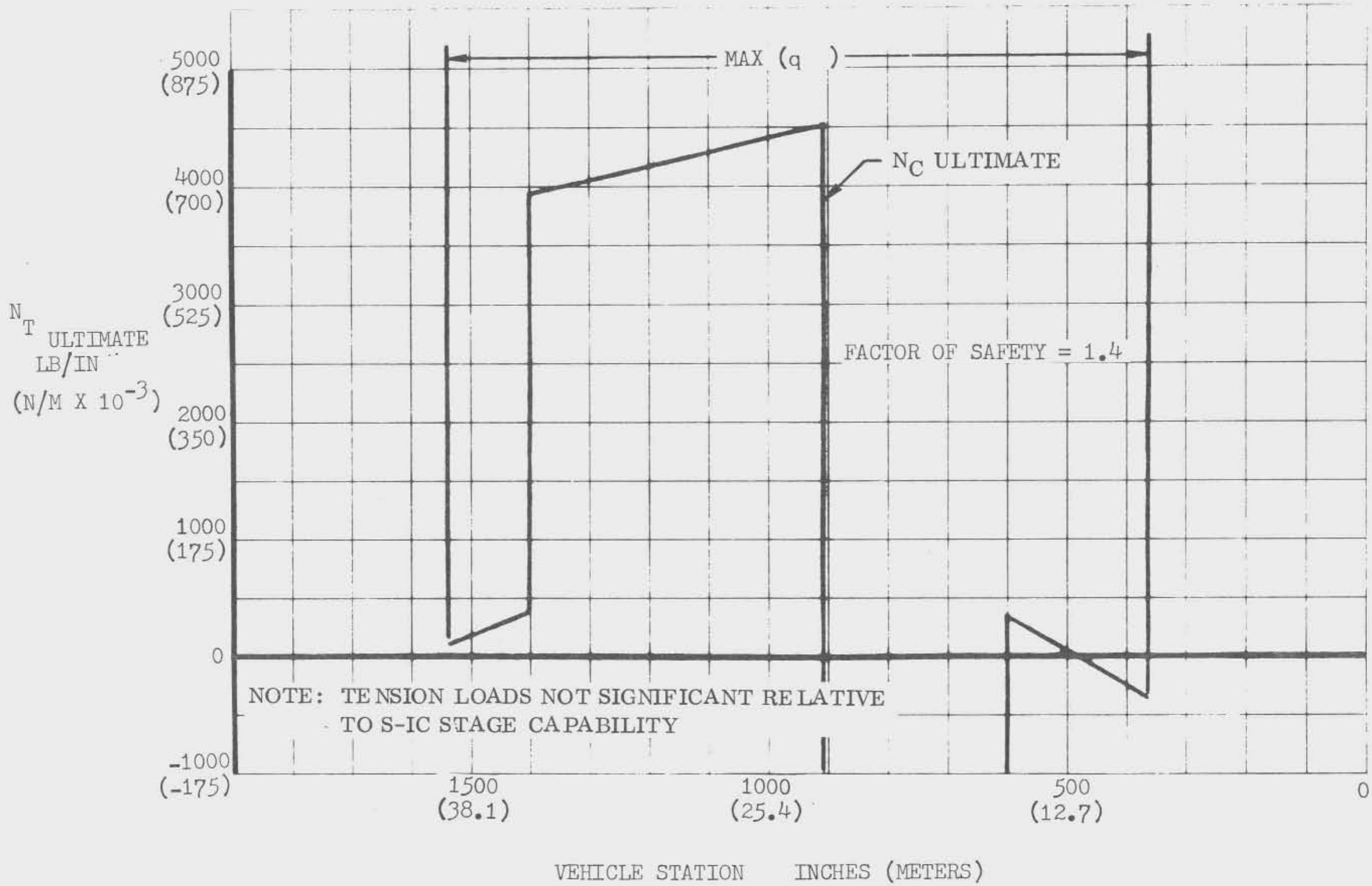


FIGURE 4.1.6.3-3 INT-20 BASELINE VEHICLE S-IC COMBINED TENSION LOADS

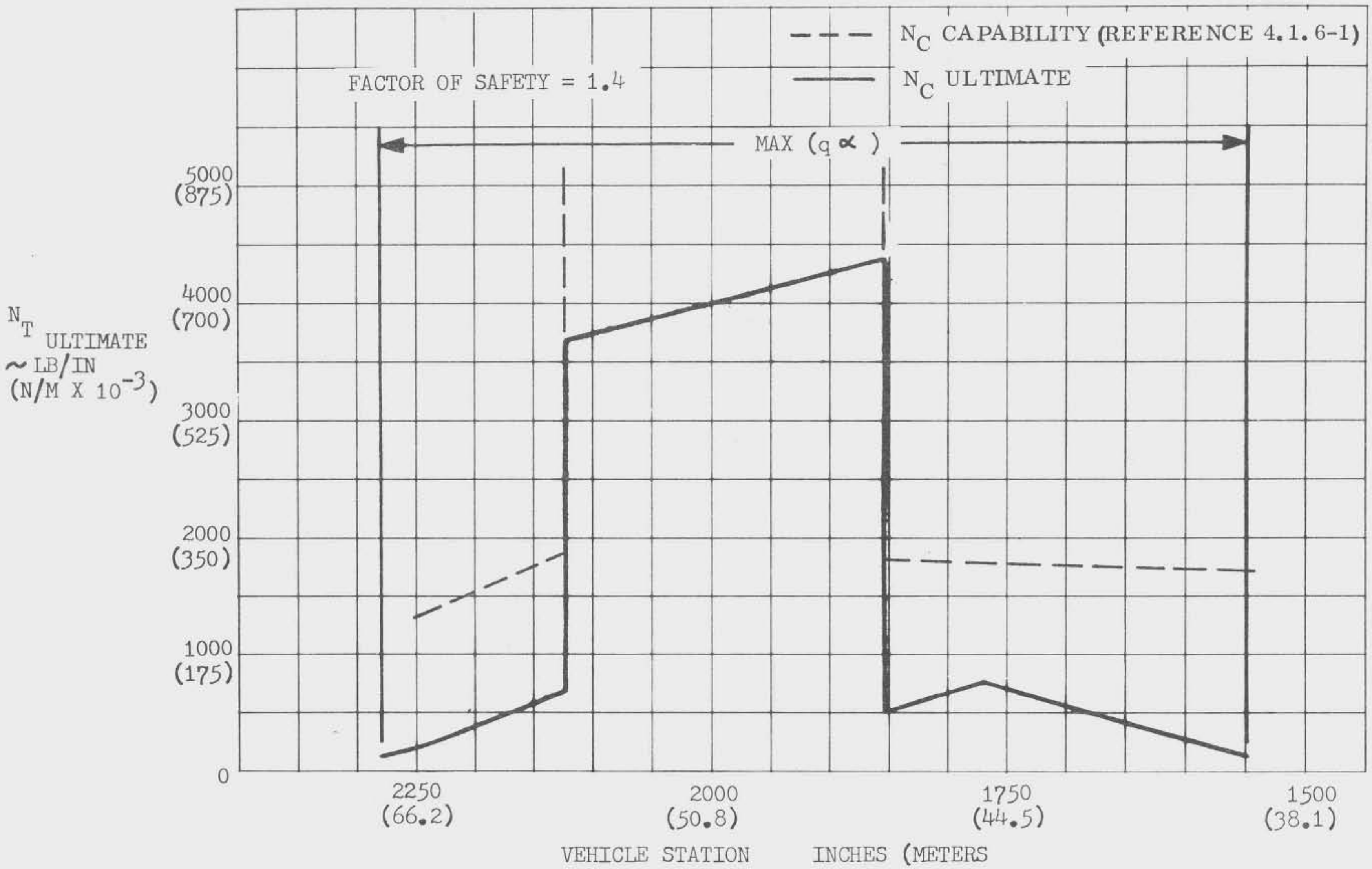


FIGURE 4.1.6.3-4 INT-20 BASELINE VEHICLE COMBINED TENSION LOADS DISTRIBUTION FOR IU, S-IVB AND S-IVB/S-IC INTERSTAGE

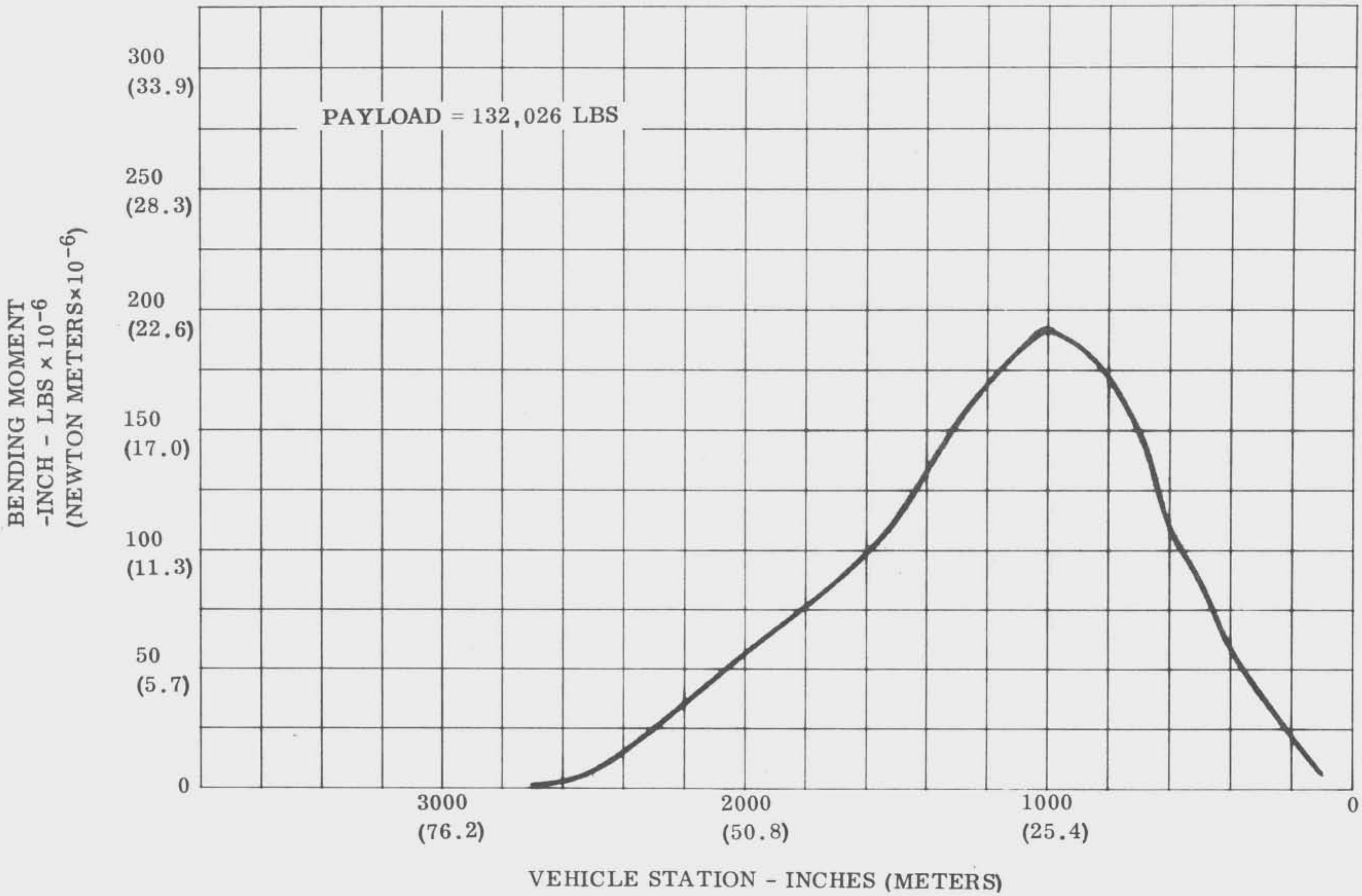


FIGURE 4.1.6.3-5 INT-20 BASELINE VEHICLE BENDING MOMENT DISTRIBUTION AT MAX ($q\alpha$)

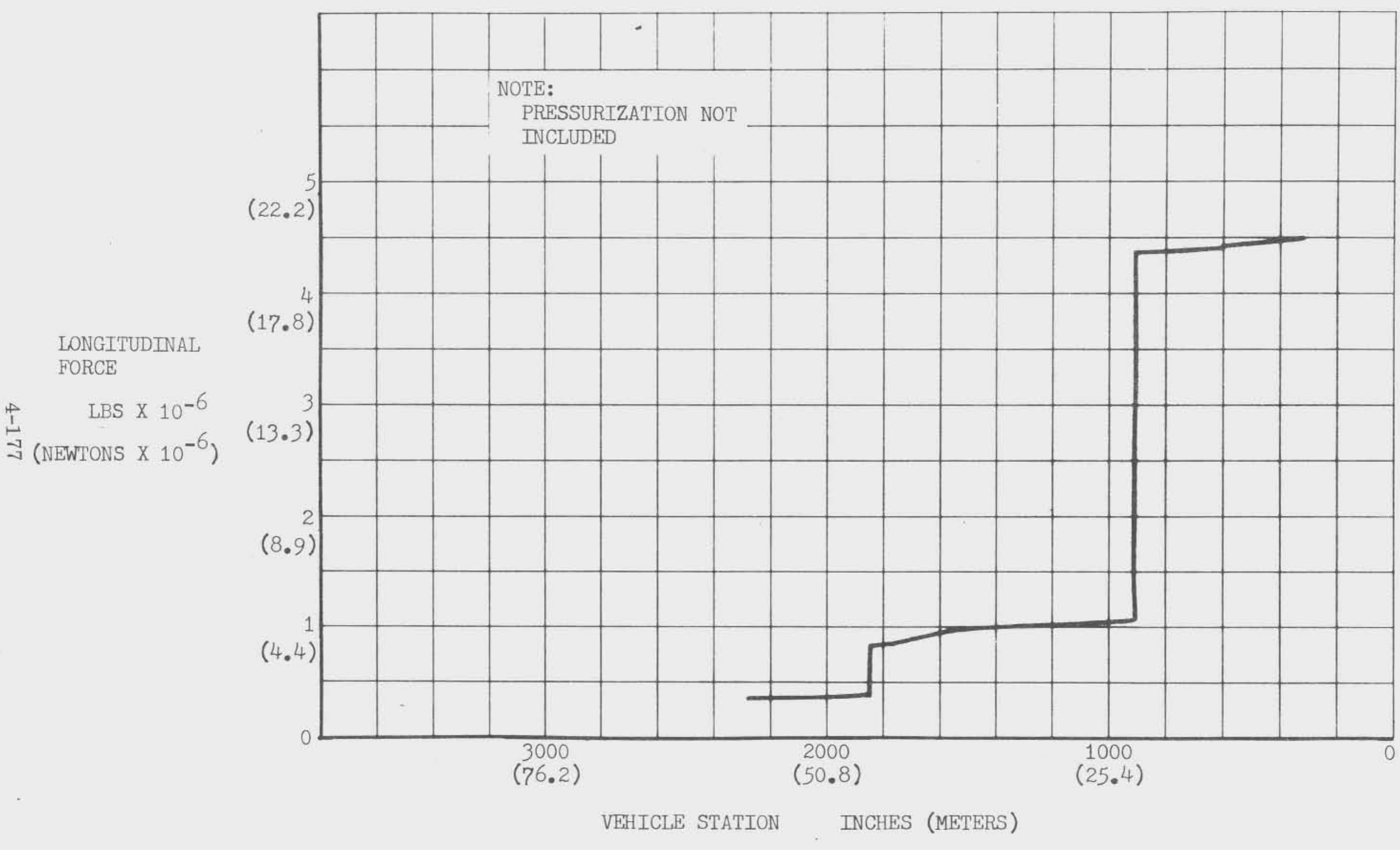


FIGURE 4.1.6.3-6 INT-20 BASELINE VEHICLE LONGITUDINAL FORCE DISTRIBUTION AT MAX (q_α)

FIGURE 4.1.6.3-7 INT-20 BASELINE VEHICLE LONGITUDINAL FORCE DISTRIBUTION AT PEAK ACCELERATION (t=146 Sec.)

TABLE 4.1.6.3-I INT-20 BASELINE VEHICLE N_c LOADS CALCULATIONS MAX (Q_α)

Station (in)	$MX10^{-6}$ (in. lb)	A $M/\pi R^2$ (lb/in)	$PX10^{-6}$ (lb)	B $P/2\pi R$ (lb/in)	C A + B (lb/in)	D 1.4 C (lb/ in)	P_U (min) (psig)	E $P_U R/2$ (lb/in)	D − E N_c ULT (lb./in)
365A			5.9577						
365F	76	617	4.4983	3,617.6	4,234.6	5,928	15.6	1,544	4,384
602A	133	1,080	4.4163	3,551.7	4,631.7	6,484	15.6	1,544	4,940
602F	133	1,080	4.4084	3,545.3	4,625.3	6,475			6,475
912A	188	1,526	4.3589	3,505.5	5,031.5	7,044			7,044
912F	188	1,526	1.0621	854.2	2,380.2	3,332	14.1	1,396	1,936
1401A	132	1,072	.9990	803.4	1,875.4	2,626	14.1	1,396	1,230
1401F	132	1,072	.9901	796.2	1,868.2	2,615			2,615
1541	107	869	.9730	782.5	1,651.5	2,312			2,312
1768	82	1,544	.8365	1,023.9	2,567.9	3,595			3,595
1854A	72.5	1,366	.8233	1,007.7	2,373.7	3,323			3,323
1854F	72.5	1,366	.3733	456.9	1,822.9	2,552	24.1	1,566.5	985
2123A	44.4	836	.3567	436.6	1,272.6	1,782	24.1	1,566.5	215
2123F	44.4	836	.3552	434.8	1,270.8	1,779			1,779
2245	30.5	574	.3481	426.1	1,000.1	1,400			1,400
2281	26.8	505	.3397	415.8	920.8	1,289			1,289

TABLE 4.1.6.3-II INT-20 BASELINE VEHICLE N_T LOADS CALCULATIONS AT MAX ($Q_{\infty C}$)

		A		B	C		E	N_T ULT
STATION	M X 10^{-6}	$M/\pi R^2$	P X 10^{-6}	$P/2\pi R$	A - B + E	P_U (MAX)	$P_U R/2$	1.4 C
(IN)	(IN-LB)	(LB/IN)	(LB)	(LB/IN)	(LB/IN)	(psig)	(LB/IN)	(LB/IN)
365F	76	617	4.4983	3,617.6	-263.2	27.65	2,737.4	-368
602A	133	1080	4.4163	3,551.7	255.8	27.65	2,737.4	358
602F	133	1080	4.4084	3,545.3	-2465.3			-3451
912A	188	1526	4.3589	3,505.5	-1979.5			-2771
912F	188	1526	1.0621	854.2	3221.5	25.5	2,549.7	4510
1401A	132	1072	.9990	803.4	2818.3	25.5	2,549.7	3945
1401F	132	1072	.9901	796.2	275.8			386
1541F	107	869	.9730	782.5	86.5			121
1768	82	1544	.8365	1,023.9	520.1			728
1854A	72.5	1366	.8233	1,007.7	358.3			502
1854F	72.5	1366	.3733	456.9	3128.9	34.15	2219.8	4380
2123A	44.4	836	.3567	436.6	2619.2	34.15	2219.8	3667
2123F	44.4	836	.3552	434.8	401.2			562
2245	30.5	574	.3481	426.1	147.9			207
2281	26.8	505	.3397	415.8	89.2			125

TABLE 4.1.6.3-III BASELINE VEHICLE N_c LOADS CALCULATIONS AT PEAK ACCELERATION (4.68 g's at t = 146 sec)

		A		B	C	D		E	D - E
Station	$M \times 10^{-6}$	$M/\pi R^2$	$P \times 10^{-6}$	$P/2\pi R$	A + B	1.4 C	P_U (min)	$P_U R/2$	N_c ULT
(in)	(in-lb)	(lb/in)	(lb)	(lb/in)	(lb/in)	(lb/in)	(psig)	(lb/in)	(lb./in)
365F	0	0	4.978	4003	4003	5604	19.5	1931	3673
602A	0	0	4.780	3844	3844	5382	19.5	1931	3451
602F	0	0	4.760	3828	3828	5359	-	-	5359
912A	0	0	4.648	3738	3738	5233	-	-	5233
912F	0	0	2.078	1671	1671	2340	18.0	1782	558
1401A	0	0	1.941	1561	1561	2186	18.0	1782	404
1401F	0	0	1.919	1543	1543	2160	-	-	2160
1541	0	0	1.877	1510	1510	2114	-	-	2114
1768	0	0	1.825	2234	2234	3128	-	-	3128
1854A	0	0	1.818	2225	2225	3115	-	-	3115
1854F	0	0	.686	840	840	1176	28.0	1820	-644
2123A	0	0	.655	803	803	1124	28.0	1820	-696
2123F	0	0	.651	797	797	1115	-	-	1115
2245	0	0	.637	780	780	1093	-	-	1093
2281	0	0	.618	756	756	1059	-	-	1059

TABLE 4.1.6.3-IV INT-20 BASELINE VEHICLE N_T LOADS CALCULATIONS AT PEAK ACCELERATION (4.68 g's AT t = 146 SEC)

		A		B		C	D	1.4D
STATION	$M \times 10^{-6}$	$M/\pi R^2$	$P \times 10^{-6}$	$P/2\pi R$	P_U (MAX)	$P_U R/2$	A - B + C	N_T ULT
(IN)	(IN - LB)	(LB/IN)	(LB)	(LB/IN)	(PSIG)	(LB/IN)	(LB/IN)	(LB/IN)
365F	0	0	4.978	4003	31.5	3118.5	-885	-1239
602A	0	0	4.780	3844	31.5	3118.5	-726	-1016
602F	0	0	4.760	3828	0	0	-3828	-5359
912A	0	0	4.648	3738	0	0	-3738	-5233
912F	0	0	2.078	1671	25.5	2524.5	854	1196
1401A	0	0	1.941	1561	25.5	2524.5	964	1350
1401F	0	0	1.919	1543	0	0	-1543	-2160
1541	0	0	1.877	1510	0	0	-1510	-2114
1768	0	0	1.825	2234	0	0	-2234	-3128
1854A	0	0	1.818	2225	0	0	-2225	-3115
1854F	0	0	.686	840	38.0	2470.0	1630	2282
2123A	0	0	.655	803	38.0	2470.0	1667	2334
2123F	0	0	.651	797	0	0	-797	-1115
2245	0	0	.637	780	0	0	-780	-1093
2281	0	0	.618	756	0	0	-756	-1059

4.1.6.4 Tank Loads

The propellant tanks are designed for compression and tension loads on the sidewalls imposed by vehicle bending moments and total tank pressure. Total tank pressure is a summation of ullage pressure, liquid head and ambient pressure. Liquid head is influenced by vehicle axial acceleration during flight.

The S-IC stage sidewall (forward skirt, LOX tank sidewall, intertank, and fuel tank sidewall) was found acceptable for the combined compressive (N_c) loads shown in paragraph 4.1.6.3. Tank bottom pressures are acceptable for the baseline payload of 132,000 pounds (59870 kg). Investigations of a vehicle with a lower payload, 50,000 pounds (22680 kg), which had an attendant increase in ballast, showed tank capability to be exceeded (see Appendix A-2). Also, the S-IC propellant tank lower bulkheads are subject to hoop compression loads that necessitate limiting axial acceleration during flight for the retrofit configuration (see subparagraph b, below).

a. Design Tank Pressures

1. The S-IC stage design tank pressures are shown in Appendix A 2, Figures A-17 and A-18.

2. The S-IVB stage design tank pressures are shown in Figure 4.1 6.4-1.

b. Hoop Compression

The hoop compression condition is described in Appendix A-2 and illustrated in Figure A-28. Briefly, two types of loads are experienced by the bulkheads tension and hoop compression. The bulkhead tends to deform, as shown, under the combined influence of low propellant level and high acceleration. The lower portion of the bulkhead (apex gore) experiences only a tension load, but the upper part (base gore) is loaded in both longitudinal tension and hoop compression. The hoop compression allowables are exceeded in the critical tanks only when the fluid level is below the general area of the lower Y-ring. (When the fluid level is higher than this, sufficient fluid pressure is applied to the base gore to reduce hoop compression deformation to below allowables.) This condition was alleviated by restricting maximum vehicle axial acceleration to 3.68 g's at first two engine cut-off and 4.68 g's at final two engine (or S-IC) cutoff.

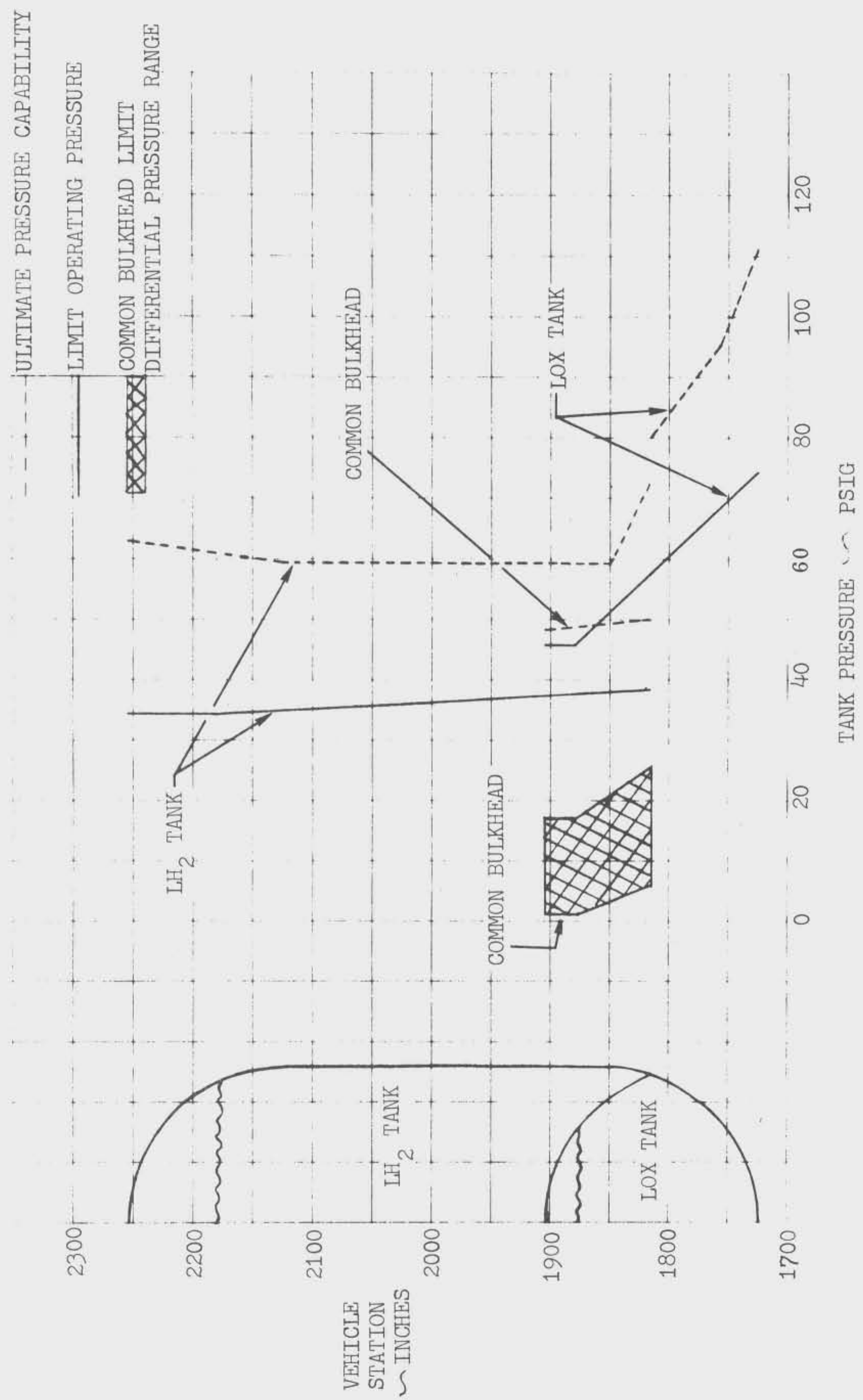


FIGURE 4.1.6.4-1 INT-20 BASELINE VEHICLE S-IVB TANK PRESSURE

4.1.7 Vehicle Mass Properties

The INT-20 baseline vehicle weights and mass characteristics were derived from the SA-511 vehicle described in Reference 4.1.7-1. General changes in vehicle stages were as follows:

a. S-IC Stage

The S-IC stage dry weight was decreased by removal of the center F-1 engine and associated propellant delivery systems. A complete description of the changes is contained in Section 4.2.2.

b. S-IC/S-IVB Interstage

The S-IC/S-IVB Interstage was modified by deletion of the retro-motors and ancillary equipment, and by the addition of insulating material. A complete description of the changes made is contained in Section 4.2.3. The overall result was a weight decrease.

c. S-IVB Stage

The S-IVB dry weight was decreased by removal of the restart capability (reversible). Other changes are described in Section 4.2.4.

d. Instrument Unit (IU)

The IU weight was increased by addition of vibration damping material mainly in the area of ST-124 platform. The IU modifications are described in Section 4.2.5.3.

4.1.7.1 Baseline Weights

Vehicle baseline weights are summarized for the S-IC stage, the S-IC/S-IVB interstage, the S-IVB stage, and the Instrument Unit (IU) in Tables 4.1.7.1-I through IV.

Drop weights at staging during flight are shown in Table 4.1.7.1-II.

4.1.7.2 Mass Distributions and Inertias

Vehicle mass and moment of inertia data were calculated using the basic SA-511 vehicle described in Reference 4.1.7-1. The data are contained in Appendix D-4 and include the following:

4.1.7.2 (Continued)

S-IC propellant depletion rates
Mass distribution and associated cantilevered masses
Propellant distribution
Vehicle accumulated weights
Flight time histories of vehicle weight, cg, and roll and pitch mass moments of inertia.

These data were used in vehicle technical design.

TABLE 4.1.7.1-II
S-IC/S-IVB INTERSTAGE WEIGHT SUMMARY

NASA SECOND GENERATION BREAKDOWN		S-IVB-511 BASELINE	S-IVB-INT-20 BASELINE
W3.13	Interstage Structure	5678	5678
W3.15	Paint and Sealer	49	49
W3.18	Heat & Flame Protection	523	523
W3.0	Interstage Structure	6250	6250
W6.2	Environ. Control System	17	17
W6.8	Telemetry and Measuring Sys.	15	15
W6.12	Range Safety Sys.	2	2
W6.17	Separation Sys.	727	257
W6.20	Systems for Total Vehicle	10	10
W6.0	Equipment and Instrumentation	771	301
WBD	INTERSTAGE DRY	7021	6457
	SERVICE ITEMS	1062	0
	INTERSTAGE AT GRD. IGN.	8083	6457

TABLE 4.1.7.1-I
S-IC STAGE WEIGHT SUMMARY

DESCRIPTION	S-IC-511 BASELINE		S-IC-INT-20 BASELINE	
STAGE STRUCTURE		140,656 (lb)		140,660 (lb)
STRUCTURAL FUEL CONTAINER	22,407		22,735	
STRUCTURAL OXIDIZER CONTAINER	35,447		35,502	
STRUCTURE FORWARD OF TANKS	5,200		5,200	
STRUCTURE BETWEEN TANKS	13,194		13,194	
THRUST STRUCTURE	47,503		47,186	
FAIRINGS AND ASSOCIATED STRUCTURE	9,054		9,054	
NON-MOVEABLE AERO CONTROL SURFACES	2,035		2,035	
BASE HEAT PROTECTION	5,350		5,288	
PAINT AND SEALER	466		466	
PROPULSION AND SYSTEM AND ACCESSORIES		139,424 (lb)		117,186 (lb)
LIQUID ROCKET ENGINE AND ACCESSORIES	93,734		74,462	
FUEL SYSTEM	13,405		12,743	
OXIDIZER SYSTEM	23,713		21,492	
STAGE CONTROL SYSTEM	8,572		8,489	
EQUIPMENT AND INSTRUMENTATION		8,636 (lb)		8,751 (lb)
STRUCTURE (FOR EQUIP AND INSTRUMENTATION)	225		225	
ENVIRONMENTAL CONTROL SYSTEM	314		314	
GUIDANCE SYSTEM	29		29	
TELEMETERING AND MEASURING EQUIP	3,908		3,982	
ELECTRICAL SYSTEM	713		747	
RANGE SAFETY EQUIPMENT	497		497	
SEPARATION SYSTEM	2,498		2,501	
PNEUMATIC SYSTEM	432		432	
CONTROL SYSTEMS ELECT.	20		24	
TOTAL DRY WEIGHT		288,716 (lb)		266,597 (lb)

TABLE 4.1.7.1-IV
INSTRUMENT UNIT WEIGHT SUMMARY

DESCRIPTION	WEIGHT (LB)
STAGE & STRUCTURE	708.5
INSTRUMENTATION	3276.7
SERVICE ITEMS	298.5
TOTAL	4283.7

TABLE 4.1.7.1-III
S-IVB STAGE WEIGHT SUMMARY

NASA SECOND GENERATION BREAKDOWN	S-IVB-511 J-2 BASELINE	S-IVB-INT-20 J-2 BASELINE
W3.3 Propellant Container	8933	8933
W3.6 Forward of Tanks	1242	1242
W3.8 Aft of Tanks	1816	1816
W3.9 Thrust Structure	774	774
W3.10 Fairings & Assoc. Struct.	197	197
W3.15 Paint & Sealer	104	104
W3.18 Heat & Flame Protection	182	67
W3.0 Structure	13,248	13,133
W4.1 Engine & Accessories	3572	3572
W4.6 Purge System For Chilldown	272	272
W4.7 Fuel System	1573	908
W4.8 Oxidizer System	1264	998
W4.9 Cryogenic Repress. System	310	254
W4.10 Stage Control Sys. Hdwe.	284	284
W4.0 Propulsion System	7275	6288
W6.1 Equip. & Instru. Struct.	430	430
W6.2 Environ. Control System	231	231
W6.5 Control System Electron.	116	116
W6.8 Telemetry & Meas. System	1165	1165
W6.10 P.U. System	175	175
W6.11 Electrical System	829	829
W6.12 Range Safety System	69	69
W6.15 Pneumatic System	298	298
W6.16 Auxiliary Prop. Sys.	855	832
W6.17 Separation System	117	117
W6.18 Ullage System	212	212
W6.20 Systems for Total Vehicle	91	91
W6.0 Equipment & Instrumentation	4588	4565
WAD STAGE DRY WEIGHT	25,111	23,986

TABLE 4.1.7.1-V

BASELINE INT-20 DROP WEIGHTS

DESCRIPTION	LBS
TOTAL WEIGHT DROP @ S-IC STAGING	340,309
S-IC (Dry)	(266,597)
S-IC Residuals	(59,765)
LOX in Tank	1,861
LOX Below Tank	20,590
LOX Pressurization Gas	5,451
Fuel in Tank	17,119
Fuel Below Tank	10,694
Fuel Pressurization Gas	512
Helium in Bottle	188
Service Items	3,350
S-IC Thrust Decay	(6,746)
LOX Outboard Engine T.D.	4,684
Fuel Outboard Engine T.D.	2,062
S-IC/S-IVB Interstage (Dry)	(6,457)
S-IVB Ullage Rocket Prop.	(257)
S-IVB Idle Mode	
LOX Idle Mode	
LH_2 Idle Mode	
S-IVB Thrust Buildup	(436)
LOX Thrust Buildup	312
LH_2 Thrust Buildup	124
S-IVB Separation Package	(51)

S-IC (LOX) Thrust Buildup 55,832

S-IC (Fuel) Thrust Buildup 24,598

TABLE 4.1.7.1-V (continued)

DESCRIPTION	LBS
S-IVB DROP WEIGHT @ S-IVB STAGING	26,629
S-IVB (Dry)*	(23,986)
S-IVB Residuals	(2,512)
LOX In Tank	104
LOX Below Tank	369
LOX Pressurization Gas	347
Fuel In Tank	843
Fuel Below Tank	45
Fuel Pressurization Gas	553
Helium In Bottle	189
APS Propellant	
Service Items	62
S-IVB Thrust Decay	(131)
LOX Thrust Decay	91
LH_2 Thrust Decay	40
S-IVB Idle Mode Prop.	
LOX Idle Mode	
LH_2 Idle Mode	
INSTRUMENT UNIT DROP WEIGHT	4,284

*Less Sep. Package (51#)
and Ullage Rocket Cases (130#)

4.1.8 Vehicle Propulsion Systems

The primary propulsion engines for the INT-20 are the 4 F-1 engines of the S-IC stage and one J-2 engine for the S-IVB stage. In addition, the S-IC stage is fitted with 8 retromotors used for staging. An attitude control thrusting system is provided for the S-IVB stage. Since the S-IC retrorockets are used for S-IC/S-IVB separation, the standard S-IVB retrorockets are omitted for the INT-20.

4.1.8.1 Propulsion Data

a. S-IC Stage

1. F-1 Engine

The F-1 engines are the same as those used for the Saturn V/S-IC stage. The center engine of the S-IC stage is omitted for the 4 F-1 configuration (see Figure 4.1.8.1-1). The four remaining outboard engines are gimballed to provide pitch, yaw, and roll control during flight. The propellants are liquid oxygen (LOX) as oxidizer and RP-1 as fuel, burned at a nominal mixture ratio of 2.27:1. Nominal sea level thrust is 1,522,000 pounds (6,770,000 newtons) per engine and nominal sea level specific impulse is 263.58 seconds.

The INT-20 engine cutoff sequence is 2-2: engines 2 and 4 shut down first to limit axial acceleration, and engines 1 and 3 shut down together at final cutoff (because either the axial acceleration limit is reached again or the LOX supply is depleted). For the baseline trajectory, the first pair of F-1 engines was cut off at 146 seconds and the second pair was cut off at 211 seconds. First cutoff and final cutoff were made at 129 and 228 seconds, respectively, for the retrofit trajectory (see Section 4.1.1.4).

The allowable engine centerline drift of each F-1 from the nominal condition will be the same as for Saturn V:

(a) From cutoff signal to 10% mainstage thrust, drift shall not exceed 1.5 degrees.

(b) From 10% to zero mainstage thrust, drift will possibly be to the corner travel limit of the actuator (7 degrees $\pm$ 0.5 degree.)

4.1.8.1 (Continued)

The non-gimballed F-1 engine thrust vector should be aligned with the vehicle centerline. Allowable F-1 engine misalignment is $\pm$ 0.442 degrees. Engine thrust imbalance can be up to $\pm$ 1.5 percent of nominal. Figure 4.1.8.1-2 presents estimated F-1 engine nominal thrust decay at altitude. The 3-sigma limits about the nominal are also shown. An averaged outboard engine thrust-decay trace, derived from AS-501, AS-502, and AS-503 flight measurements, is shown for reference in Figure 4.1.8.1-3.

2. Extended F-1 Engine Burn Time

The extended F-1 engine operating time (up to about 230 seconds compared with about 160 seconds for Saturn V) is feasible. NAR-Rocketdyne states that the projected F-1 engine operating time is as long as 340 seconds - continuous duration - without engine modification. However, it is recommended (see Section 5.2) that a long-duration (230 seconds) test be made on an engine test stand at MSFC. The turbopump has been demonstrated satisfactorily for durations up to 300 seconds. During an engine firing, the turbopump bearing temperature increases with time. Equilibrium conditions are not reached but the rate of temperature increase decreases with time. Based on an extrapolation of test results (see Figure 3.1.1.4-1), the maximum allowable (redline) bearing temperature should not be reached within 340 seconds.

3. F-1 Engine Propulsion and Mechanical Subsystems

Changes in S-IC propulsion and mechanical subsystems resulting from the deletion of an F-1 engine are described in section 4.2.2.1, b.

4. Retromotors

Eight retrorocket motors are used in staging the S-IC. A pair of motors is located in each of the four engine fairings. The motor, excluding end brackets, is approximately 86 inches long. Specification thrust characteristics of the retromotors are shown in Figure 4.1.8.1-4 for motor temperatures of + 30^{o} F, + 70^{o} F, and + 120^{o} F. The actual measured retromotor effective thrust, from AS-501, AS-502, and AS-503 flight data, is shown for reference in Table 4.1.8.1-I.

4.1.8.1 (Continued)

b. S-IVB Stage

1. J-2 Engine

The single J-2 engine is the same as that used for the Saturn V application. The engine is gimballed to provide pitch and yaw control during flight. The propellants are liquid oxygen and liquid hydrogen burned at nominal mixture ratio of 5:1 (oxidizer: fuel). Nominal vacuum thrust is 205,000 pounds (911,840 newtons), with a vacuum specific impulse of 426 seconds. Nozzle area ratio is 27.5:1. In the normal Saturn V/S-IVB stage configuration, the J-2 engine has the capability for one restart. The INT-20 requires only a single burn so restart capability was deleted. This does not result in changes to the engine itself, but some simple system modifications are recommended. Propulsion system changes are described in section 4.2.4.3.

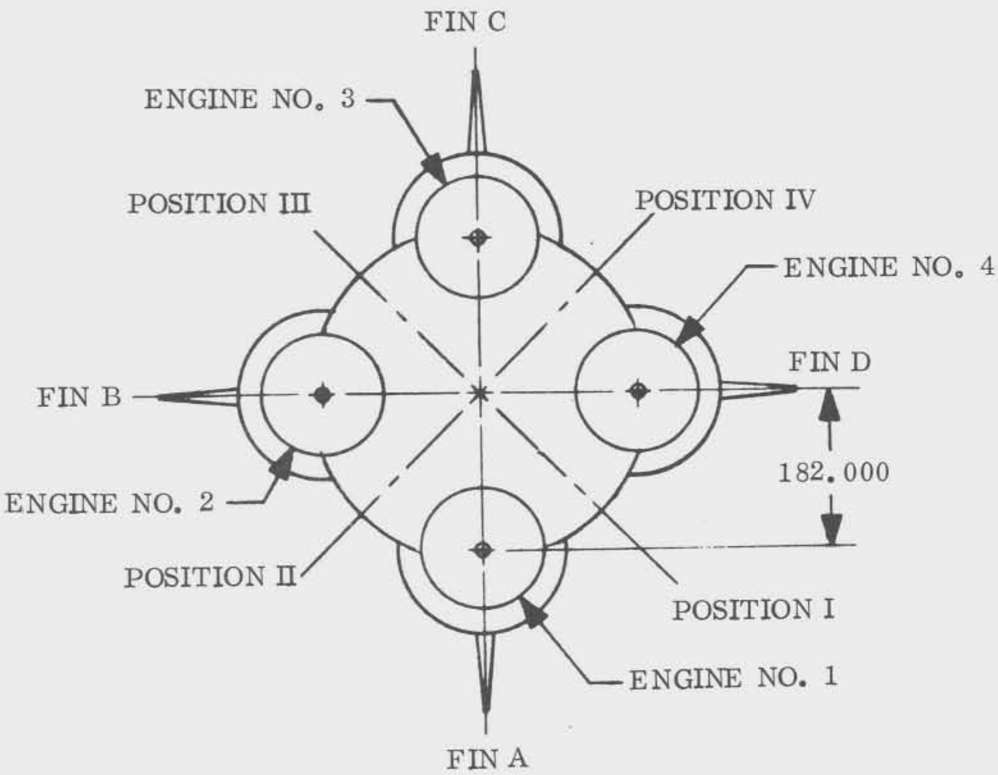
2. Auxiliary Propulsion System (APS)

The Saturn V/S-IVB APS is used for the INT-20 application. The attitude control engines of the APS provide control for the three axes of the vehicle during the coast and roll control during stage burn. The ullage engines used for propellant settling at J-2 restart are deleted for the INT-20. The APS is described in section 4.2.4.3.

4.1.8.2 Fluid Systems Requirements

The S-IC stage fluid power system, together with the thrust vector system, make up the flight control system. Design changes required for the fluid power system consist of deletion of the center engine ground hydraulic supply and return ducting and capping the center engine branches on the supply and return duct manifolds (see Section 4.2.2.1). There will be no changes to the thrust vector control system.

There are no changes to the basic fluid power system of the S-IVB stage. However, some modifications to the J-2 engine control helium supply and deletion of the continuous vent system are recommended (see Section 4.2.4.3).



REAR VIEW

FIGURE 4.1.8.1-1 INT-20 BASELINE VEHICLE F-1 ENGINE ARRANGEMENT

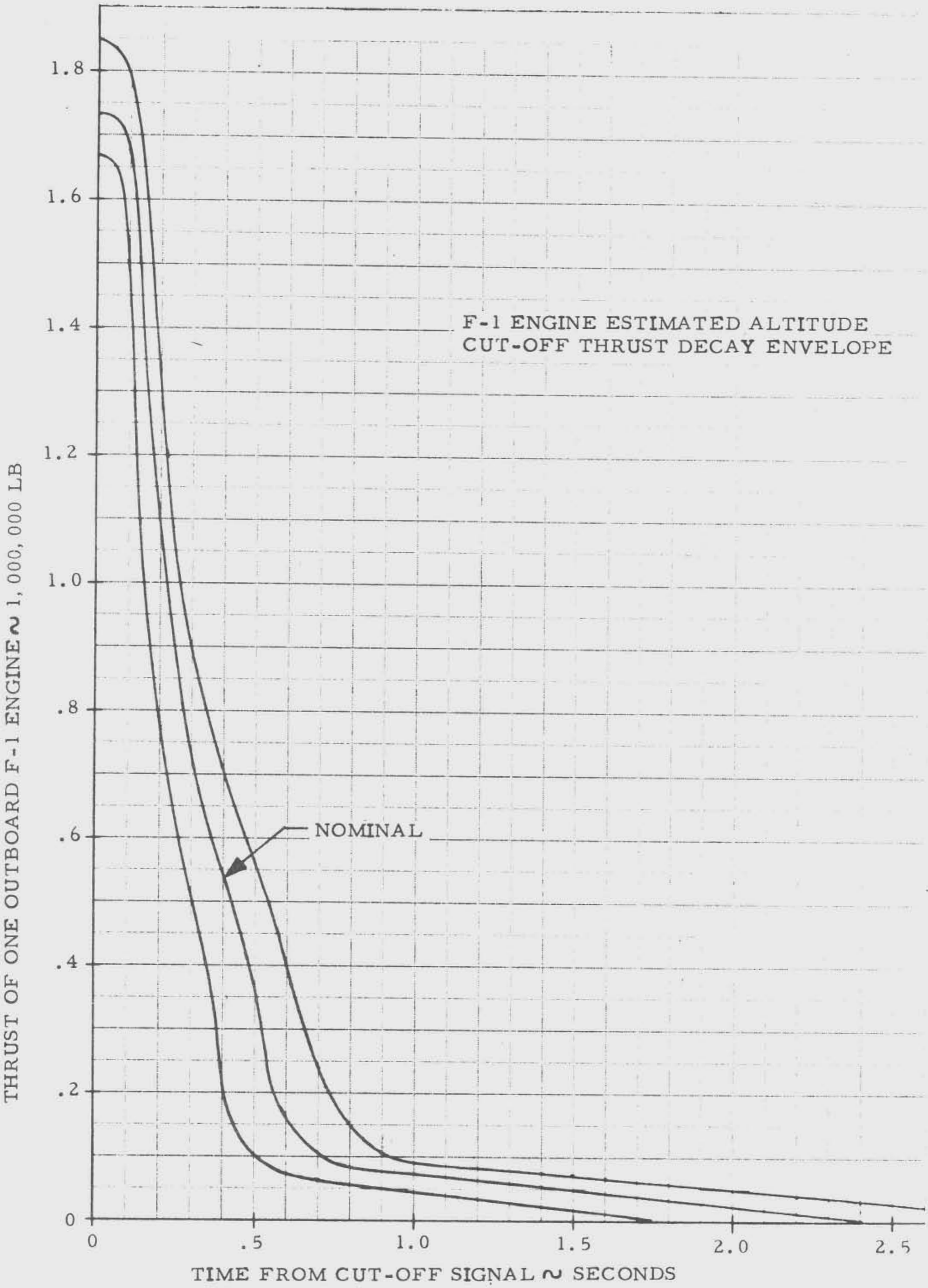


FIGURE 4.1.8.1-2 ESTIMATED F-1 ENGINE THRUST DECAY

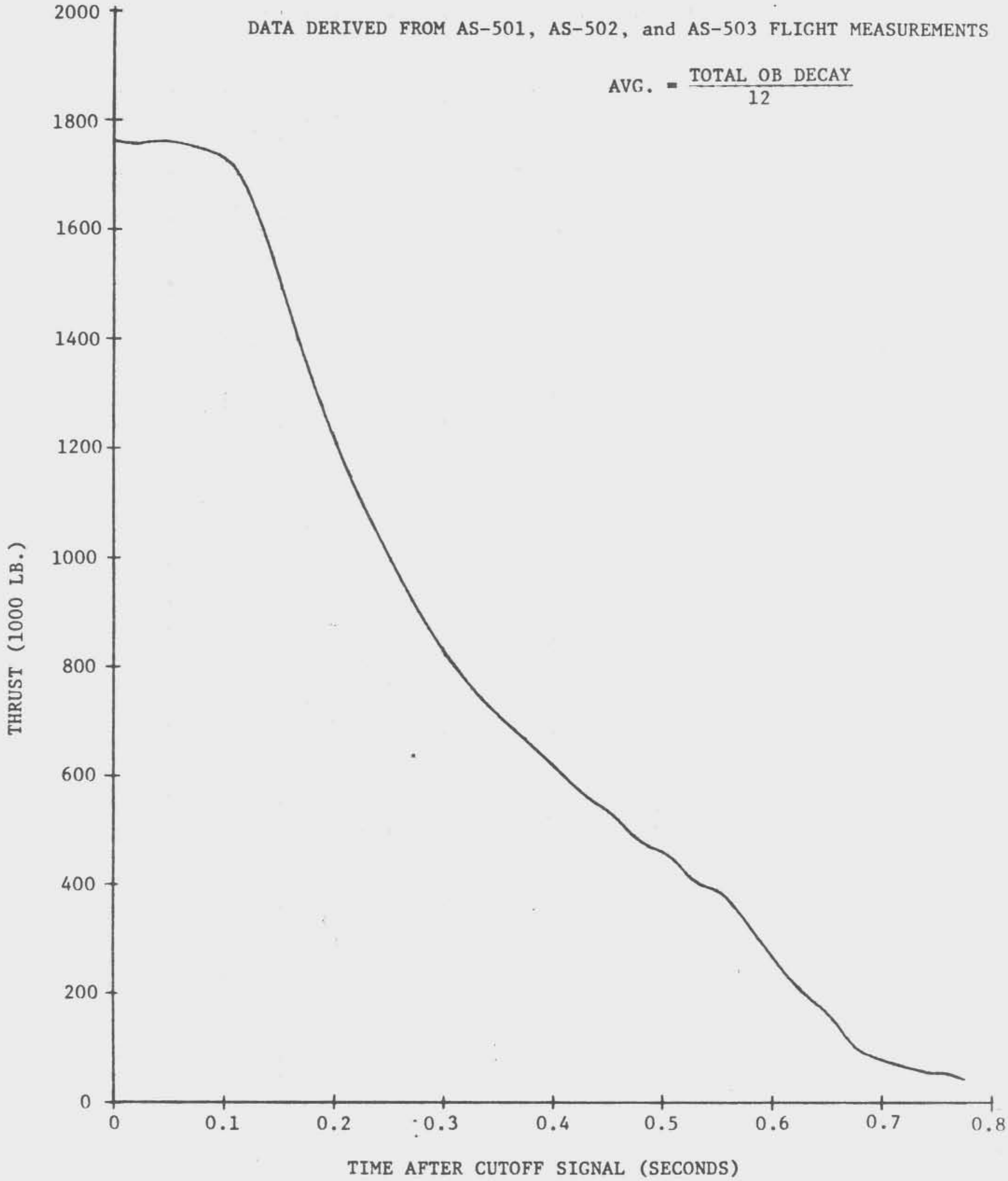


FIGURE 4.1.8.1-3 AVERAGED OUTBOARD F-1 ENGINE THRUST DECAY

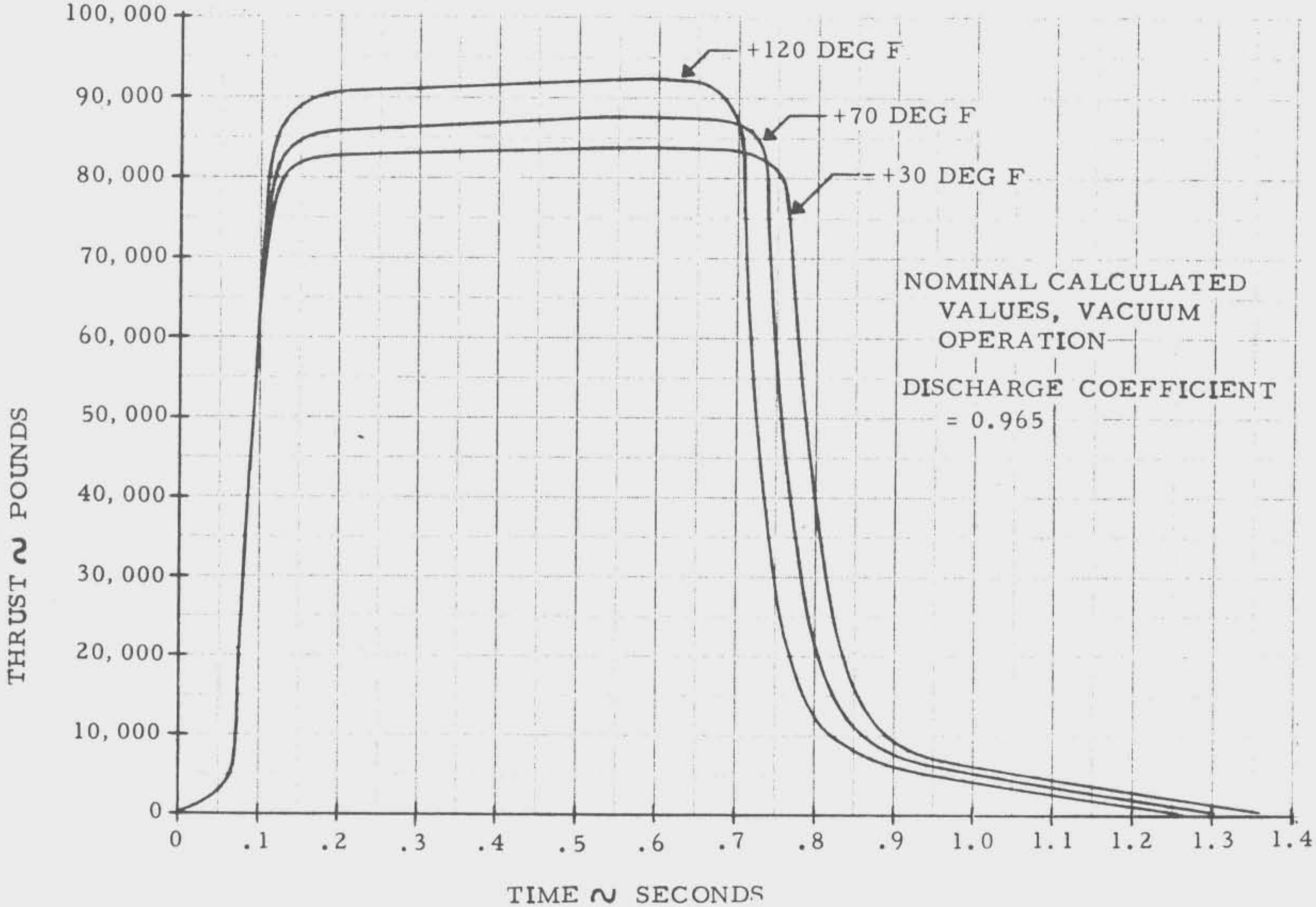


FIGURE 4.1.8.1-4 S-IC RETROMOTOR (TE-424) THRUST HISTORY

TABLE 4.1.8.1-I FLIGHT DATA--MEASURED RETROMOTOR THRUST

	RETROMOTOR AVERAGE EFFECTIVE THRUST ~ POUNDS							
	FIN A		FIN B		FIN C		FIN D	
POSITION	I	II	II	III	III	IV	IV	I
AS-501	92378	89188	96446	92100	90065	91697	89465	92731
AS-502	89691	91053	91238	88824	90587	87231	91495	89645
AS-503	84905	83870	84926	84725	80446	85534	86449	85534

4.1.9 Safety and Abort

The payload was undefined for the Baseline INT-20 vehicle (the MLV shape was used) and no launch escape system (LES) was provided. The emergency detection system (EDS) is unchanged, although requirements for EDS monitoring are reduced (S-IC engine No. 5 and S-II stage are omitted). The EDS ignores the absence of the S-IC engine and the S-II stage is not included in the sequence of events (see Section 4.2.5.3).

a. S-IC Stage Engine Out

1. Engine Cutoff Sequencing

The IU software functions will be revised to provide a normal or reversed engine cutoff sequence. Normally, engine "g" limit cutoff commands are provided to engines 2 and 4 at approximately 146 seconds and 211 seconds for engines 1 and 3 (baseline). In the event that either engine 1 or 3 is cutoff prior to engines 2 and 4, the IU will reverse cutoff sequence causing the remaining engine 1 or 3 to cutoff at 146 seconds allowing engines 2 and 4 to continue burning until cutoff prior to possible propellant depletion (see Section 4.2.5.3).

2. S-IC Engine-Out Control

A brief analysis of post-engine-out control capability of the INT-20 vehicles was made for comparison with Saturn V capability (see Section 4.1.4.1). The analysis was made at the flight condition of occurence of maximum dynamic pressure, (q) max, with the design 95 percentile wind profile for the month of March. Engine-out control responses showed that the time to double amplitude (TDA) and control authority (CA) are smaller for the Saturn V than for the INT-20. This indicates that the post-engine-out control capability of the INT-20 is less than that of a typical Saturn V and that a detailed analysis should be made to determine INT-20 controllability in an abort situation.

4.2 BASELINE VEHICLE DESIGN

The Boeing Company, as prime contractor, performed the S-IC stage and vehicle analysis/design tasks, the S-IC stage and vehicle resources study, and integrated the overall study efforts. The McDonnell Douglas Astronautics Company, under subcontract to Boeing, performed the S-IVB stage design analysis and resources tasks and was responsible for defining the S-IC/S-IVB interface. The Federal Systems Division of the International Business Machines Company (IBM), also under sub-contract to Boeing, performed the design analysis and resources tasks for the Instrument Unit and stage astrionics systems.

4.2.1 Vehicle Arrangement

The baseline INT-20 vehicle selected for design studies is shown in Figure 4.2.1-1. The configuration is made up of the following components:

a. An S-IC stage with F-1 engines (center engine removed)

b. An S-II/S-IVB interstage (retrorocket motors deleted), with aft interface adapted to be compatible with the S-IC forward face.

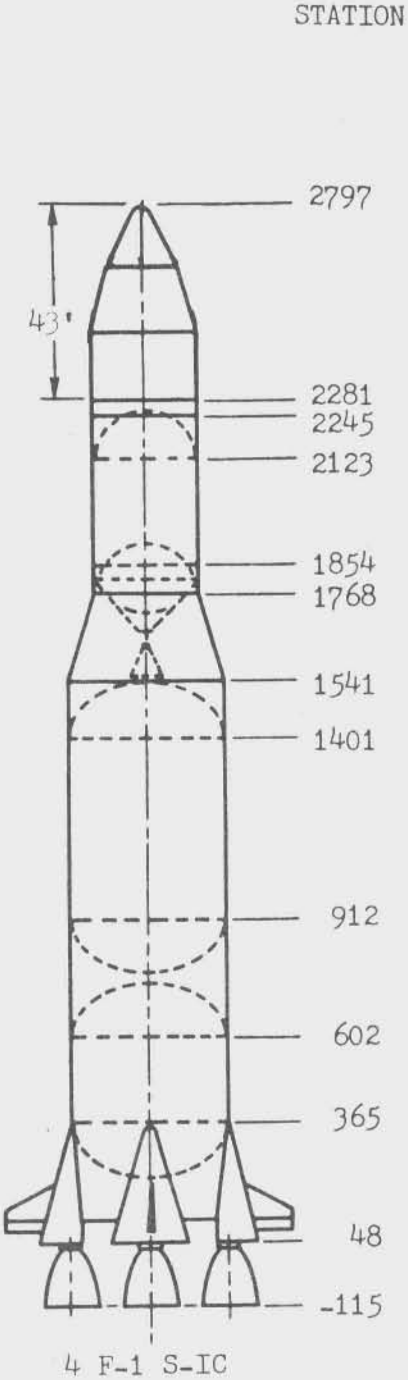
c. A 500-series S-IVB stage.

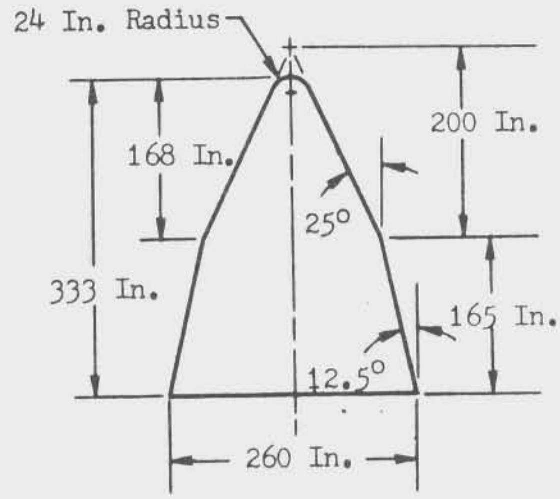
d. A 500-series Instrument Unit (IU), and

e. A 43-foot long payload, comprised of an MSFC double-angle nose cone (MLV shape) plus a 15 foot (13.1 meters), 260 inch (6.6 meter) diameter cylinder.

4.2.2 S-IC Stage and GSE/ESE Impact

The documented S-IC-11 configuration was used as the INT-20 baseline S-IC stage, with changes minimized. The changes to pneumatic equipment, and test and checkout equipment are described in sub paragraph 4.2.2.1.





MLV NOSE CONE

NOTE:
1. INT-20 BASED ON SATURN V AS-511 DESIGN
2. J-2 ENGINE ON S-IVB
3. LAUNCH ESCAPE SYSTEM (LES) NOT CONSIDERED FOR PHASE II ANALYSIS AND DESIGN

FIGURE 4.2.1-1 INT-20 BASELINE VEHICLE

4.2.2.1 S-IC Stage

The baseline S-IC stage configuration for the INT-20 vehicle is defined as the documented S-IC-11 configuration revised as delineated in this report. The basic design philosophy used to establish the configuration was to minimize such changes, consistent with INT-20 and applicable Saturn V criteria. Consideration was also given to maintaining the capability to convert from an INT-20 to a Saturn V and to the cost factors relating to the stage items which required revision.

a. Structures subsystems

1. Forward skirt (60B14009)

   No design changes will be required to the forward skirt for INT-20. The S-IC/S-IVB interface will be accomplished by means of an adapter ring which is compatible with the existing interface bolt patterns of both the S-IC and S-IVB (Method 1 of FIGURE 4.2.2.1-1). The added adapter ring will be supplied by McDonnell Douglas. An alternate direct interface method could be used for the baseline INT-20. It consists of using a modified S-IC hole pattern which is compatible with both the S-IC and S-IVB (Method 2 of FIGURE 4.2.2.1-2).

2. Oxidizer tank (60B03101)

   The oxidizer tank design changes will be in the area of the inboard LOX suction fitting and result from deletion of the inboard LOX suction duct. A flat plate cover with a floating flange, which uses the existing LOX suction duct seal, will be added to close the oxidizer tank at the inboard suction fitting. (Method 2 of FIGURE 4.2.2.1-3). The configuration of the cover and floating flange will be the same as presently used for hydrostatic test. The floating flange material, however, will be 2219-T87 instead of the 7075-T6 now used. The inboard LOX standpipe will be deleted. A support ring will be added to the inside of the suction fitting to replace the support provided to the cruciform baffle by the standpipe flange. Existing standpipe and suction duct attachment provisions will be used for attachment of the added ring and cover.

3. Intertank (60B29800)

   There will be no required design changes to the intertank.

4. Fuel tank (60B25001)

   New covers will be required for the inboard fuel suction elbows

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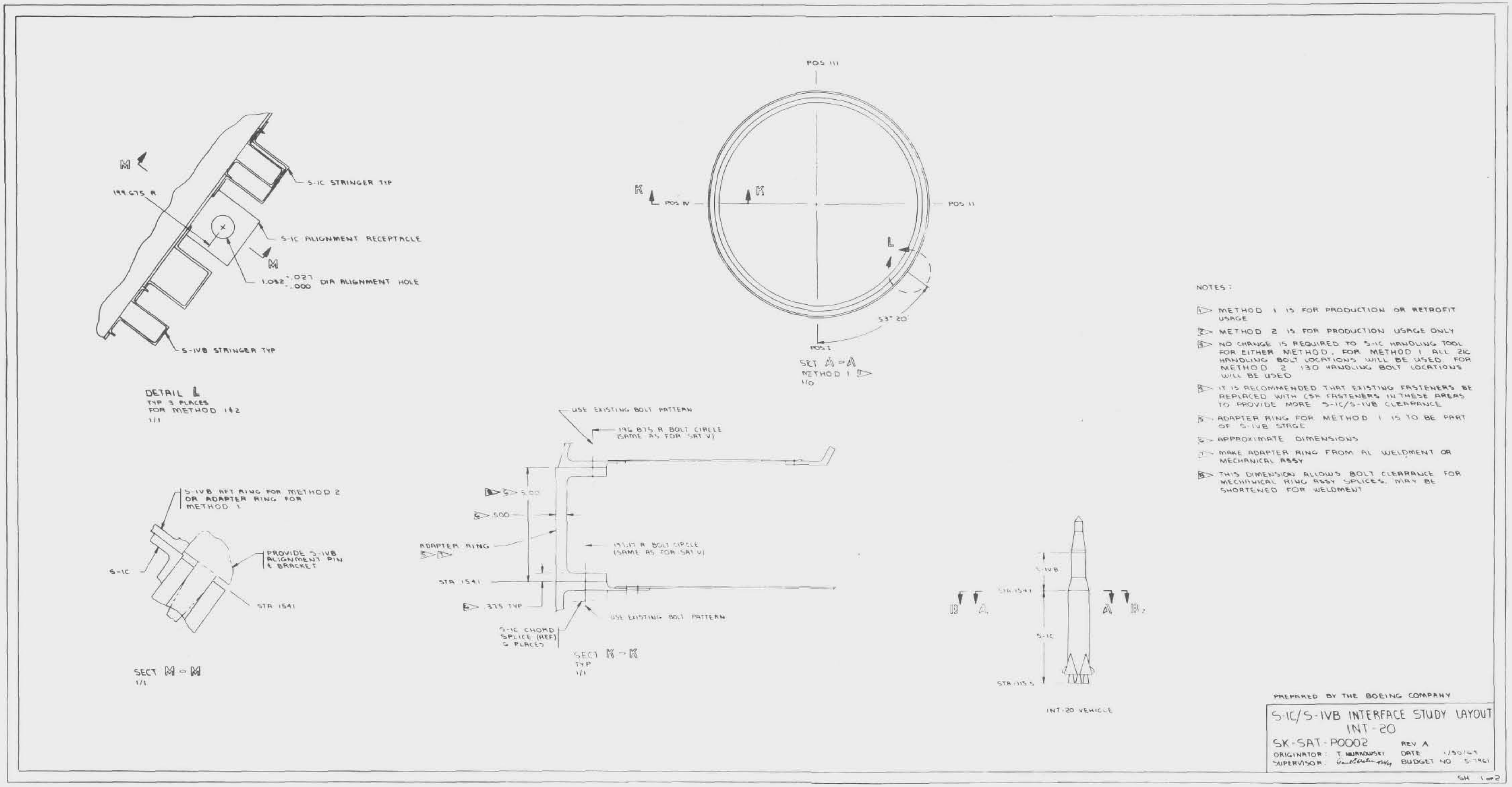

FIGURE 4.2.2.1-1

FIGURE 4.2.2.1-1 S-IC/S-IVB INTERFACE STUDY LAYOUT INT-20

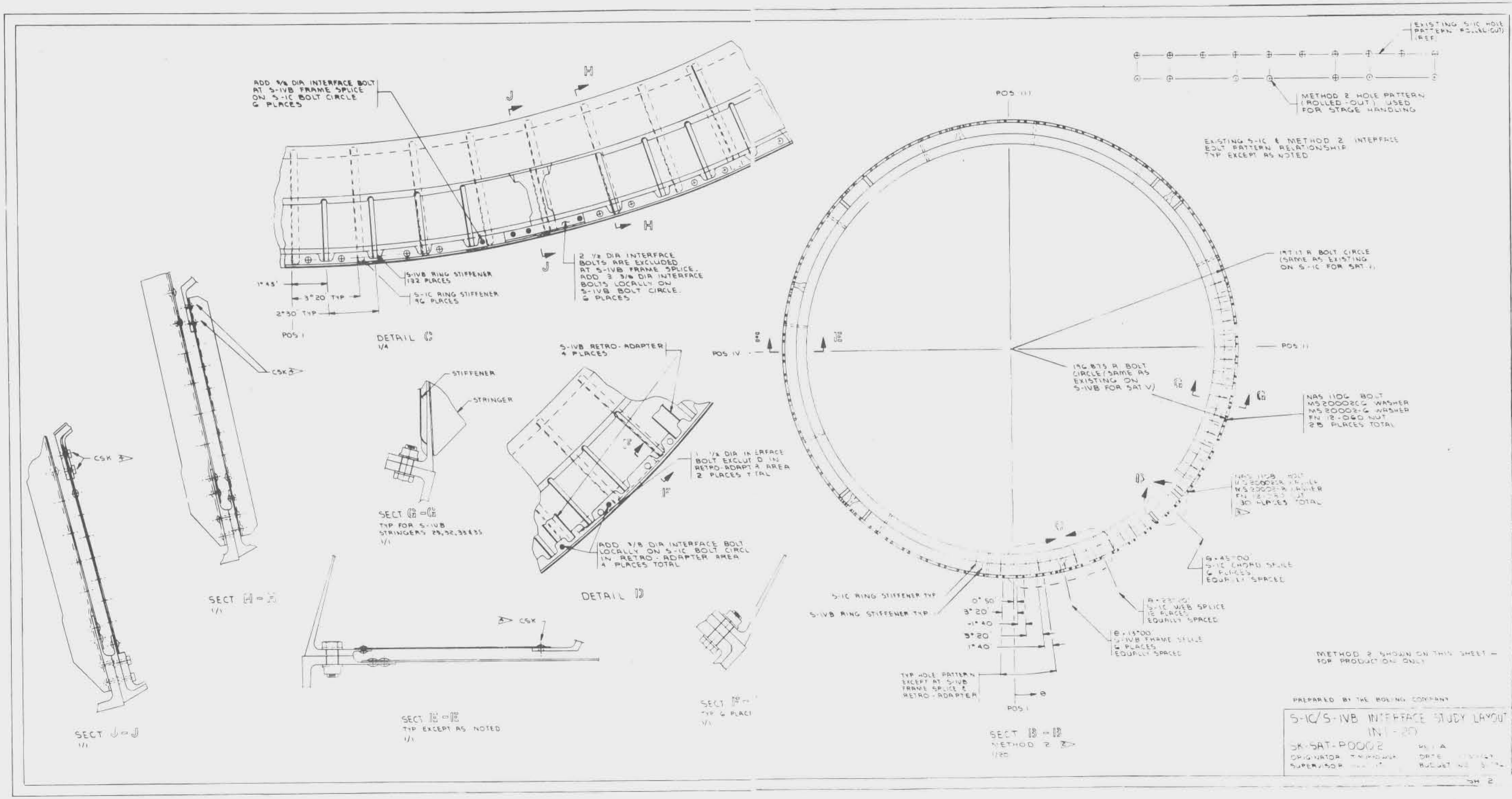

FIGURE 4.2.2.1-2

FIGURE 4.2.2.1-2 S-IC/S-IVB INTERFACE STUDY LAYOUT INT-20

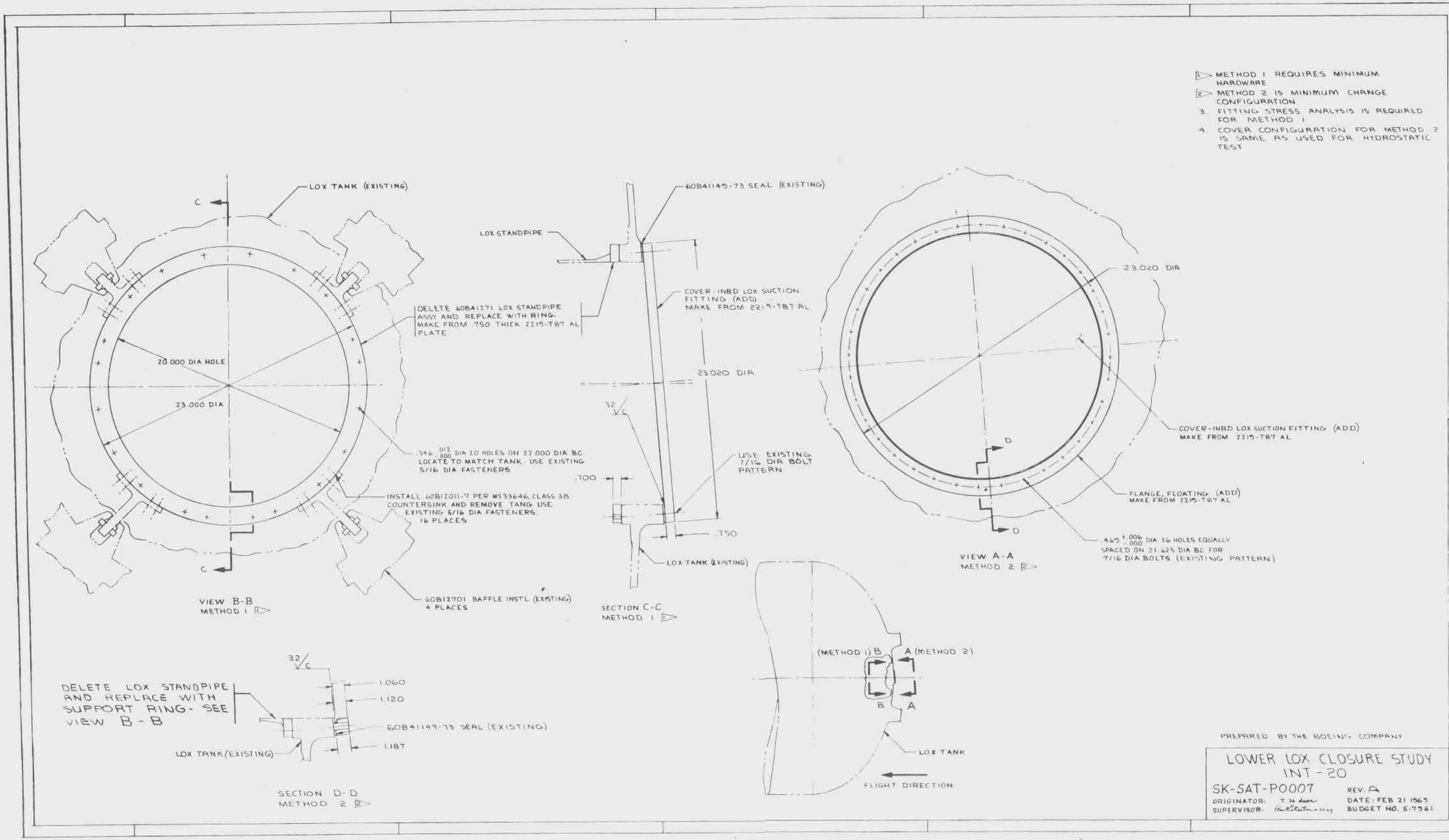

FIGURE 4.2.2.1-3 LOWER LOX CLOSURE STUDY INT-20

FIGURE 4.2.2.1-3 .

4.2.2.1 (Continued)

and the inboard LOX tunnel. The upper fuel instrumentation cover will be revised to provide capability for installing one additional pressure switch. The lower fuel tank bulkhead base gores must be revised to provide increased hoop compression capability (see Section 4.4 for definition of minimum-modification S-IC stage).

(a) Inboard fuel suction elbow closures

Flat plate covers, which use existing fuel suction duct at-attachment provisions and seals, will be added to close the fuel tank at inboard fuel suction elbows (FIGURE 4.2.2.1-4).

(b) Inboard LOX tunnel cover

A non-structural and non-sealing cover will be added at the forward end of the inboard LOX tunnel to prevent glas flow between the thrust structure and intertank (FIGURE 4.2.2.1-4). Existing tunnel handling holes will be used for attachment of the cover.

(c) Revised lower fuel base gores

The eight lower fuel tank bulkhead base gore segments will be revised to increase the thickness in the area near the Y-ring (FIGURE 4.2.2.1-5). This increase in thickness is recommended for all follow on S-IC stages of an INT-20/S-IC production mix to minimize tooling and structural capability differences.

(d) Revised instrumentation cover

The instrumentation cover assembly will be revised to add an additional pressure port and nut plate, to increase the pressure switch attach land and to add attachment inserts (FIGURE 4.2.2.1-6).

5. Thrust structure (60B18054)

Thrust structure design changes will consist of deleting support provisions for the center engine and the inboard fuel suction ducts. Consideration will also be given to the KSC installed (75M14644) slow release system.

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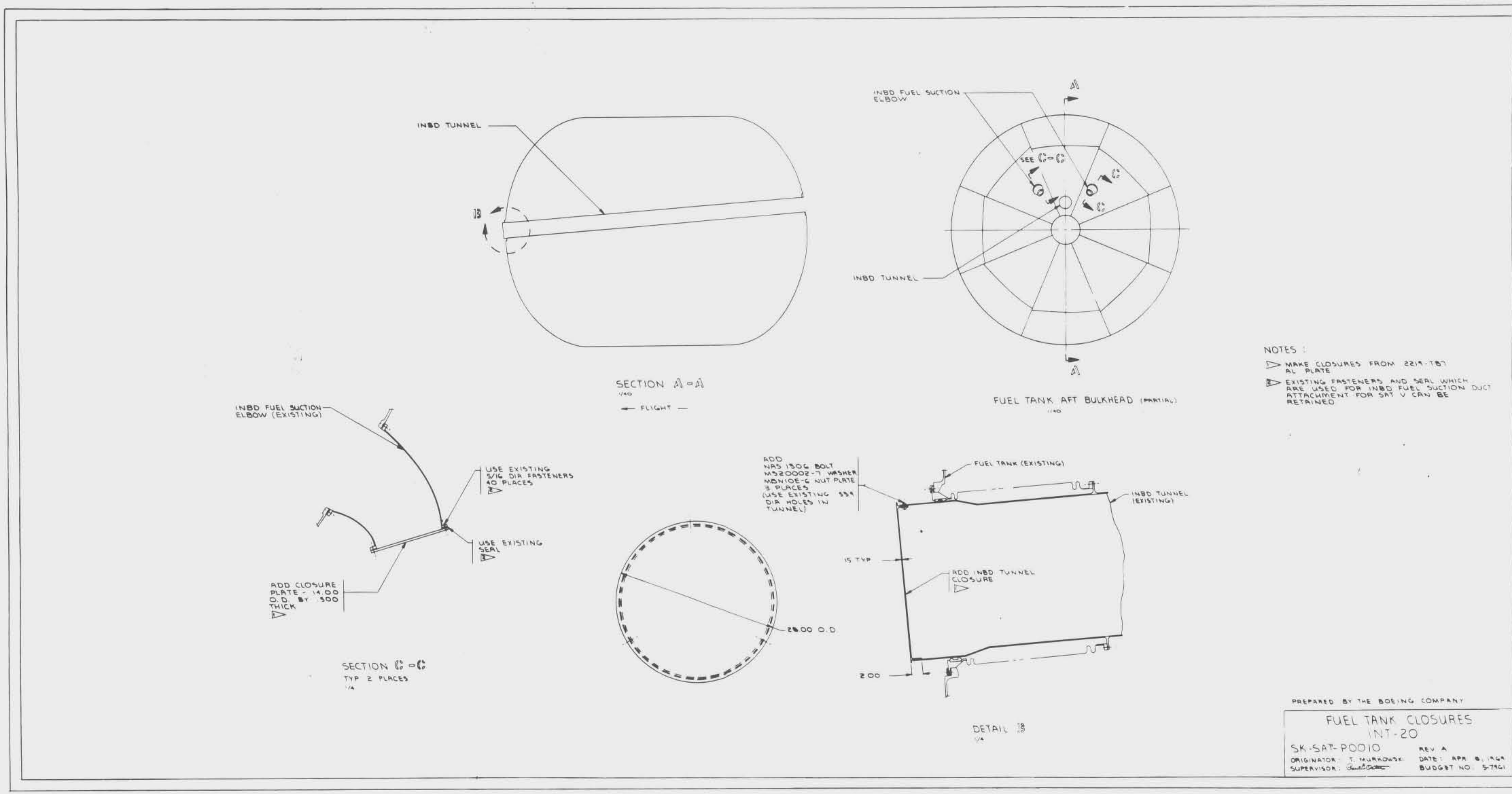

FIGURE 4.2.2.1-4 FUEL TANK CLOSURES INT-20

FIGURE 4.2.2.1-4 .

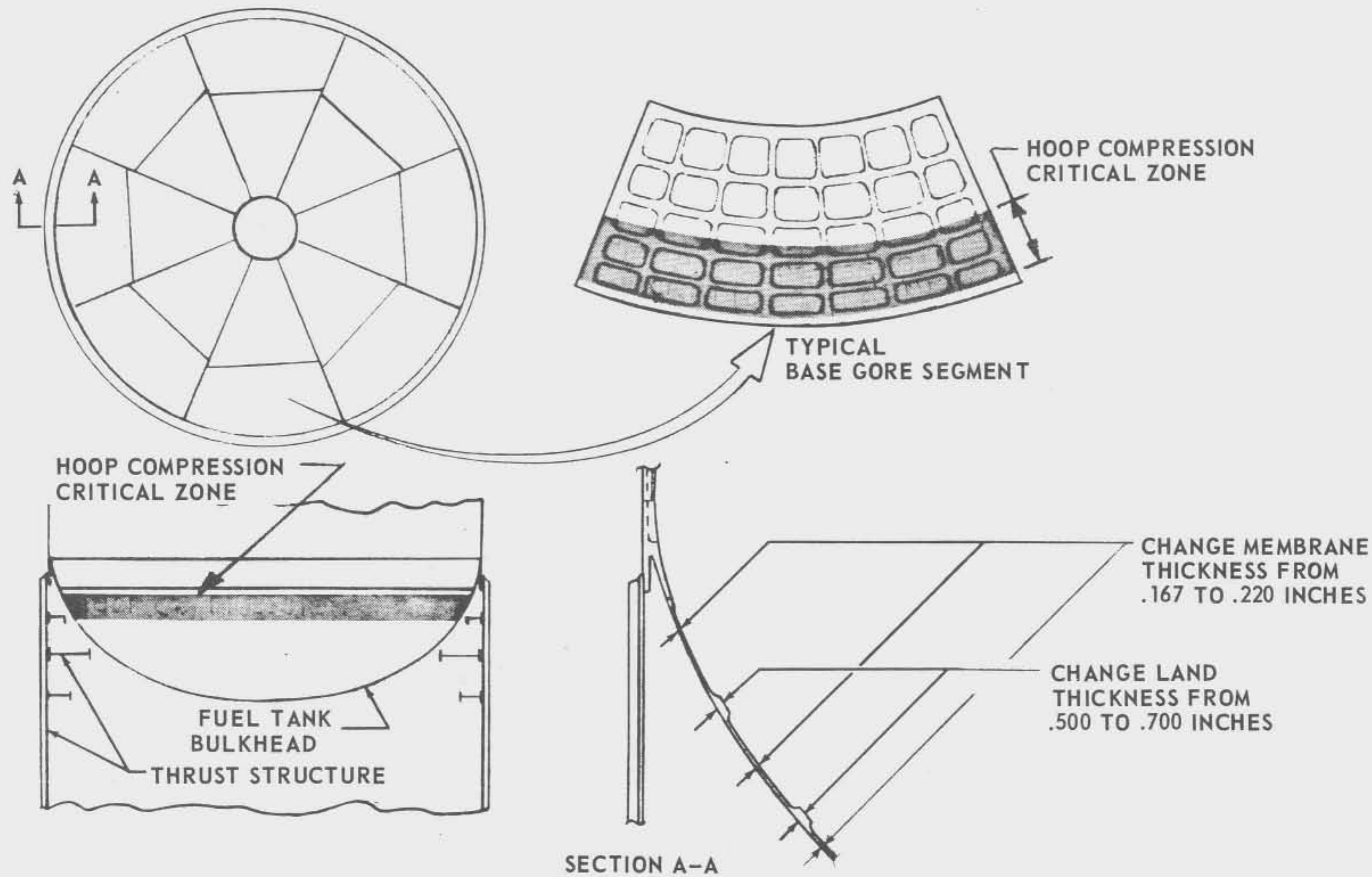

FIGURE 4.2.2.1-5 S-IC FUEL TANK AFT BULKHEAD DESIGN CHANGE

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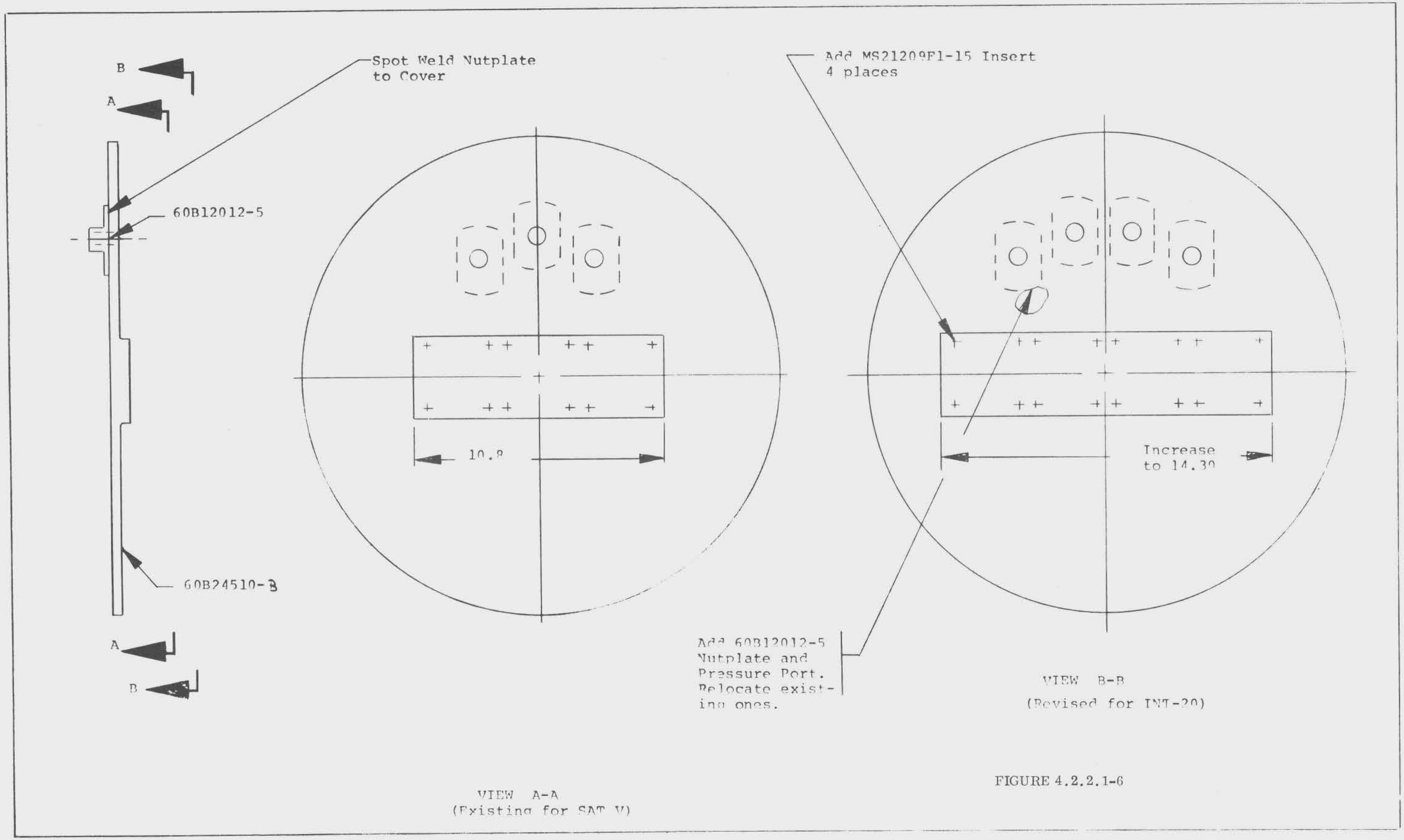

FIGURE 4.2.2.1-6 REVISED UPPER FUEL INSTRUMENTATION COVER

4.2.2.1 (Continued)

(a) Center engine supports

The center engine support struts, strut insulation, strut fittings, the associated strut attach hardware and the center engine adapter fitting will be deleted (FIGURE 4.2.2.1-7). The strut fitting and adapter fitting attach hardware will be retained because they also provide common attachment for adjacent structure.

(b) Inboard fuel suction duct supports

The eight inboard fuel suction duct support links and associated attach hardware in the inboard propellant duct support structure will be deleted (FIGURE 4.2.2.1-8).

(c) Slow release system

The existing capability for varying the number of slow release devices for mission flexibility will be used to attain the required number for INT-20 (FIGURE 4.2.2.1-9).

6. Heat shield (60B20800)

The base heat shield will be revised to delete penetrations provided for the center engine and its associated systems. The array of small heat shield panels in the center area will be replaced with standard square panels. For flight, six standard flight panels will be used (Method 2 of FIGURE 4.2.2.1-10). For static firing, six standard static firing honeycomb and six standard static firing steel back-up panels will be used (FIGURE 4.2.2.1-13). The center engine flame curtain will also be deleted.

The center area heat shield support structure will be changed to a simple square beam grid compatible with the six square heat shield panels (FIGURE 4.2.2.1-11). Existing beams and attach brackets (same as used for adjacent structure) will be used. A new inconel bracket will be added to support the panels at the deleted center engine adapter location. The added bracket will be attached at existing bolt locations (FIGURE 4.2.2.1-12).

7. Structures supplemental data

The identification of the above changes with INT-20 criteria is contained in Appendix A, Section 1.0. Section 2.0 of Appendix A

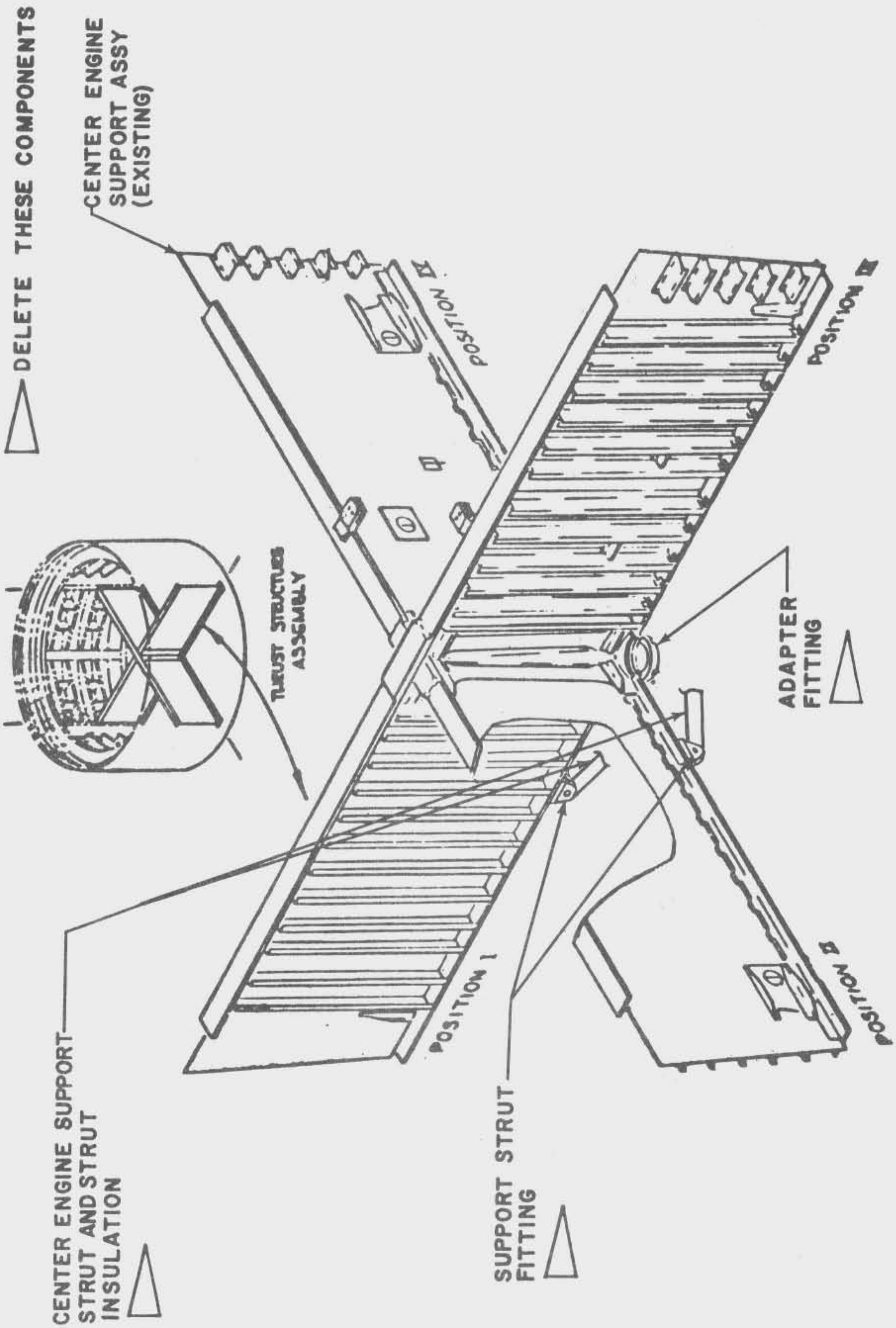

FIGURE 4.2.2.1-7 CENTER ENGINE SUPPORT DELETION

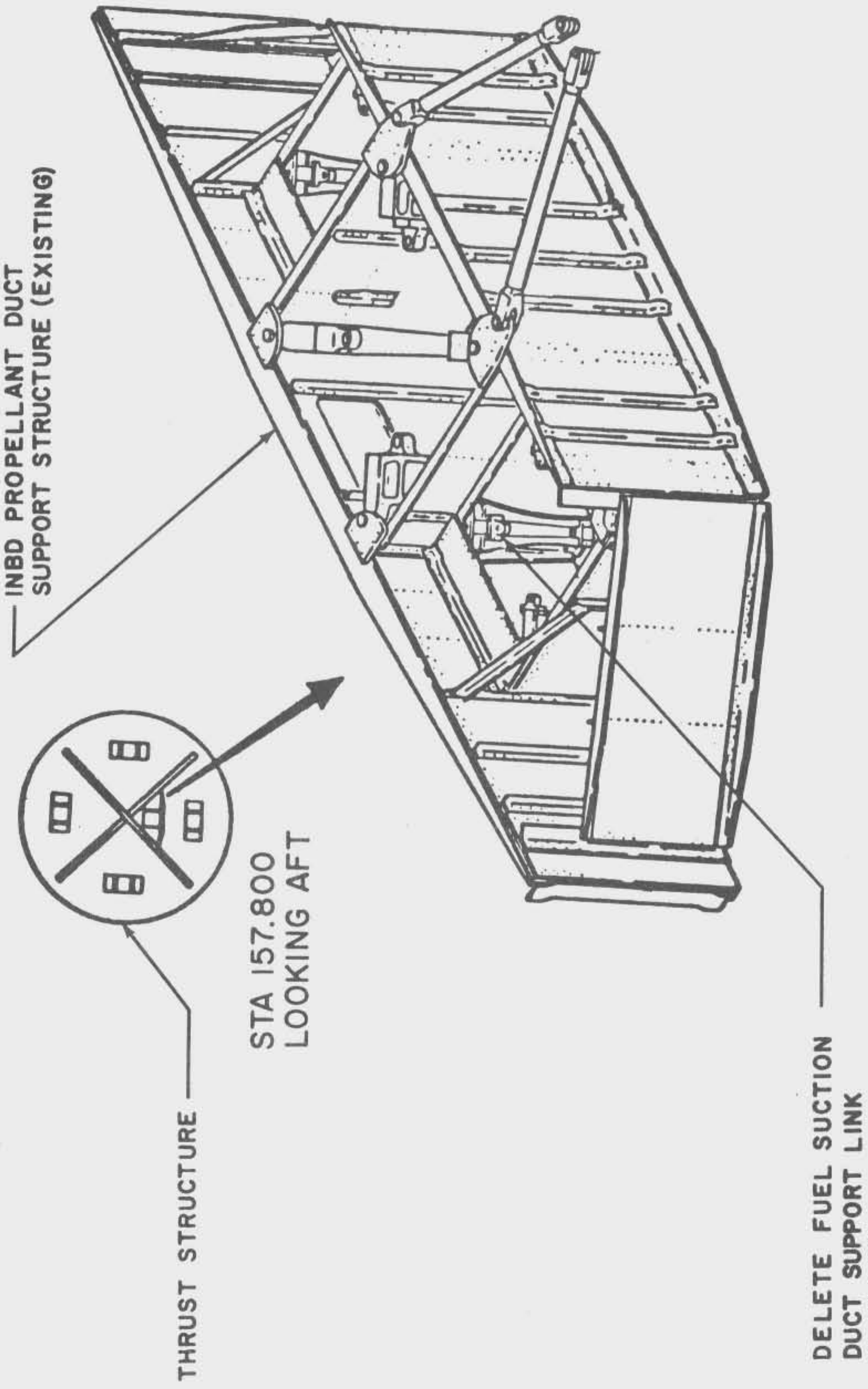

FIGURE 4.2.2.1-8 FUEL SUCTION DUCT SUPPORT DELETION

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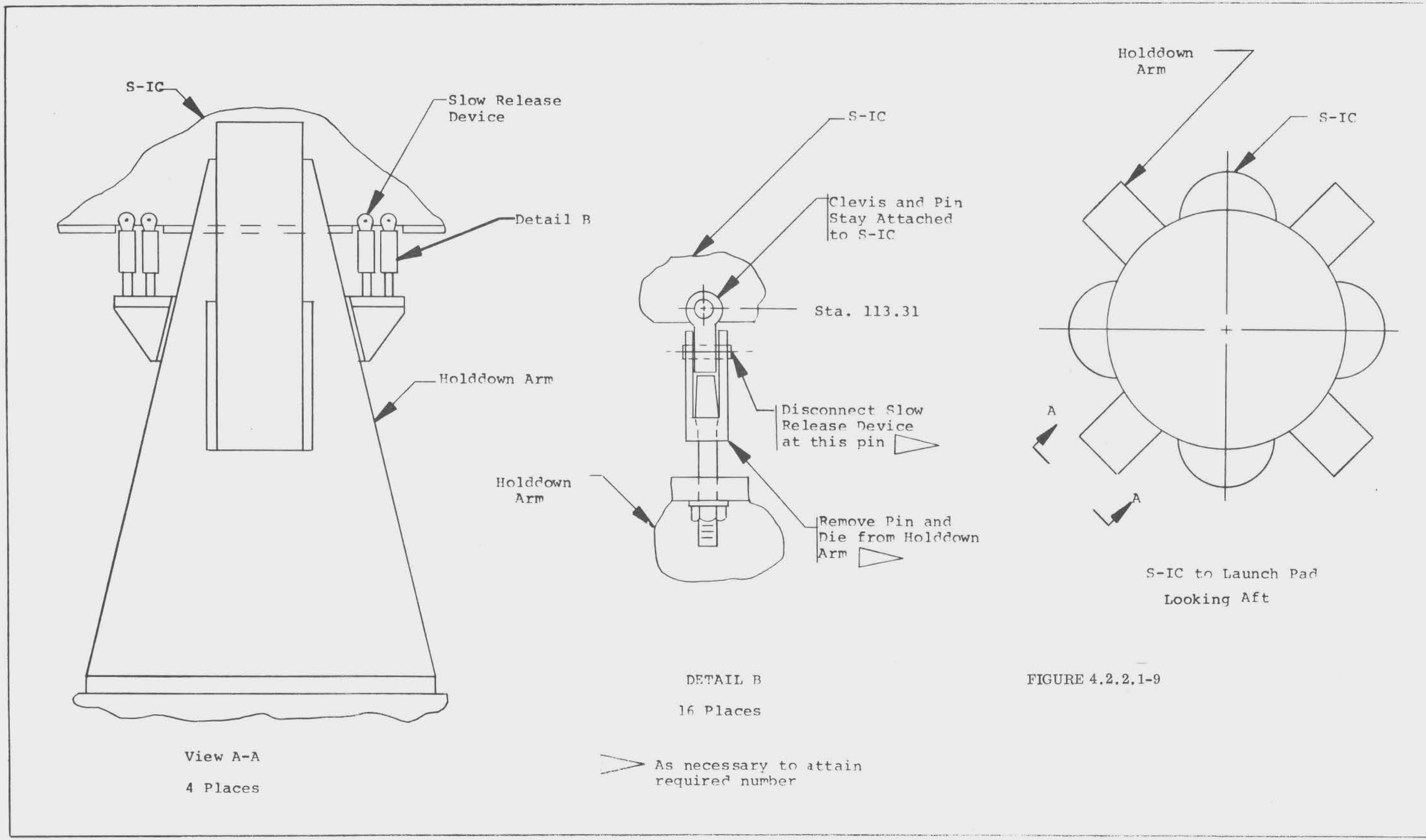

FIGURE 4.2.2.1-9 SLOW RELEASE SYSTEM

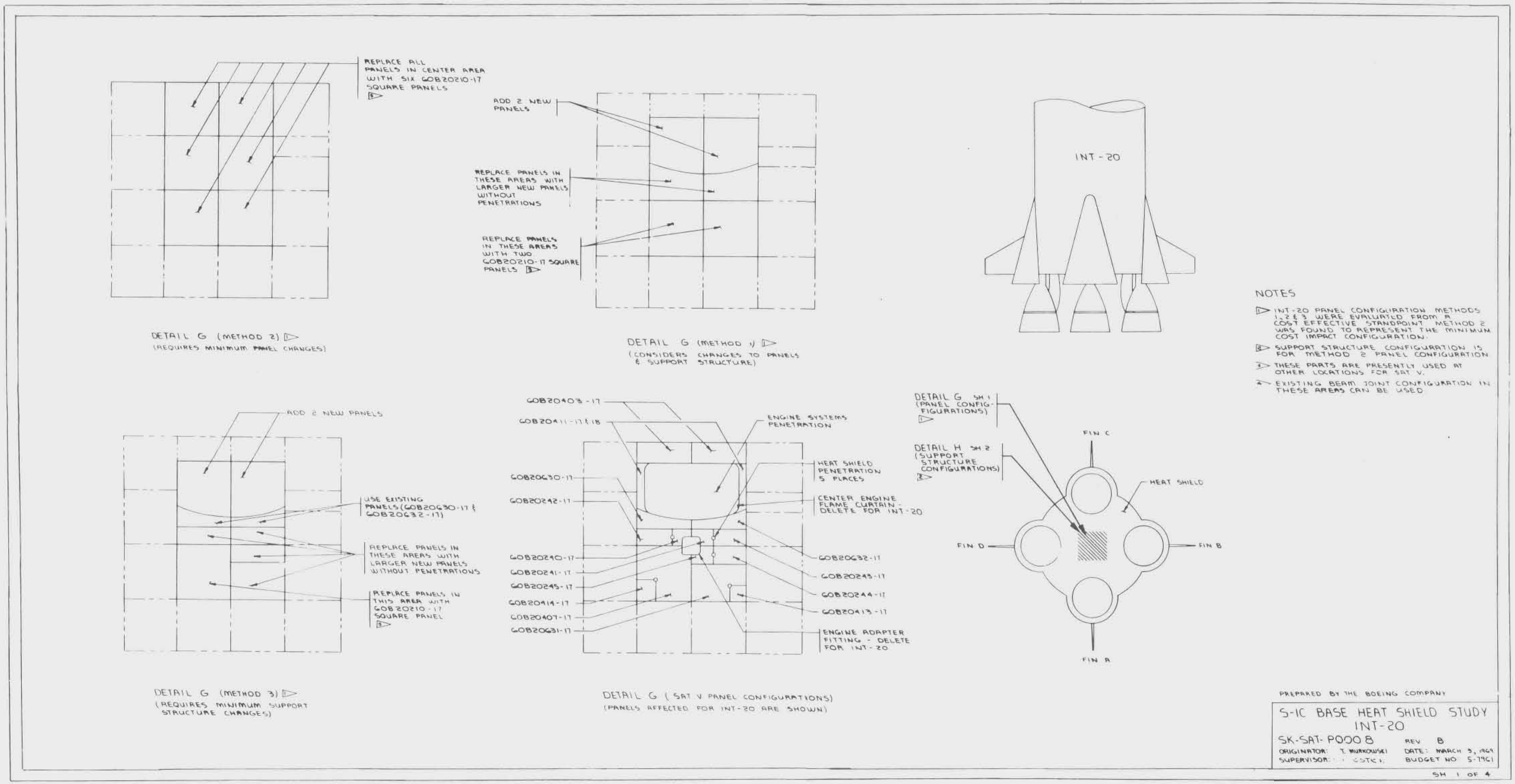

FIGURE 4.2.2.1-10

FIGURE 4.2.2.1-10 S-IC BASE HEAT SHIELD STUDY INT-20

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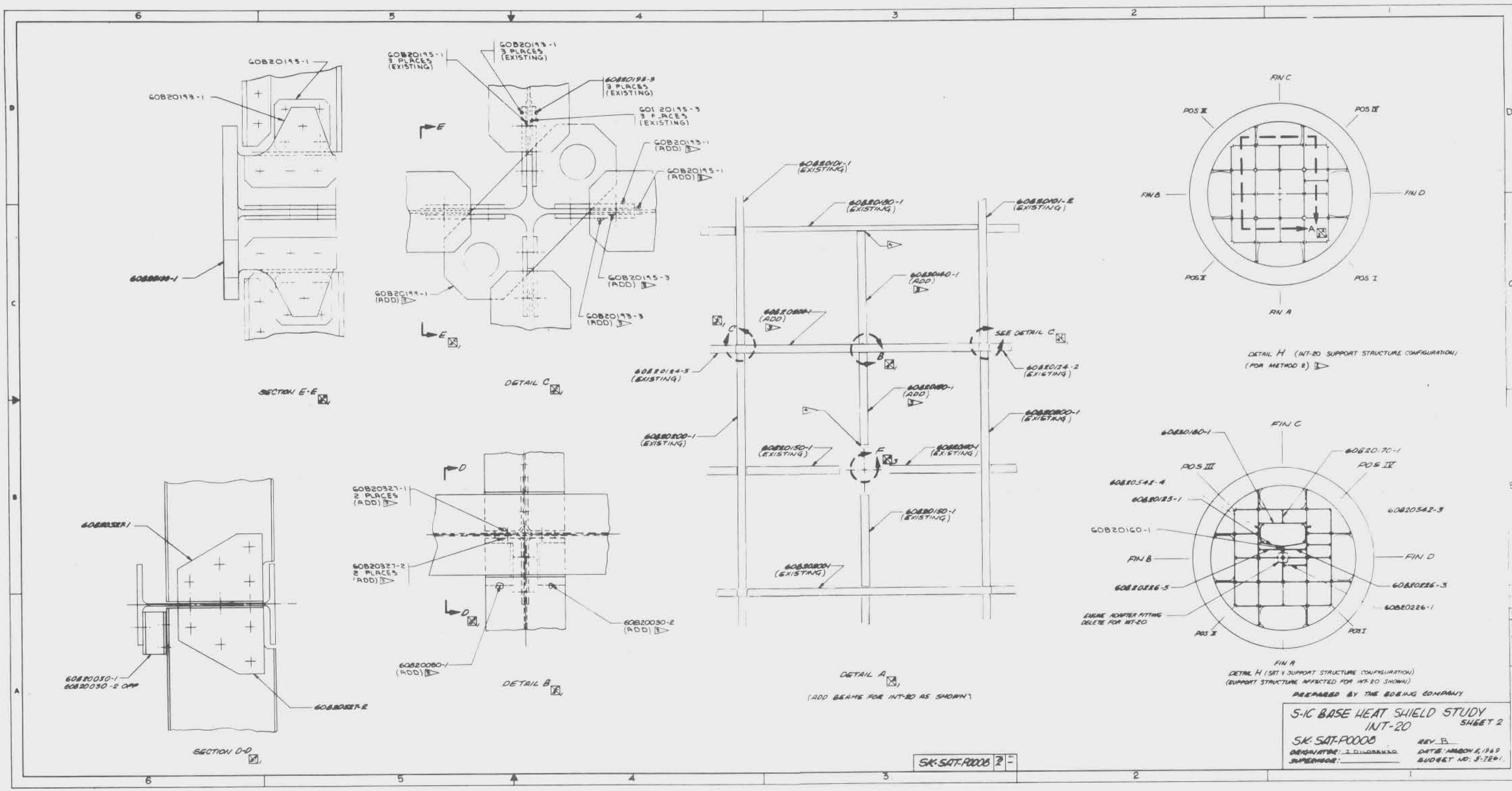

FIGURE 4.2.2.1-11

FIGURE 4.2.2.1-11 S-IC BASE HEAT SHIELD STUDY INT-20

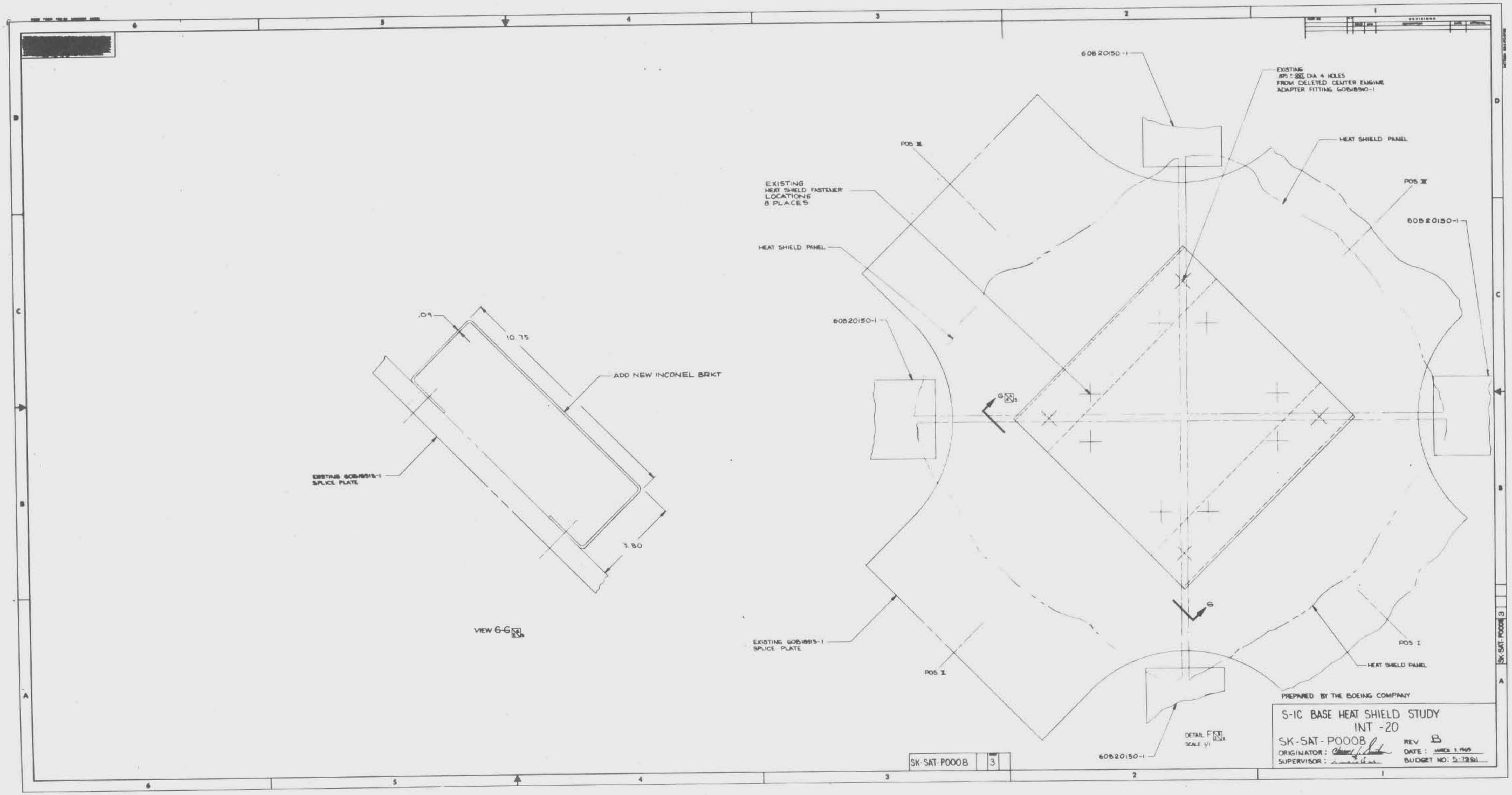

FIGURE 4.2.2.1-12 .

FIGURE 4.2.2.1-12 S-IC BASE HEAT SHIELD STUDY INT-20

4.2.2.1 (Continued)

contains configuration trade study and technical support study data. Section 3.0 of Appendix A is a listing of parts deleted, added or revised and their respective weights.

2. Other structure changes

A number of S-IC stage components of the S-IC-11 thru S-IC-15 configuration are identified as both susceptible to stress corrosion cracking and critical to mission accomplishment. These stress corrosion susceptible stage parts are categorized into three levels of criticality. For stages thru S-IC-15, periodic inspection of accessible category I (Most critical) items was implemented by ECP 434. Material substitution, to eliminate stress corrosion susceptibility of category I, II and III parts, is considered mandatory for follow-on production stages (including stages for INT-20). The design change definition to accomplish the material substitution is a line item of the S-IC-16 and on standard stage configuration and cost study and is therefore not included in this report.

b. Propulsion and mechanical subsystems

For purposes of this study the S-IC propulsion/mechanical subsystems are identified as follows:

Oxidizer System
Fuel System
Auxiliary System
Flight Control System
Engine and related components

Changes for these systems will consist primarily of eliminating the branch ducting for the center engine. The design objective was to reduce leak points and to eliminate all ducting components not required for the satisfactory operation of the 4 engine configuration. Ducting support hardware common to retained components or integral with the structure will not be deleted or revised.

1. Oxidizer system

(a) Oxidizer fill and drain (60B41012)

No design changes to this system will be required.

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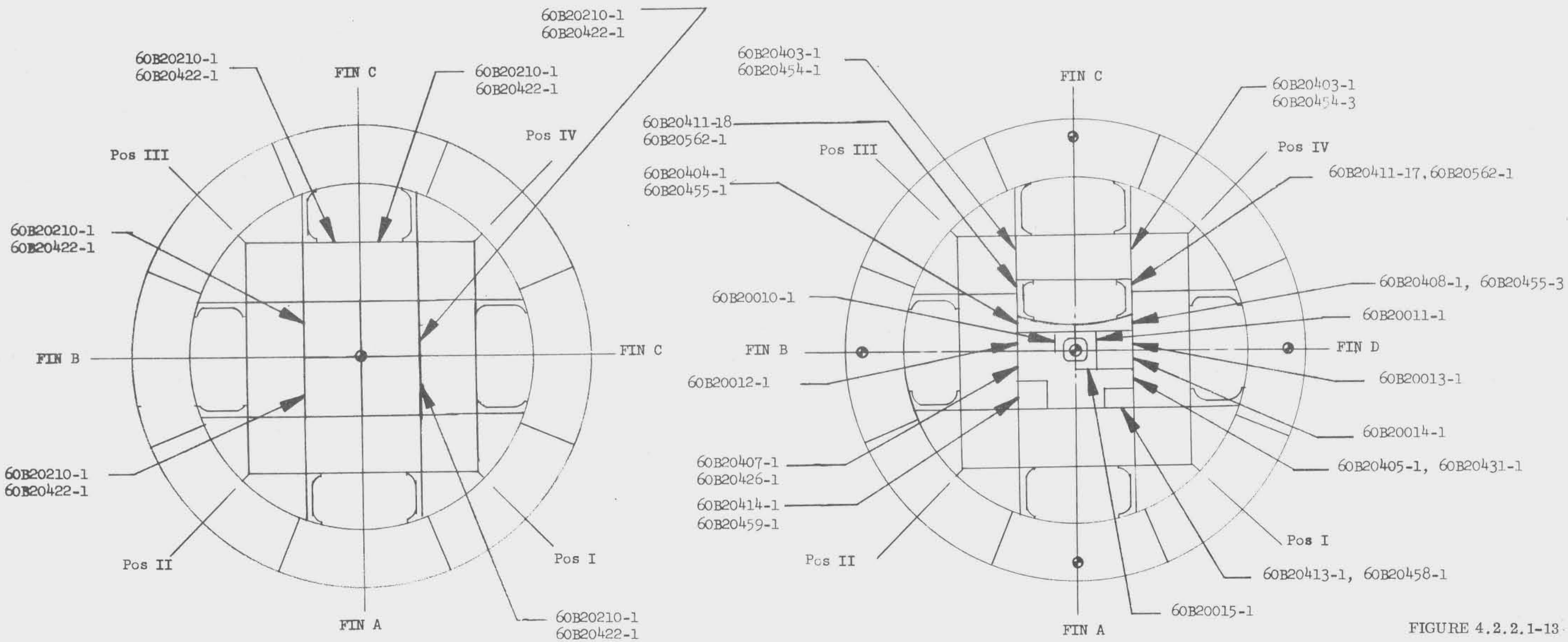
NOTE: The first part number shown is the honeycomb static test panel. The second part number shown is the steel back-up static test panel. If only one part number is shown it is the steel back-up panel. No honeycomb panels are used for static test in these areas.



PROPOSED STATIC FIRING CONFIGURATION

EXISTING STATIC FIRING CONFIGURATION

FIGURE 4.2.2.1-13

PREPARED BY THE BOEING COMPANY

S-IC BASE HEAT SHIELD STUDY
INT-20

SHEET 4

SK-SAT-P0008

REV B

ORIGINATOR: Di Lorenzo

DATE: 4-11-69

SUPERVISOR:

BUDGET NO: 5-7961

FIGURE 4.2.2.1-13 S-IC BASE HEAT SHIELD STUDY

4.2.2.1 (Continued)

(b) Oxidizer feed system (60B41014)

The inboard LOX suction duct assembly (including roller brackets), LOX prevalve and PVC duct will be deleted (FIGURE 4.2.2.1-14). A support adapter which attaches to the inboard propellant duct support structure will be added to support the LOX interconnect spool. Closure plates which use existing seals will be provided to seal the upper and lower ends of the interconnect spool (FIGURE 4.2.2.1-15).

All LOX cutoff sensors in engines 2, 4 and 5 delivery systems will be deleted and associated bosses will be plugged. An additional sensor will be installed in an existing boss at the forward end of the LOX suction ducts for engines 1 and 3 (FIGURE 4.2.2.1-16). This will result in two LOX cutoff sensors in each of engine's 1 and 3 LOX feed systems, thus allowing the retention of the 2 out of 4 voting logic.

(c) Oxidizer conditioning system

(1) LOX interconnect system (60B41014)

The normally closed interconnect valve at engine position 2 will be replaced with a spool (FIGURE 4.2.2.1-17 & -18). The control pressure line to that valve will be deleted and a cap added at the aft umbilical flight plate. A temperature transducer will be installed in an existing boss in the center LOX spool.

(2) LOX bubbling system (60B41221)

The center engine bubbling system will be deleted from the branch tee to the center LOX spool. The tee will be capped and the LOX spool boss will be plugged (FIGURE 4.2.2.1-19).

(d) Oxidizer pressurization (60B51400)

The GOX return duct between the center engine interface and the GOX manifold will be deleted (FIGURE 4.2.2.1-20). The GOX manifold will be capped at the center engine port (FIGURE 4.2.2.1-21 , Item 1). All associated bolt-on bracketry will be deleted.

For the prepressurization control system the existing pressure switch (24.2 - 26.5 psia) will be replaced with a like switch with a pressure setting of 27.5 - 29.0 psia.

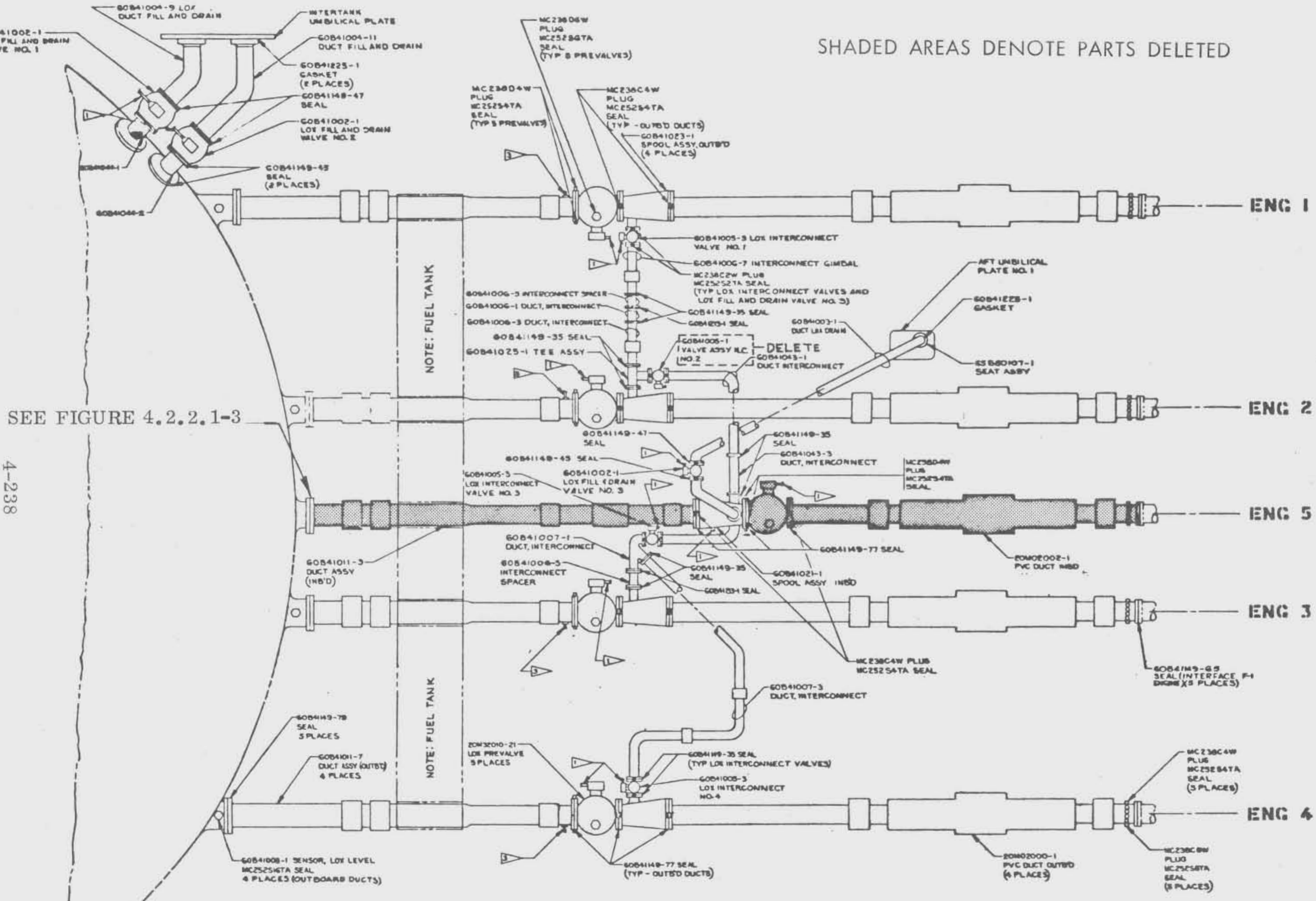


FIGURE 4.2.2.1-14 LOX DELIVERY SYSTEM

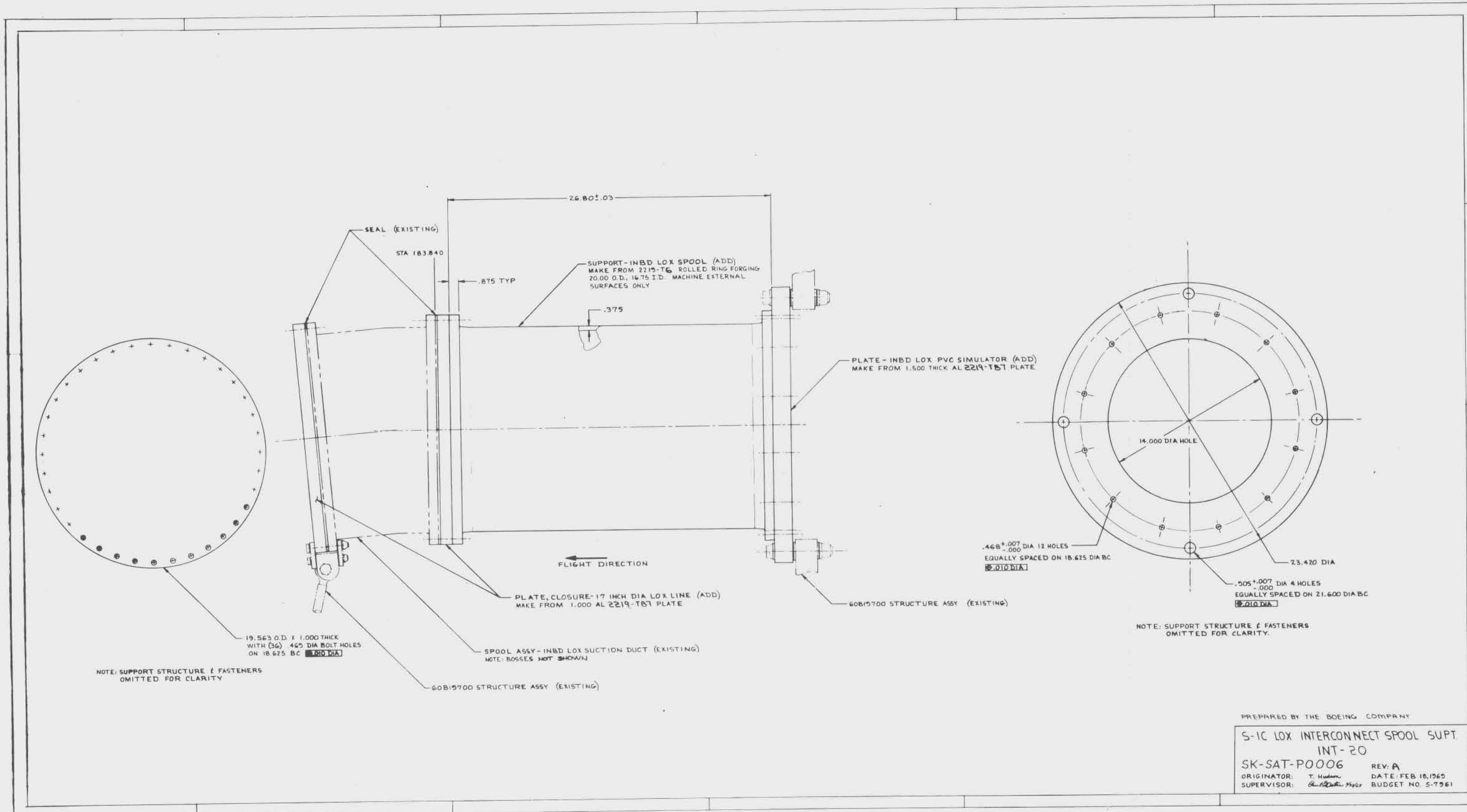


FIGURE 4.2.2.1-15 S-IC LOX INTERCONNECT SPOOL SUPT. INT-20

FIGURE 4.2.2.1-15

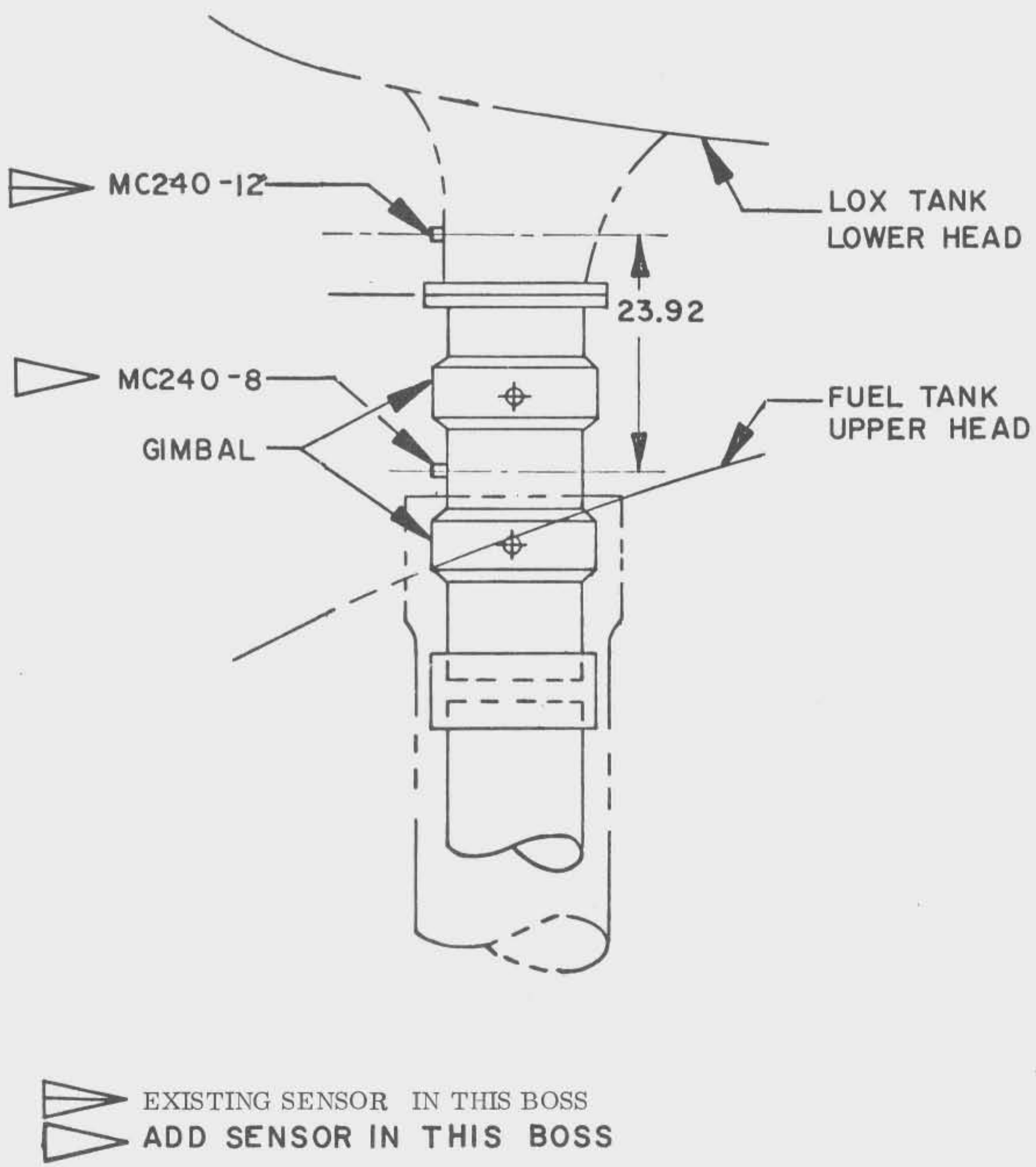


FIGURE 4.2.2.1-16 CUT-OFF SENSOR INSTALLATION

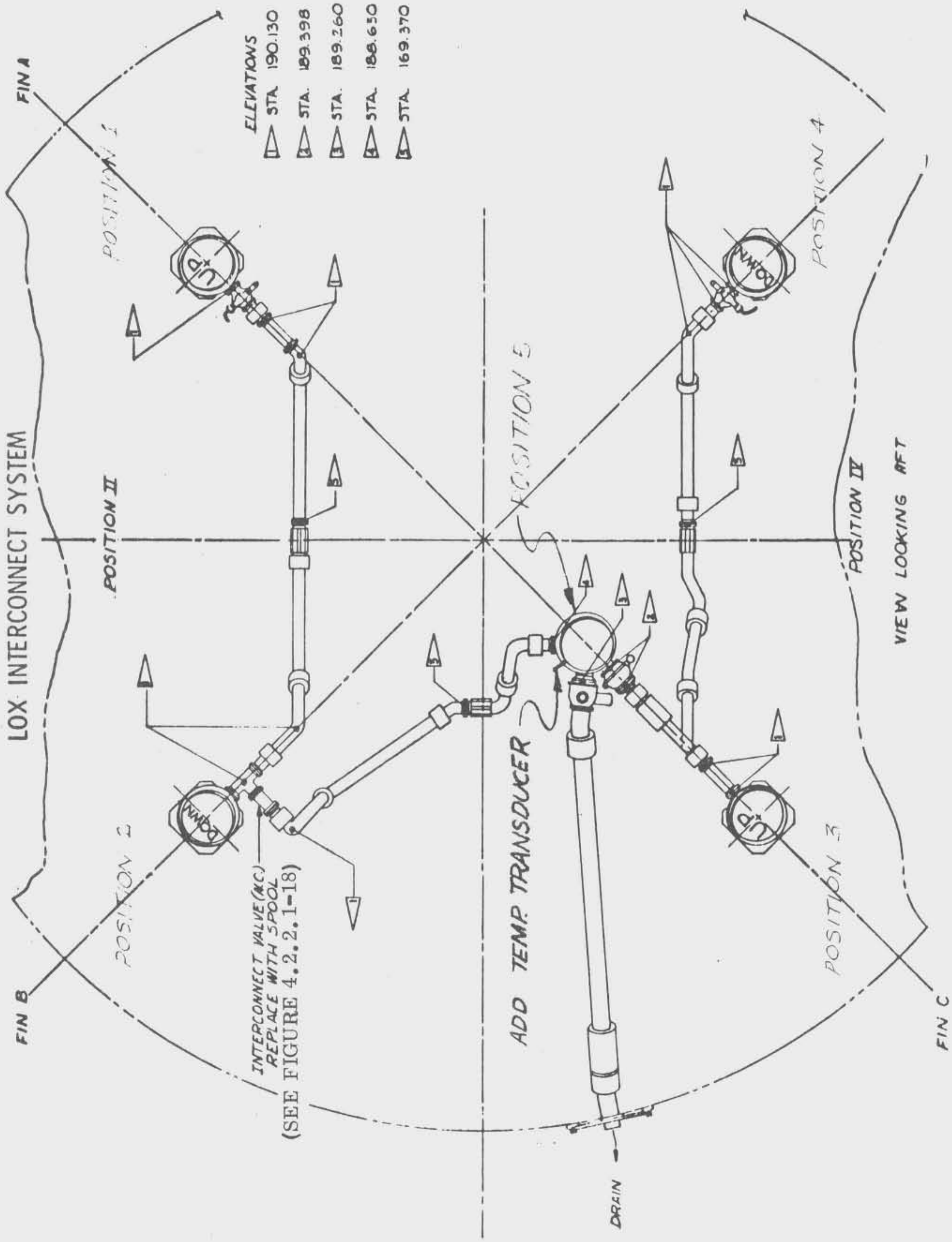


FIGURE 4.2.2.1-17 LOX INTERCONNECT SYSTEM

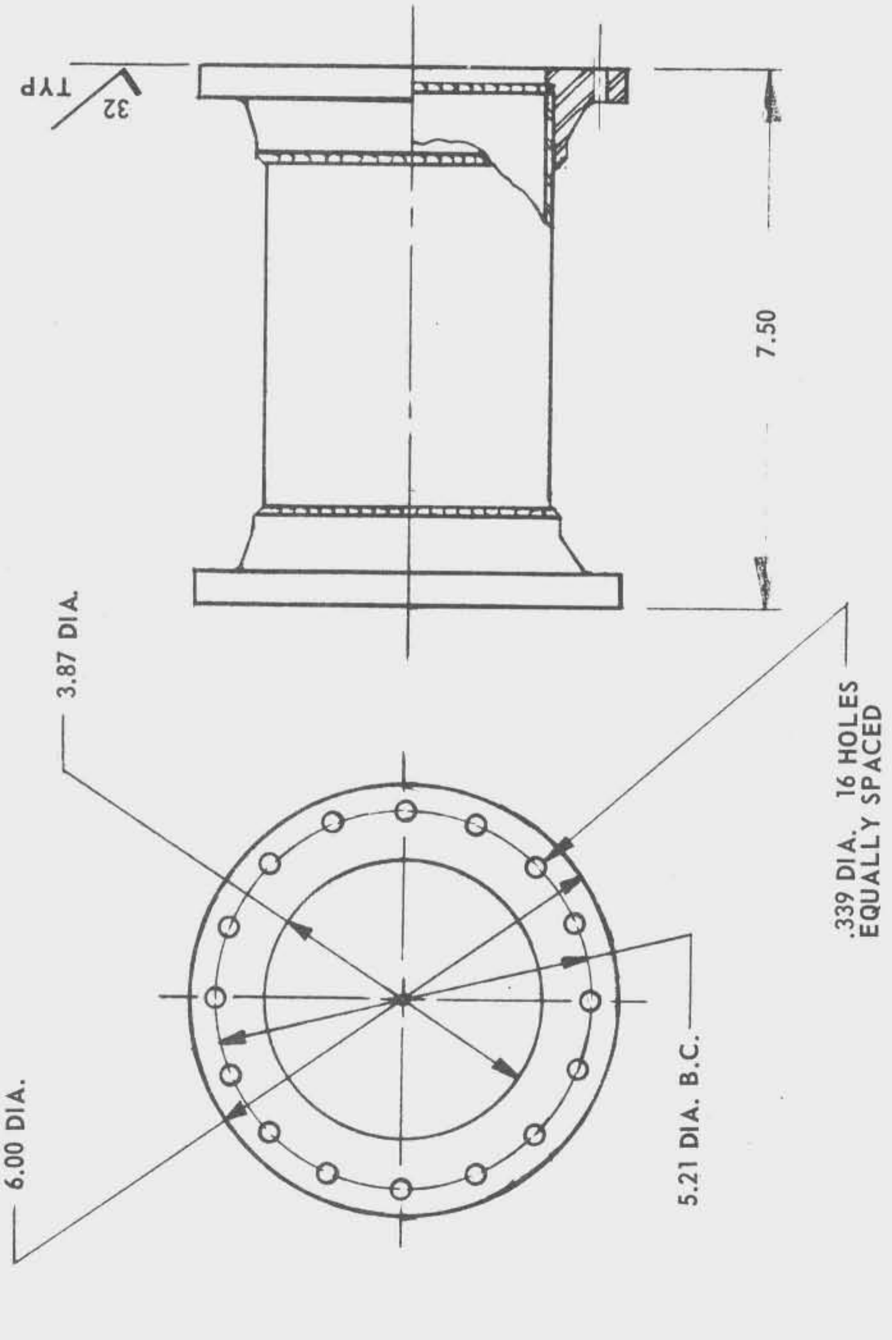


FIGURE 4.2.2.1-18 LOX INTERCONNECT SPOOL ASSEMBLY
(SIMILAR TO 60B41006-5)

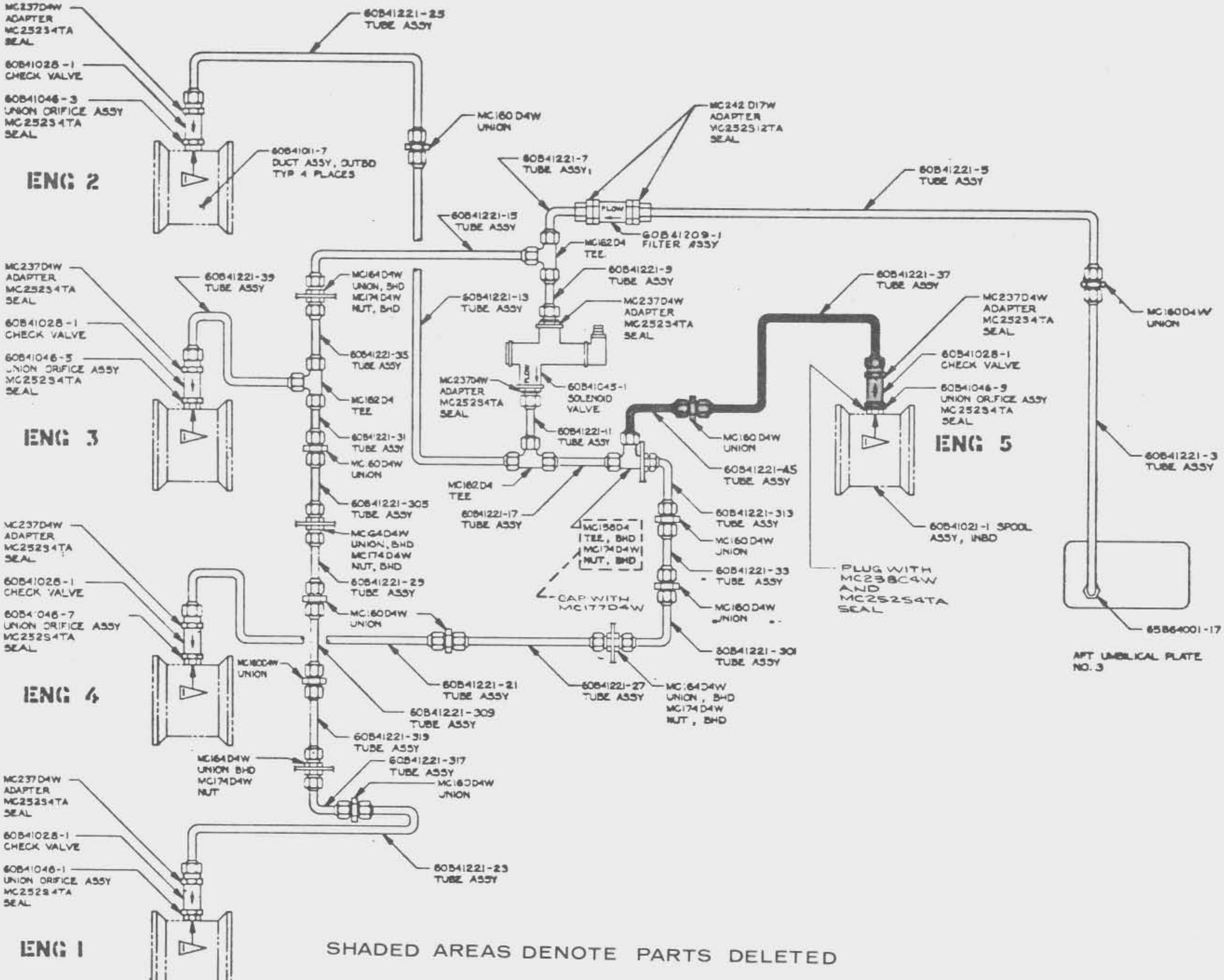


FIGURE 4.2.2.1-19 LOX BUBBLING SYSTEM

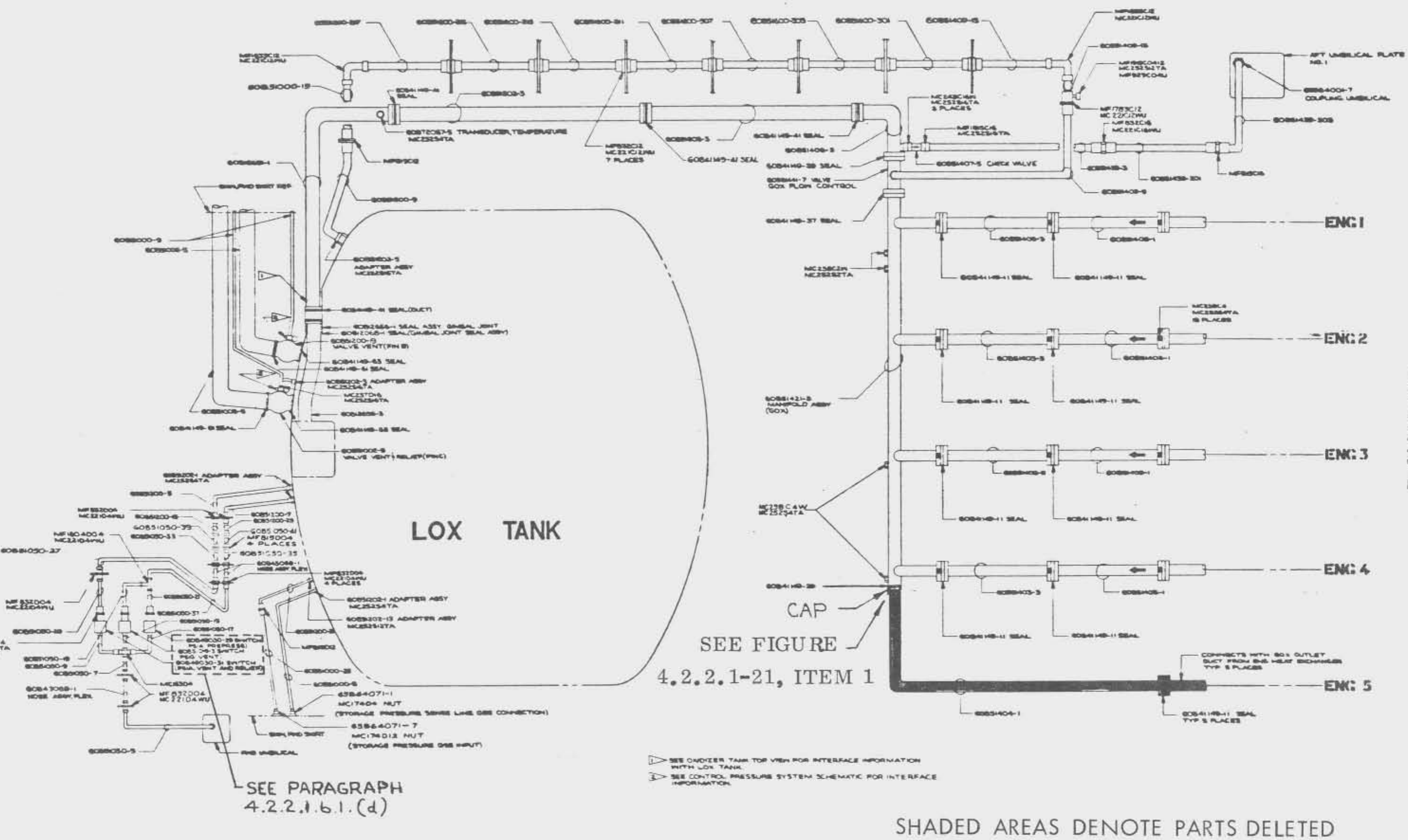
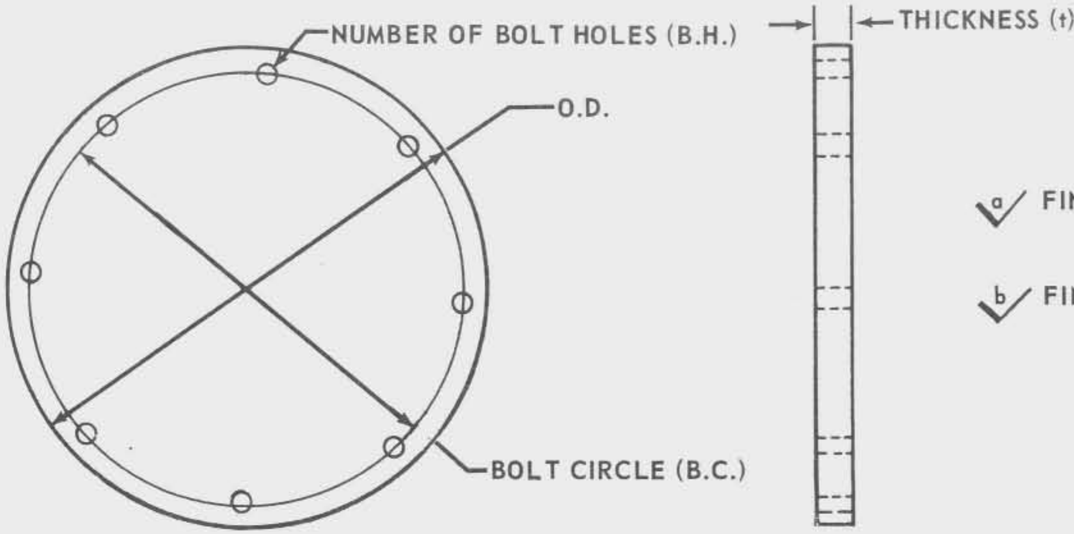


FIGURE 4.2.2.1-20 LOX PRESSURIZATION SYSTEM



a✓ FINISH ON SEALING SURFACE

b✓ FINISH ON ALL EXCEPT SEALING SURFACE

ITEM NO.	SYSTEM	O.D.	B.C.	B.H.	t	MATERIAL	a✓	b✓	QTY/STAGE
1	GOX	4.75	4.125	12	.5	CRES.	32	125	1
2	HELIUM SUPPLY	3.64	2.88	6	.5	CRES.	32	125	1
3	HELIUM RETURN	3.39	2.63	6	.5	INCONEL	32	125	1
4	HYDRAULIC SUPPLY & RETURN	3.060	2.44	6	.5	CRES.	32	125	2
5									
6									

FIGURE 4.2.2.1-21 MISCELLANEOUS CLOSURES

4.2.2.1 (Continued)

In the tank pressure relief system the existing pressure switch (29.7 - 31.5 psia) will be replaced with a like pressure switch which has a pressure setting of 32.5 - 34.5 psia.

2. Fuel system

(a) Fuel fill and drain (60B43014)

To satisfy INT-20 loading criteria, the fuel loading probe will be lengthened 14 in. (FIGURE 4.2.2.1-22 &-23)

(b) Fuel feed system (60B43014)

All inboard fuel feed system hardware aft of the fuel suction elbows will be deleted (FIGURE 4.2.2.1-24).

(c) Fuel pressurization system (60B49600)

The helium supply and return ducts between the inboard engine interface and the respective helium manifolds will be deleted (FIGURE 4.2.2.1-25). The inboard engine branches from the supply and return manifolds will be capped (FIGURE 4.2.2.1-21, Items 2 & 3). All center engine oriented bolt-on brackets will be deleted.

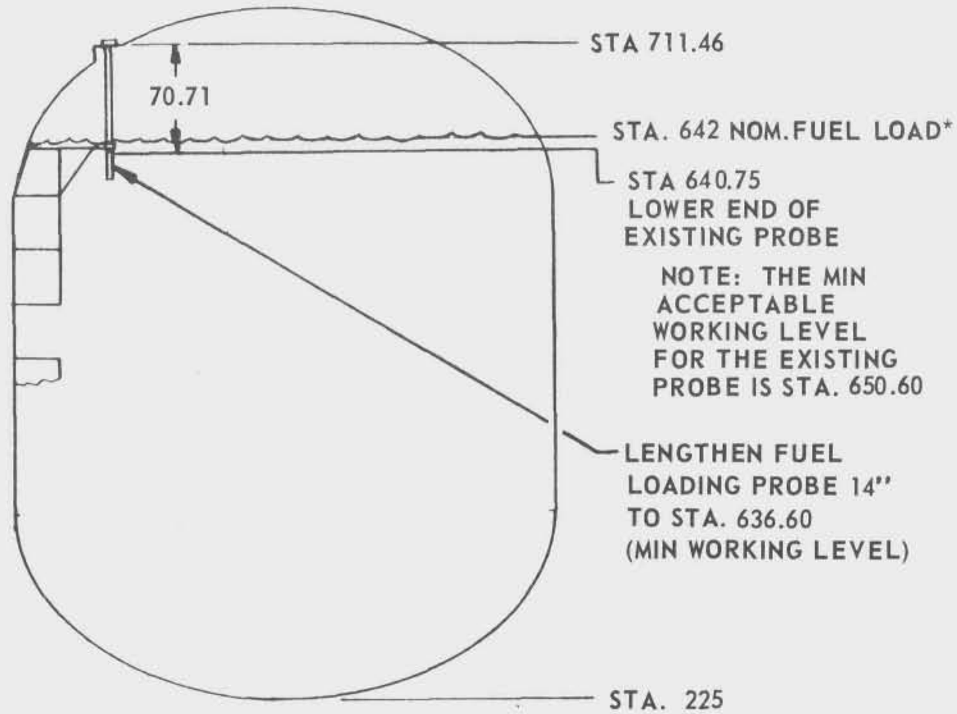
The five orifice plates in the helium pressurization supply control system located in the intertank area will be replaced with similar orifice plates of suitable orifice size. Orifice sizes and valve operating times will be established by final pressurization system analysis.

In the fuel pressurization control system a new pressure switch with a setting of 32.5 - 34.5 psia will replace the existing 27.5 - 29.0 psia pressure switch. For the fuel tank pressure relief system a pressure switch (35.7 - 38.5 psia) will be added for relief redundancy during launch and early flight. The existing relief pressure switch (29.7 - 31.5 psia) will be enabled at approximately T + 50 seconds. Replace the existing 60B49003-1 relief valve with the 60B49003-13 relief valve (used for static firing) which has a mechanical relief setting of 35.8 - 39.8 psia.

3. Auxiliary system

The S-IC auxiliary systems are divided into three general areas:

Control pressure system



*MIN. PREDICTED FUEL LOAD IS STA. 640
MAX. PREDICTED FUEL LOAD IS STA. 644

FIGURE 4.2.2.1-22 FUEL TANK LOADING PROBE

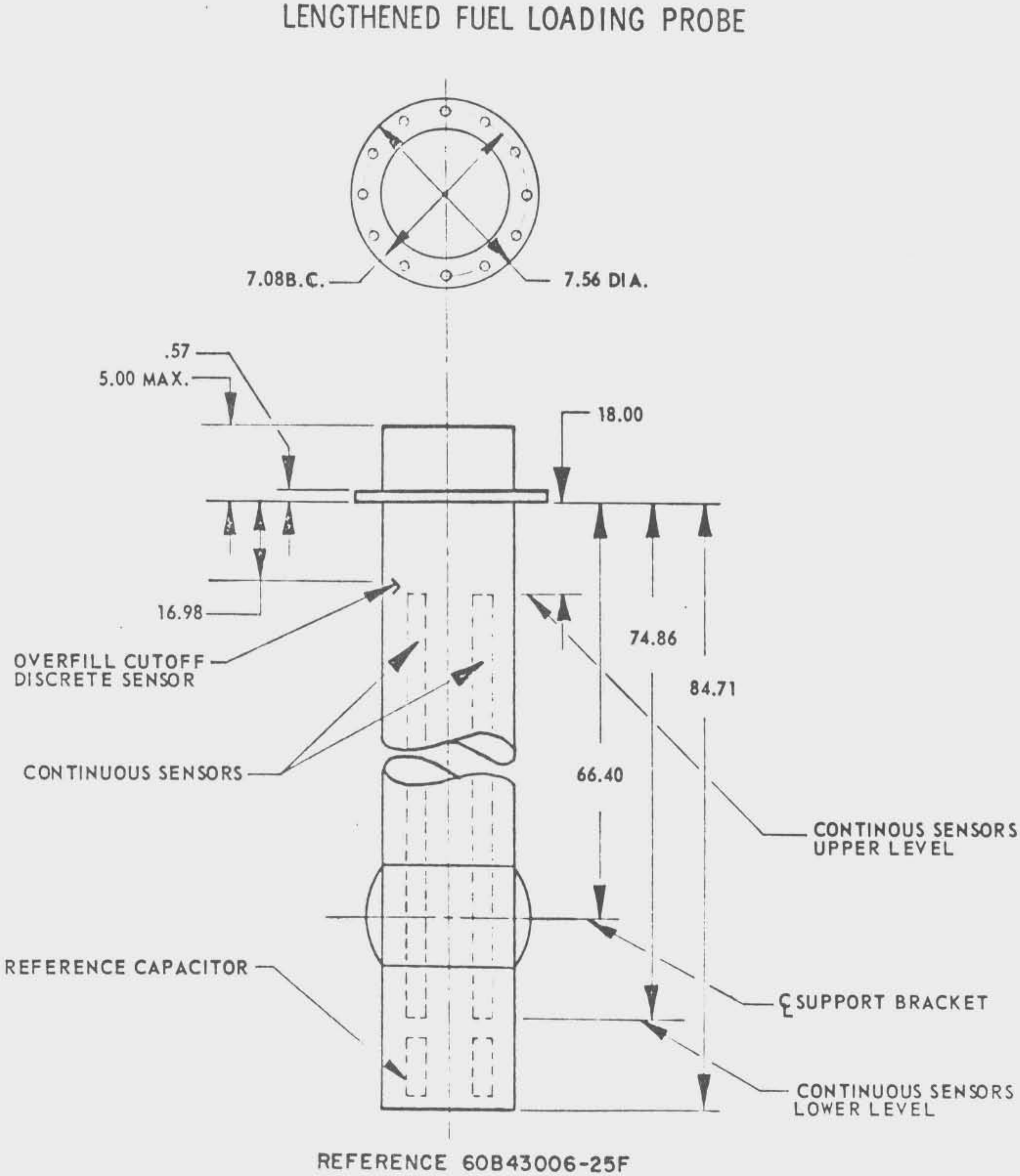


FIGURE 4.2.2.1-23 LENGTHENED FUEL LOADING PROBE

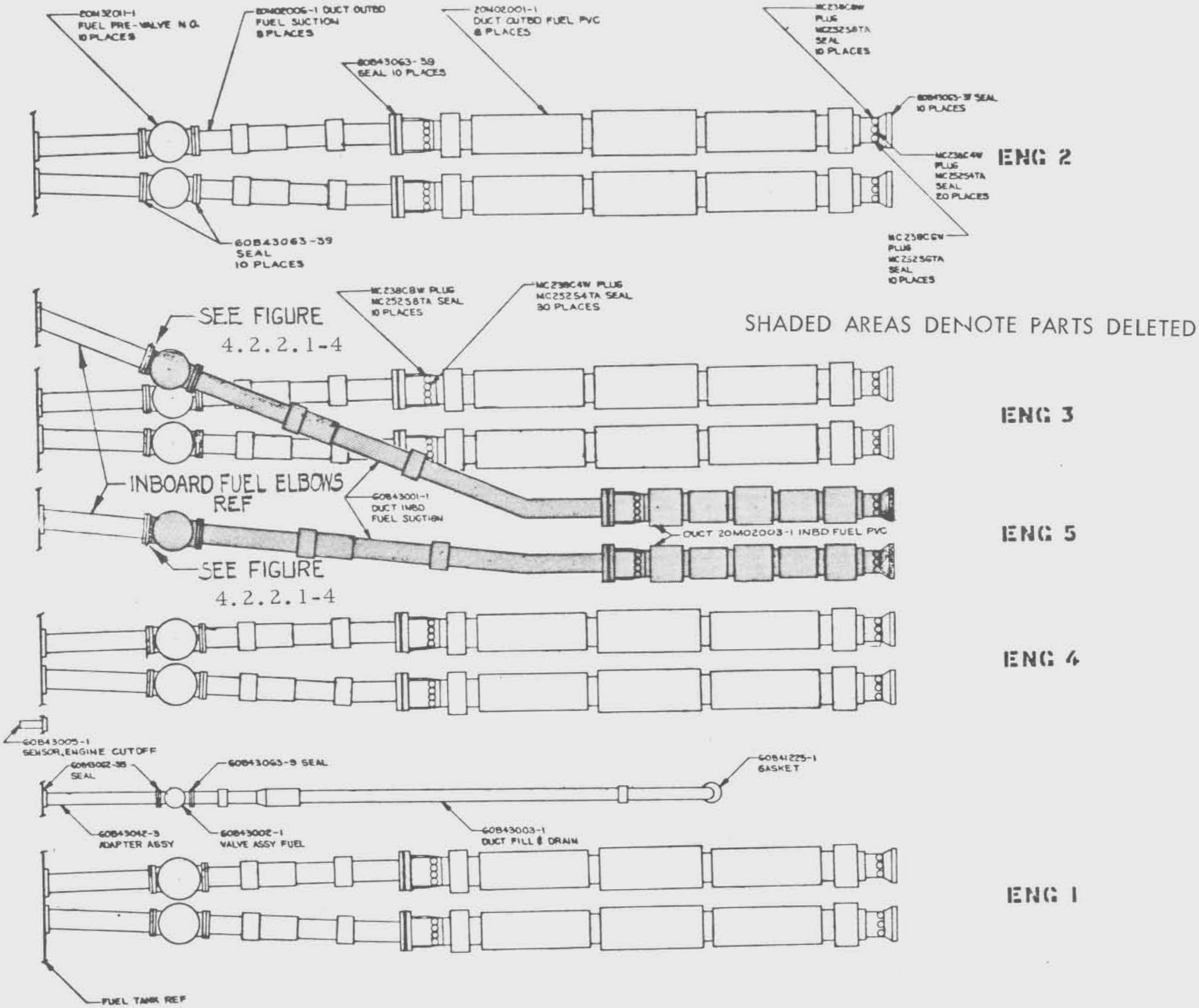


FIGURE 4.2.2.1-24 FUEL DELIVERY SYSTEM

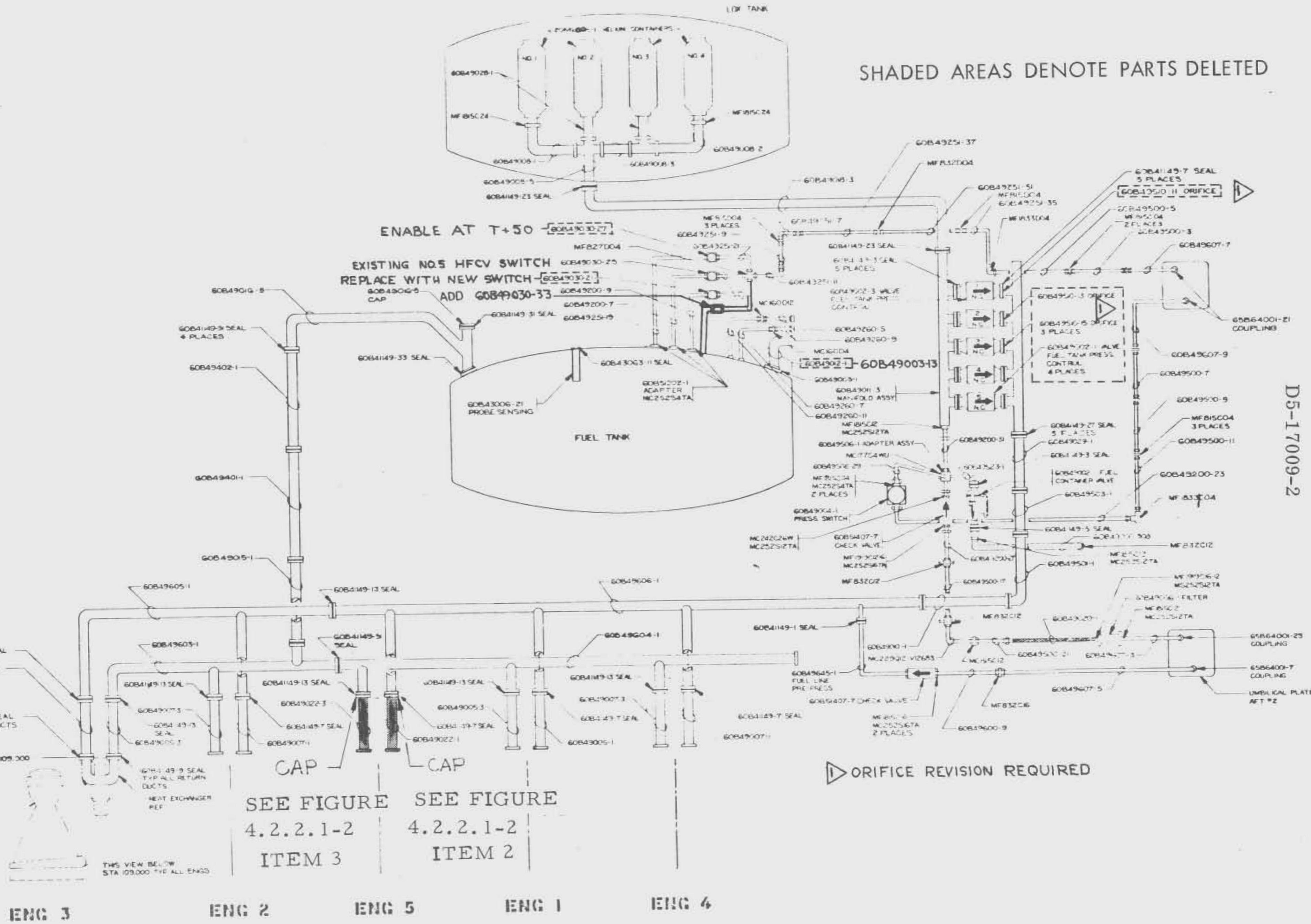


FIGURE 4.2.2.1-25 FUEL PRESSURIZATION SYSTEM

4.2.2.1 (Continued)

Environmental control system
Engine support purge systems

(a) Control pressure system (60B52500)

The following valves are currently operated by the S-IC onboard control pressure system.

- LOX fill and drain
- Fuel fill and drain
- LOX interconnect
- Fuel vent and relief
- LOX vent and relief
- LOX prevalves
- Fuel prevalves

Design changes to this system will consist of deleting the control pressure systems associated with the inboard engine prevalves and the No. 2 LOX interconnect valve (FIGURES 4.2.2.1-26 &-27).

(b) Environmental control system

There will be no design changes made to this system.

(c) Engine support purge systems

(1) Turbopump oxidizer seal (60B37601)

The turbopump oxidizer seal purge to the center engine will be deleted in its entirety on the operational configuration (FIGURE 4.2.2.1-28). On the first flight vehicles this system will be used as defined in 4.2.2.1.b.3.(c) (2), below.

(2) Radiation calorimeter purge

INT-20 instrumentation requirements call for a radiation calorimeter on the first flight stages. This calorimeter will be located in the base heat shield on a 24 inch radius at Position IV. The center engine turbopump oxidizer seal purge line will be modified as follows to purge this calorimeter.

The center engine turbopump oxidizer seal purge

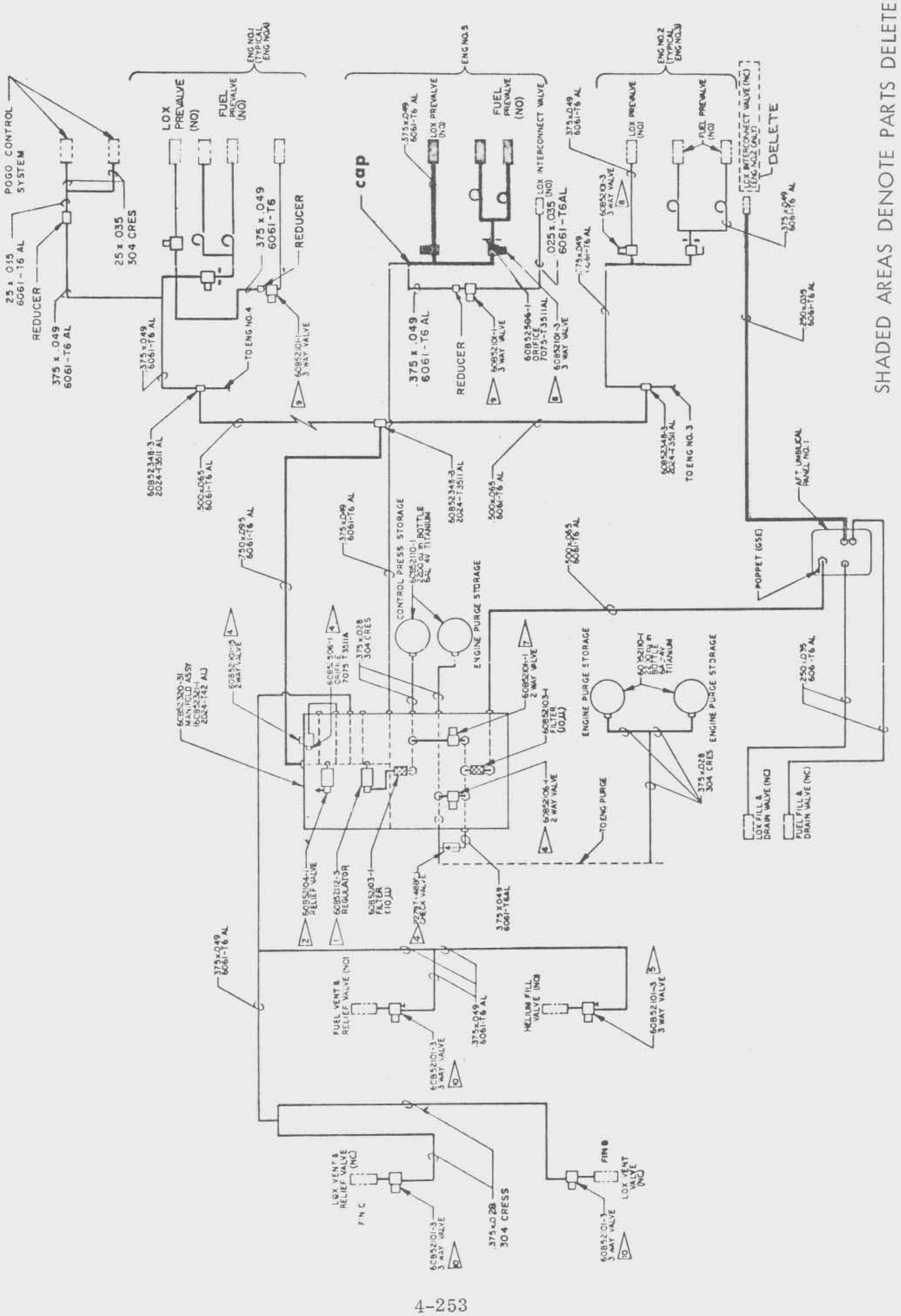


FIGURE 4.2.2.1-26 GN_2 CONTROL PRESSURIZATION

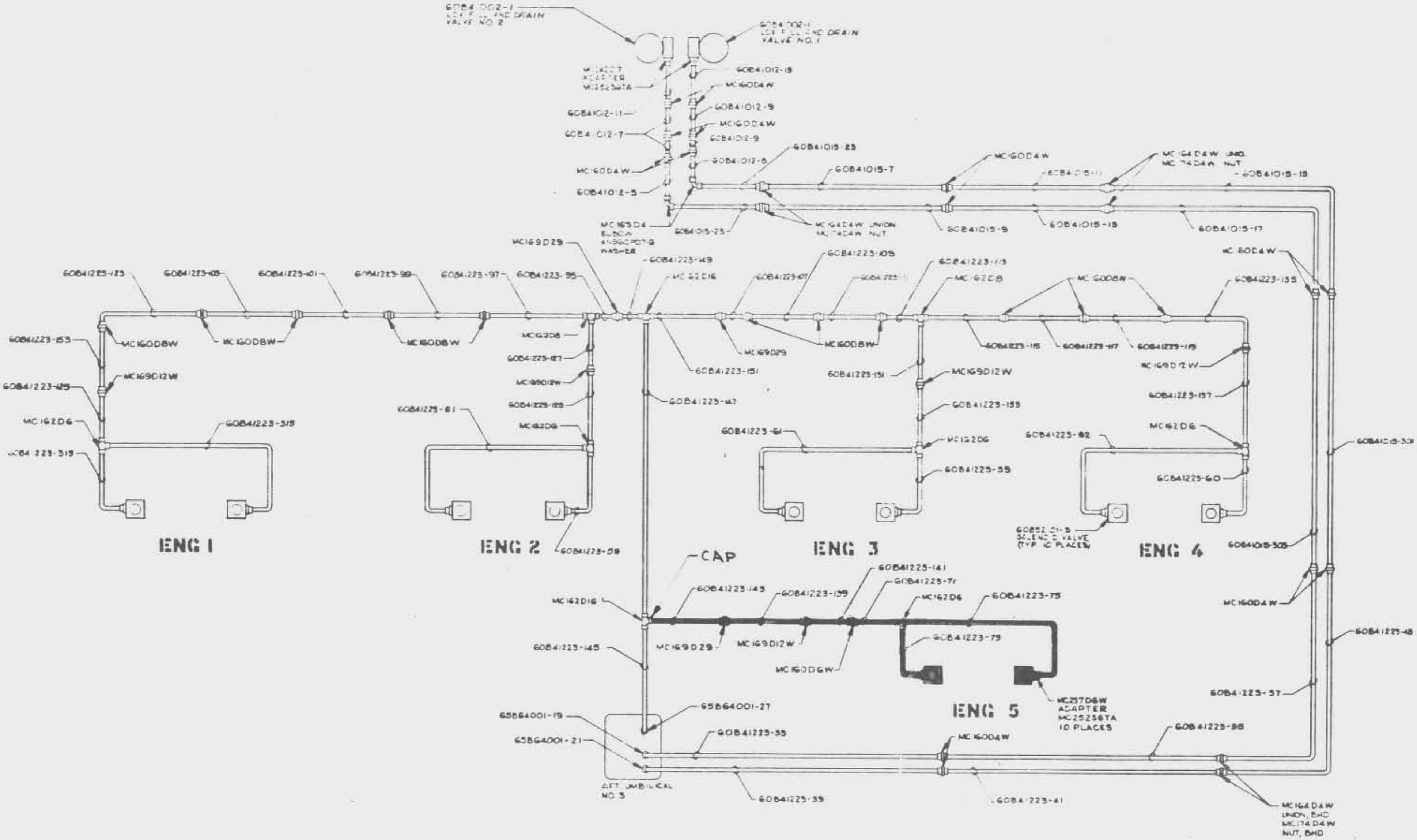
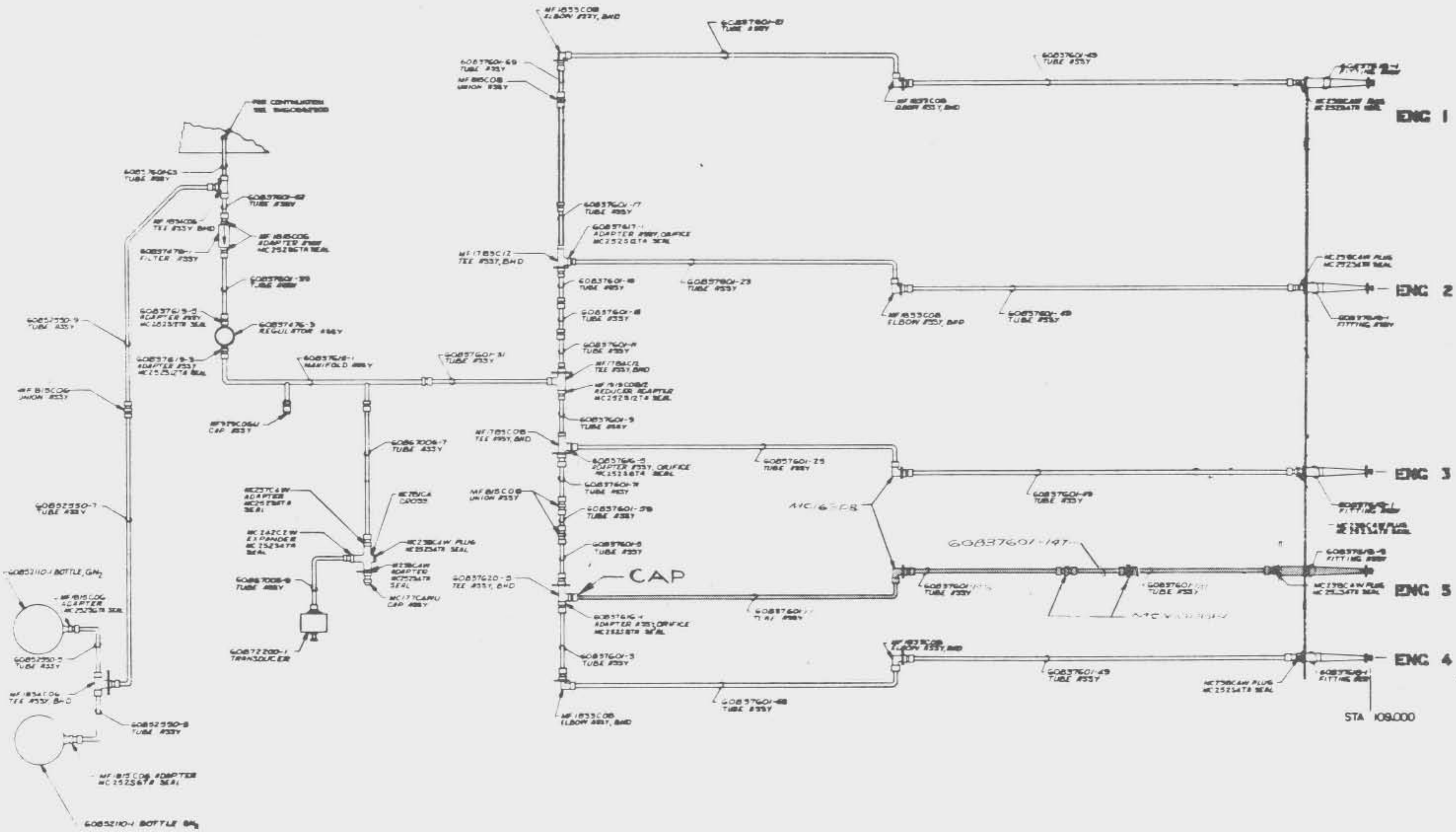
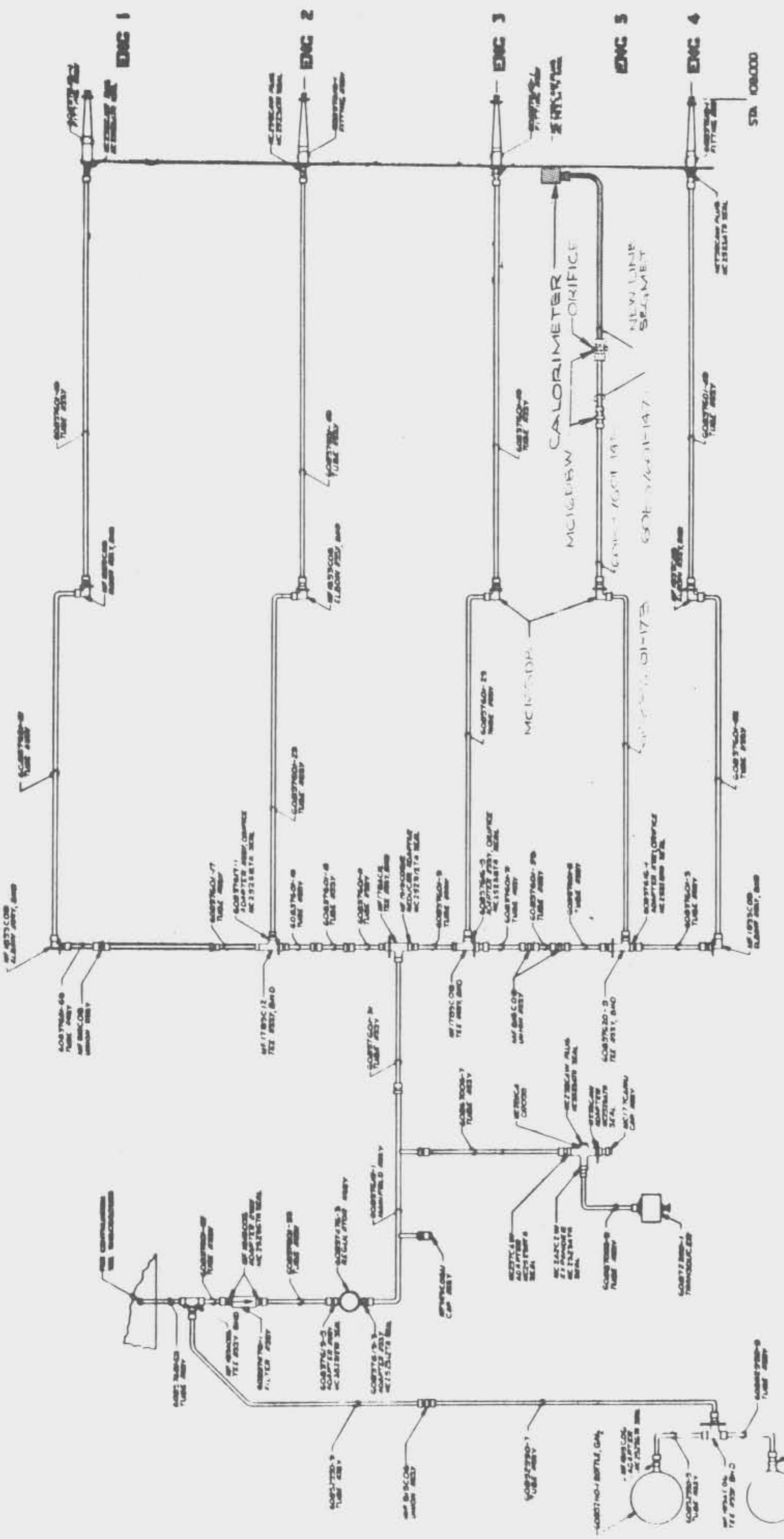


FIGURE 4.2.2.1-27 ON BOARD GN_2 CONTROL



SHADED AREA DENOTES PARTS DELETED

FIGURE 4.2.2.1-28 LOX SEAL, GG ACTUATOR HOUSING AND CALORIMETER PURGE SYSTEMS (OPERATIONAL CONFIGURATION)



SHADED AREA DENOTES PARTS ADDED

FIGURE 4.2.2.1-29 LOX SEAL, GG ACTUATOR HOUSING AND CALORIMETER SYSTEMS (FIRST FLIGHT CONFIGURATION)

4.2.2.1 (Continued)

line will be deleted from the engine interface fitting to the first upstream union. An orifice will be installed in that union and a new line segment will be added from the union to the above calorimeter (FIGURE 4.2.2.1-29).

(3) LOX dome and gas generator LOX injector purge (60B37600)

The center engine branch line will be deleted and the manifold duct assembly will be plugged (FIGURE 4.2.2.1-30).

(4) Engine cocoon thermal conditioning purge (60B37602)

The manifold tee supplying the center engine will be plugged and all tube assemblies down-stream of that fitting will be deleted (FIGURE 4.2.2.1-30).

(5) Thrust OK checkout system (60B37600)

The center engine branch tee will be capped and all tube assemblies downstream of that fitting will be deleted (FIGURE 4.2.2.1-31).

(6) Thrust chamber prefill system (60B37550)

All hardware downstream of the center engine tee will be deleted and the tee will be plugged (FIGURE 4.2.2.1-32).

(7) POGO suppression system (60B41840)

The following design change definition is based on the current POGO suppression system configuration (ECP-446R3 and ECP-512R2) which includes provisions for supplying helium to the center engine prevalve.

The tee supplying the center engine will be plugged and all downstream tubing will be deleted (FIGURE 4.2.2.1-33).

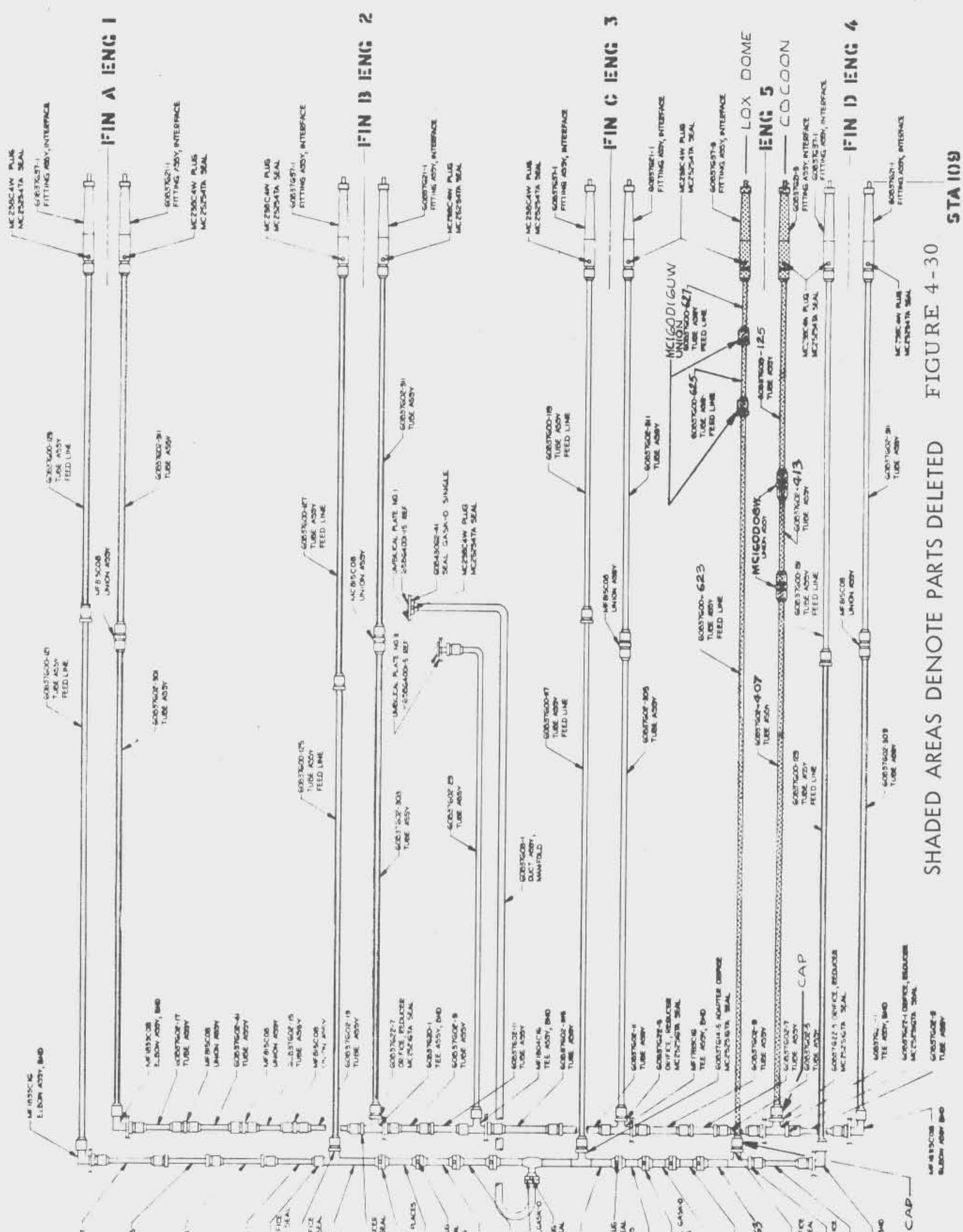


FIGURE 4.2.2.1-30 LOX DOME, GG LOX INJECTOR AND ENGINE COCOON PURGE SYSTEM

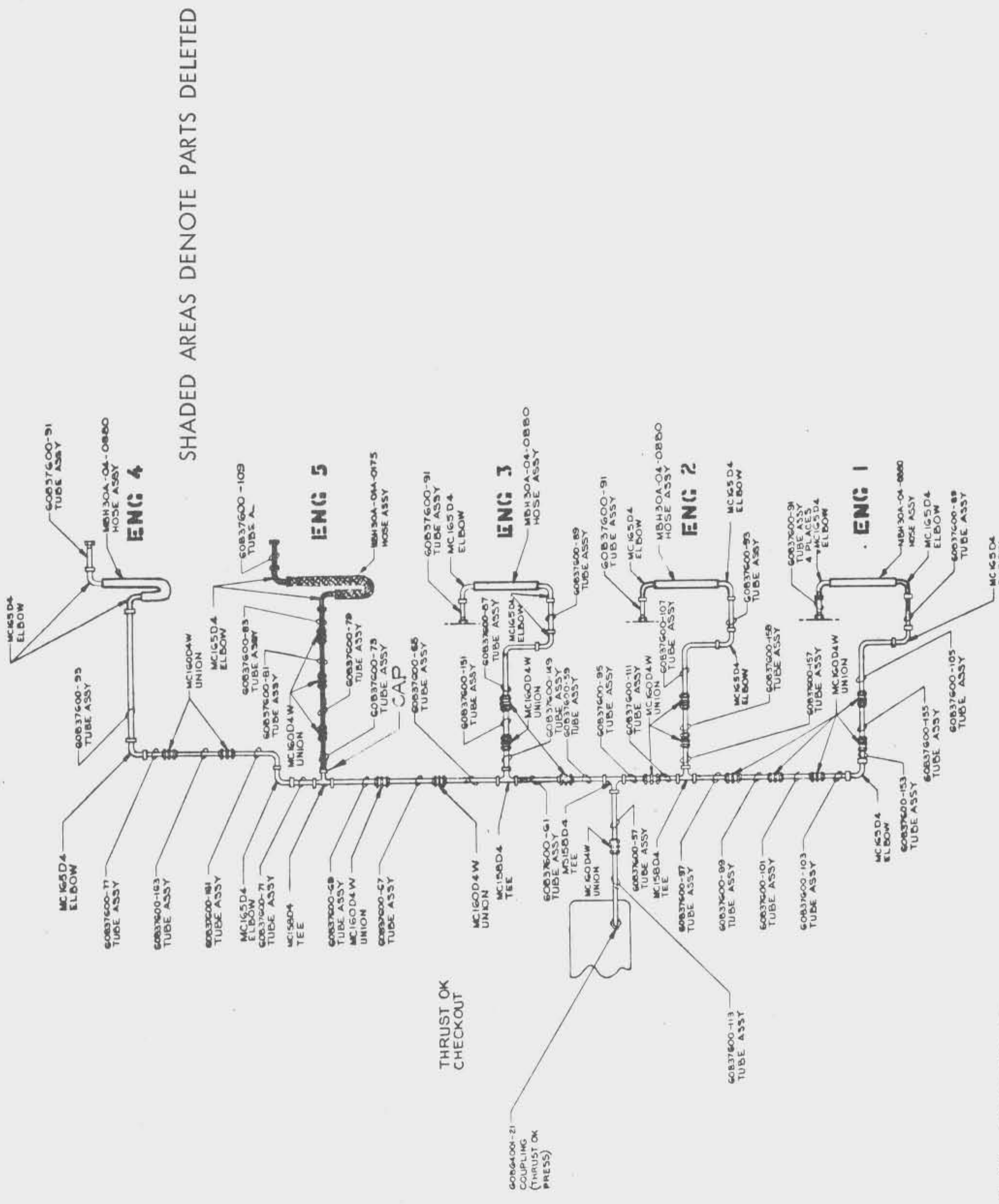


FIGURE 4.2.2.1-31 THRUST OK CHECKOUT SYSTEM

SHADED AREAS DENOTE PARTS DELETED

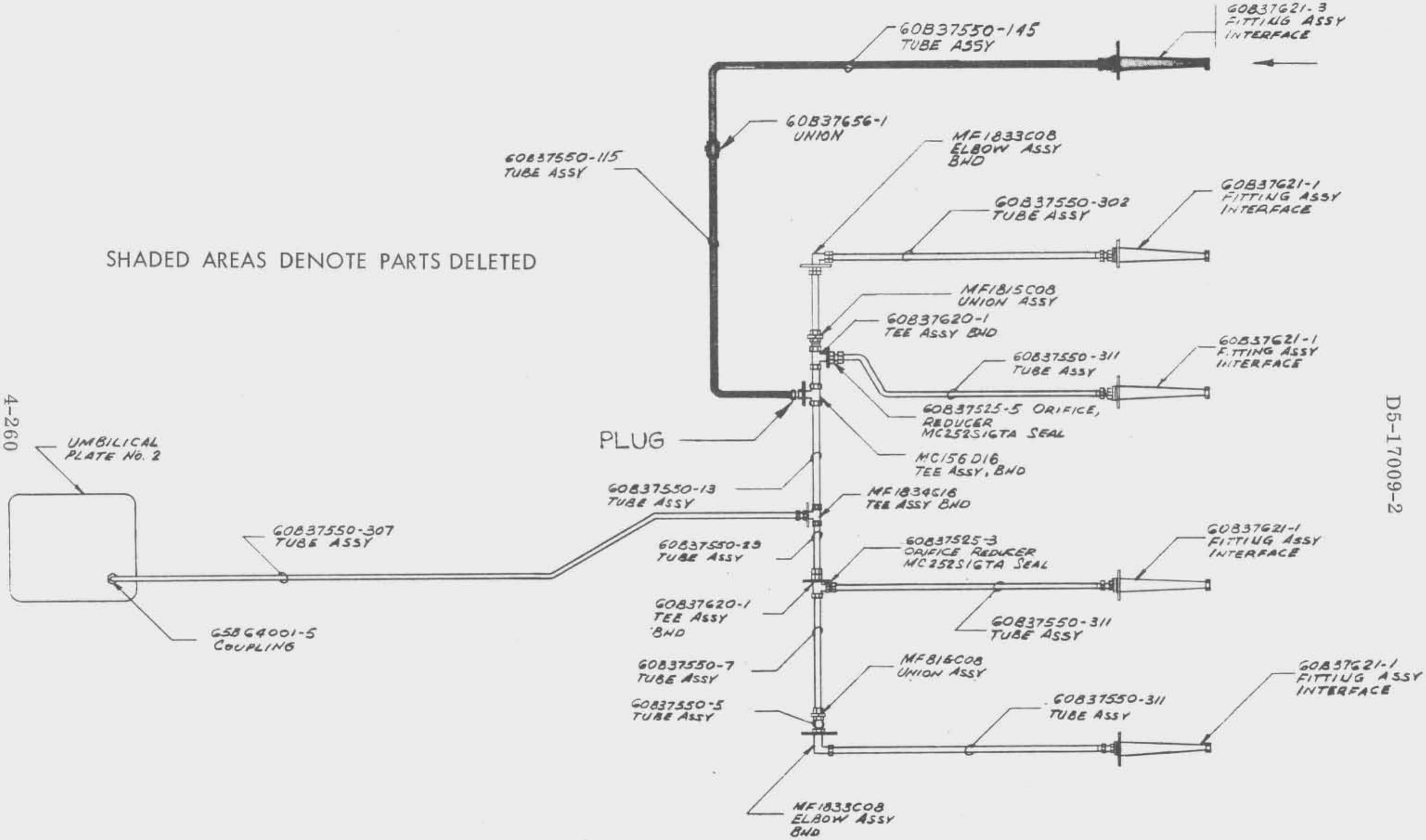
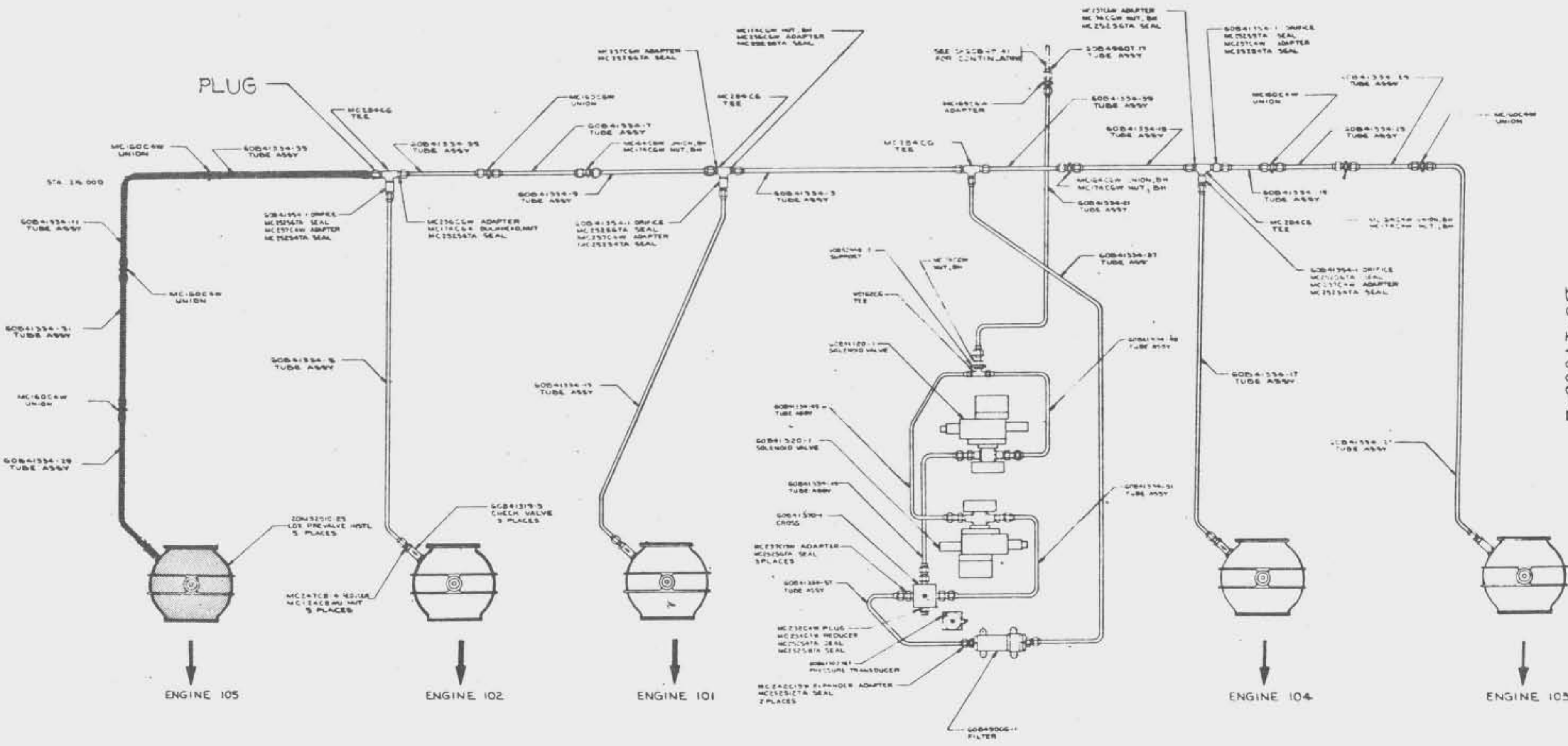
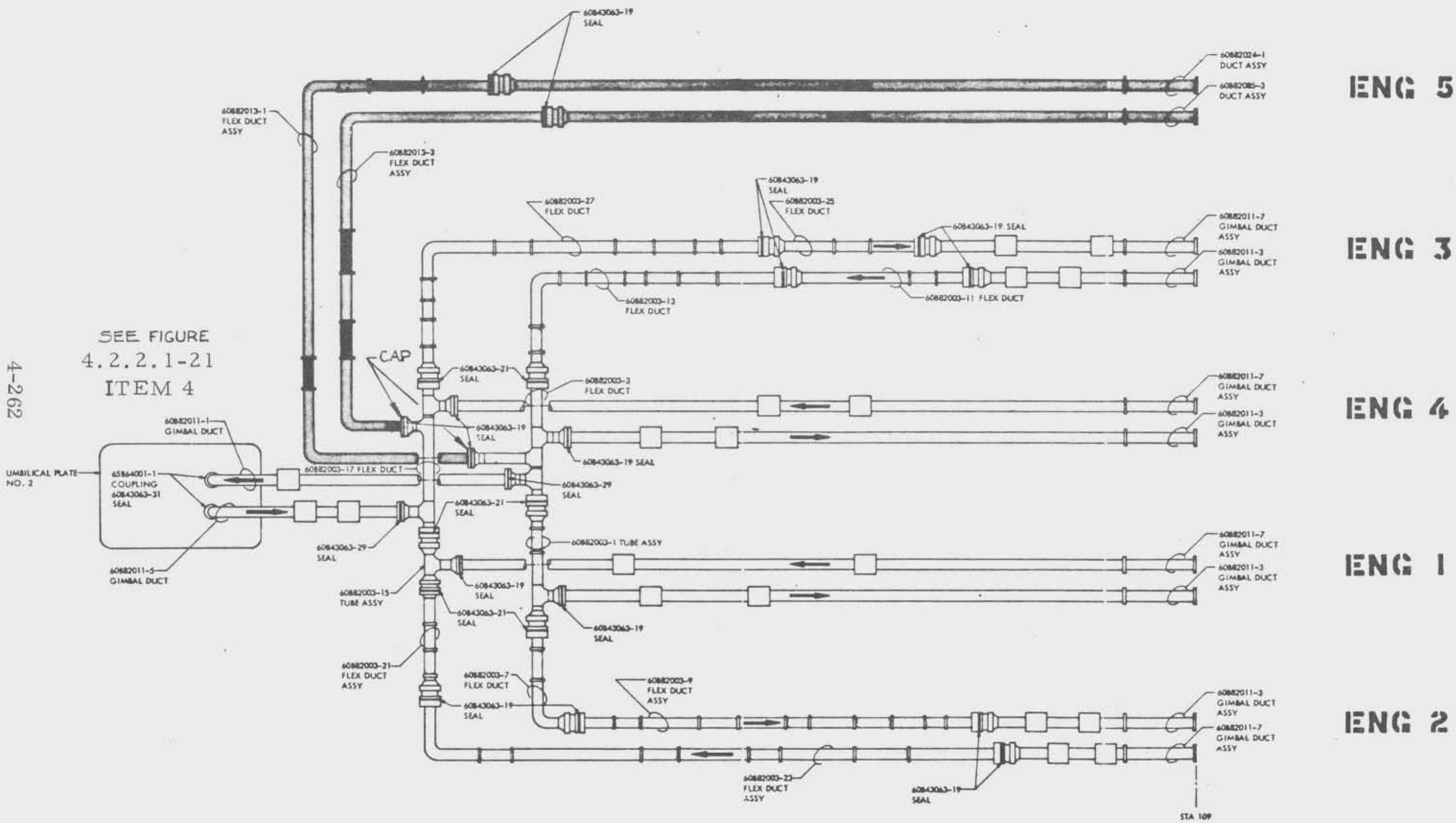


FIGURE 4.2.2.1-32 PRE-FILL SYSTEM



SHADED AREAS DENOTE PARTS DELETED

FIGURE 4.2.2.1-33 POGO SUPPRESSION SYSTEM



SHADED AREAS DENOTE PARTS DELETED

FIGURE 4.2.2.1-34 FLUID POWER SYSTEM

4. Flight control subsystem

The flight control subsystem is made up of the fluid power system and the thrust vector control system.

(a) Fluid power system (60B82000)

Design changes to the fluid power system will consist of (1) deleting the center engine ground hydraulic supply and return ducting (FIGURE 4.2.2.1-34), and (2) capping the center engine branches on the supply and return duct manifolds (FIGURE 4.2.2.1-21, Item 4).

(b) Thrust vector control system (60B84000)

There will be no design changes made to this system.

5. Engine and related components (60B37450)

The design changes required to the engines and related components will consist of deleting the center F-1 engine (including loose equipment), the associated static firing GN_2 purge, all center engine attachment and support hardware, and the center engine thermal insulation.

6. Propulsion/Mechanical systems supplemental data

The identification of the above changes with INT-20 criteria is contained in Appendix A, Section 1.0. Section 2.0 of Appendix A contains configuration trade studies and technical support study data. Section 3.0 is a listing of parts deleted, added or revised and their respective weights.

4.2.2.1 (Continued)

c. Electrical/Electronic Subsystems

The Electrical/Electronic Subsystems design changes consist of deactivating center engine circuitry and measurements, revising the engine cutoff and fuel tank vent circuitry, adding measurements, revising the S-IC/S-IVB functional interface, and lengthening the interface cabling to the S-IVB stage. These design changes represent a minimal impact and provide for configuration reversibility.

1. Power Generation and Distribution

There are no required design changes to the electrical power system. Power distribution changes will be implemented by adding or deleting distributor wiring and adding or stowing cabling. Stowed cabling with pins at potentials above ground will be identified with tags.

2. S-IC/S-IVB Interface

The S-IC/S-IVB functional interface will be changed to delete three center engine thrust OK measurements and add a simulated S-II/S-IVB separation indication to the I. U., as shown in Table 4.2.2.1-I. Interface cables used to route signals to the I. U. will be lengthened, as shown in Figure 4.2.2.1-35

3. Sequence and Control

(a) S-IC Stage Functions

Center engine circuitry will be deactivated by deleting distributor wiring and stowing unused cabling. The unused components and cabling will not be deleted except the engines 2, 4 and 5 LOX Level Cutoff Sensors.

The engine cutoff circuitry will be modified to provide a sequenced cutoff of engines 2 and 4 and engines 1 and 3 by independent I. U. commands, through the switch selector. The normal sequence will be engines 2 and 4 cutoff at approximately 146 seconds and engines 1 and 3 cutoff at approximately 211 seconds. The engine cutoff commands are initiated by the I. U. by a longitudinal acceleration "g" limit. An I. U. command is also provided to enable propellant depletion cutoff. The LOX depletion cutoff circuitry will be revised to add redundant sensors for engines 1 and 3, sensors for engines 2 and 4 will be deleted.

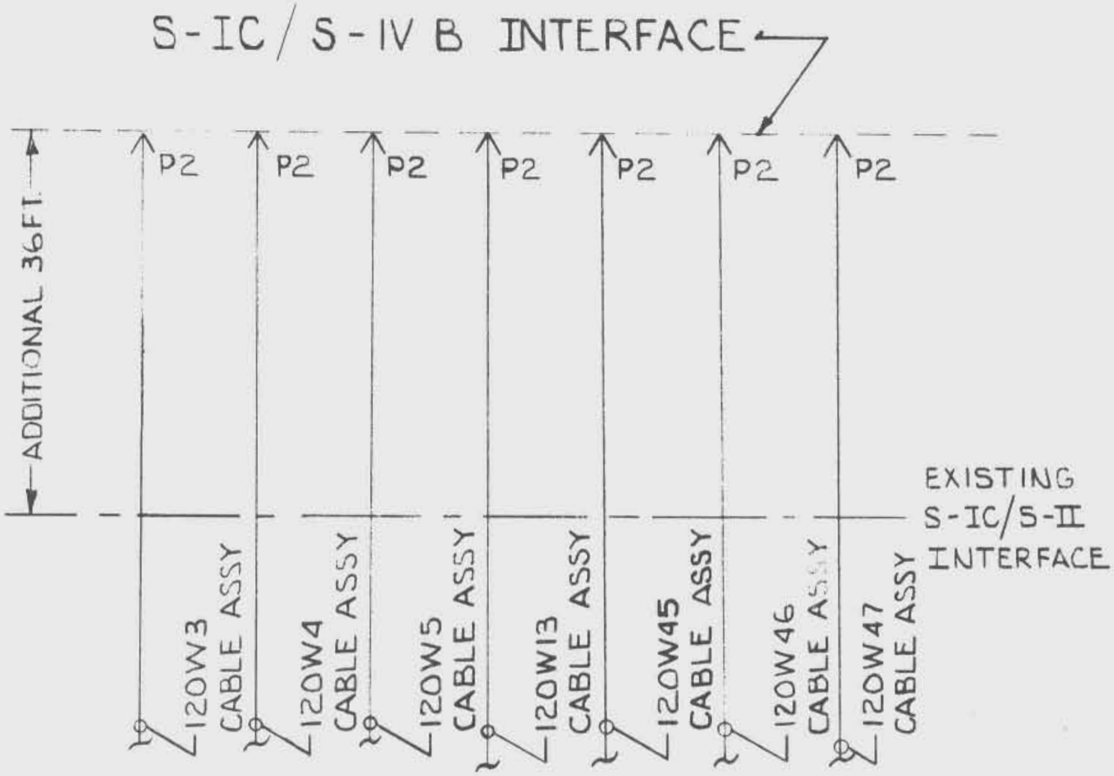


FIGURE 4.2.2.1-35 S-IC/S-IVB INTERFACE CABLING CHANGE

TABLE 4.2.2.1-I S-IC/S-IVB FUNCTIONAL INTERFACE CHANGES

CABLE	CONNECTOR/PIN	FUNCTION	REMARKS
120W45	P2-E	Meas. Engine No. 5 Thrust Relay No. 1 Thrust OK	Delete
120W46	P2-E	Meas. Engine No. 5 Thrust Relay No. 2 Thrust OK	Delete
120W47	P2-E	Meas. Engine No. 5 Thrust Relay No. 3 Thrust OK	Delete
120W13	P2-G	Meas. Simulated S-II/S-IVB Separation	Add

4.2.2.1 (Continued)

In the event engine 1 or 3 is cutoff prior to engines 2 and 4, the I.U. will reverse the engine cutoff sequence. The reverse sequence will result in cutoff of the remaining engine 1 or 3 at approximately 146 seconds and cutoff of engines 2 and 4 prior to possible propellant depletion. Existing circuitry will allow the I.U. to detect premature cutoff of engine 1 or 3.

The capability will be maintained to initiate thrust not OK cutoff of any engine or Range Safety, Emergency Detection System, or two adjacent engines out cutoff of all engines. Functional diagrams of the present and proposed S-IC engine cutoff circuitry are shown in Figures 4.2.2.1-36 and 4.2.2.1-37, respectively. Figure 4.2.2.1-38 is a functional schematic diagram of the proposed S-IC engine cutoff circuitry.

The engine cutoff circuitry modification will be accomplished by revising distributor wiring, lengthening two cable branches and adding redundant LOX level sensors.

The fuel tank vent and relief pressure system will be modified to add an additional pressure switch and inhibit the present pressure switch until T + 50 seconds. The new pressure switch will be utilized from prepressurization to T + 50 seconds. This change will be implemented by revising cabling, a junction box and distributor wiring and adding a relay card. Figure 4.2.2.1-39 is a functional schematic of the proposed change.

(b) I.U. Functions

The I.U. will be revised to provide a normal or reverse engine cutoff sequence. The normal sequence consists of providing "g" limit cutoff commands for engines 2 and 4 at approximately 146 seconds and for engines 1 and 3 at approximately 211 seconds. In the event engine 1 or 3 is cutoff prior to engines 2 and 4, the I.U. will utilize a reverse sequence. The reverse sequence results in cutoff of the remaining engine 1 or 3 at approximately 146 seconds and cutoff of engines 2 and 4 prior to possible propellant depletion. The I.U. will also provide propellant depletion and fuel vent and relief valve enable commands, as listed in Table 4.2.2.1-II.

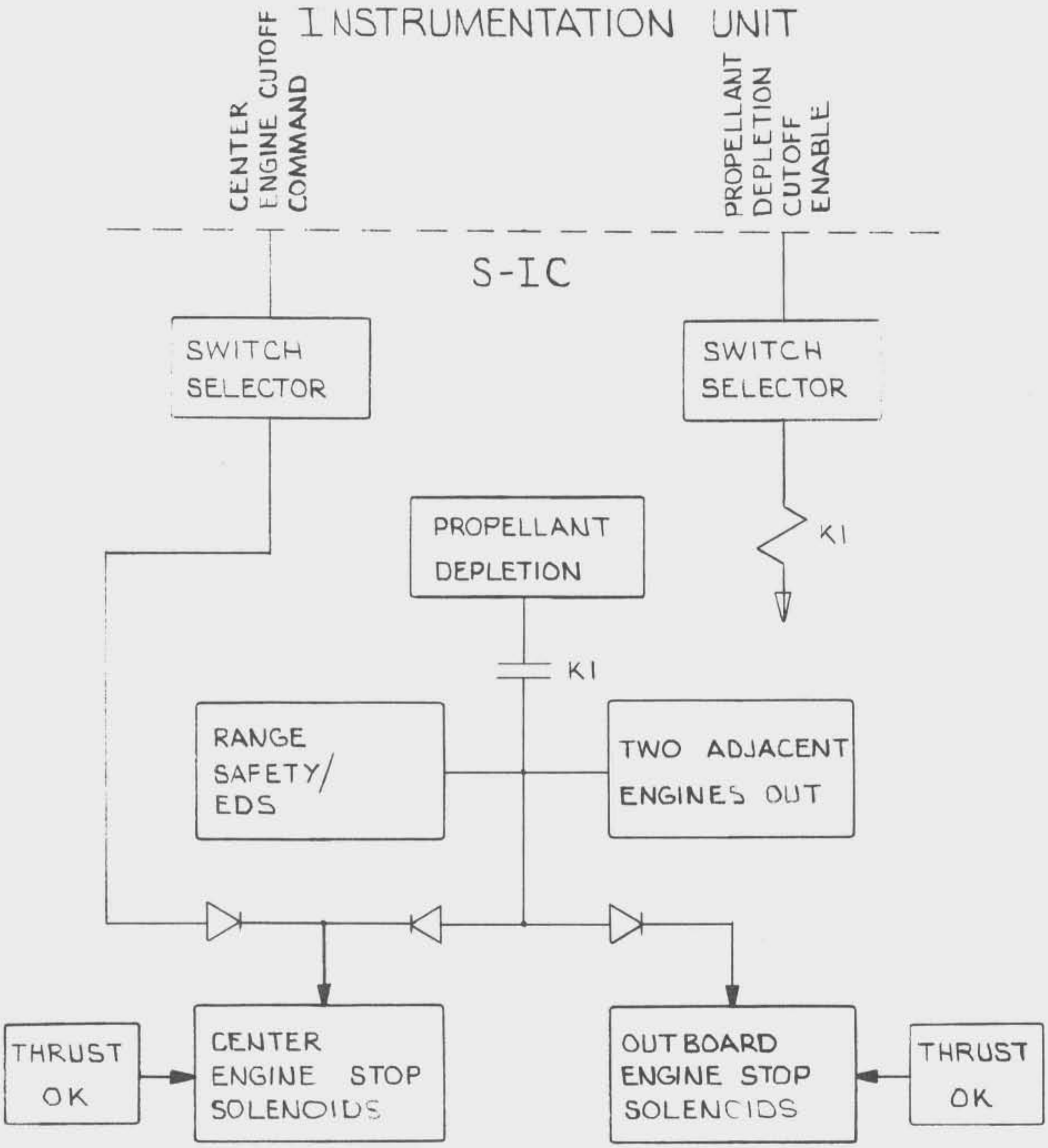


FIGURE 4.2.2.1-36 PRESENT S-IC ENGINE CUTOFF FUNCTIONAL DIAGRAM

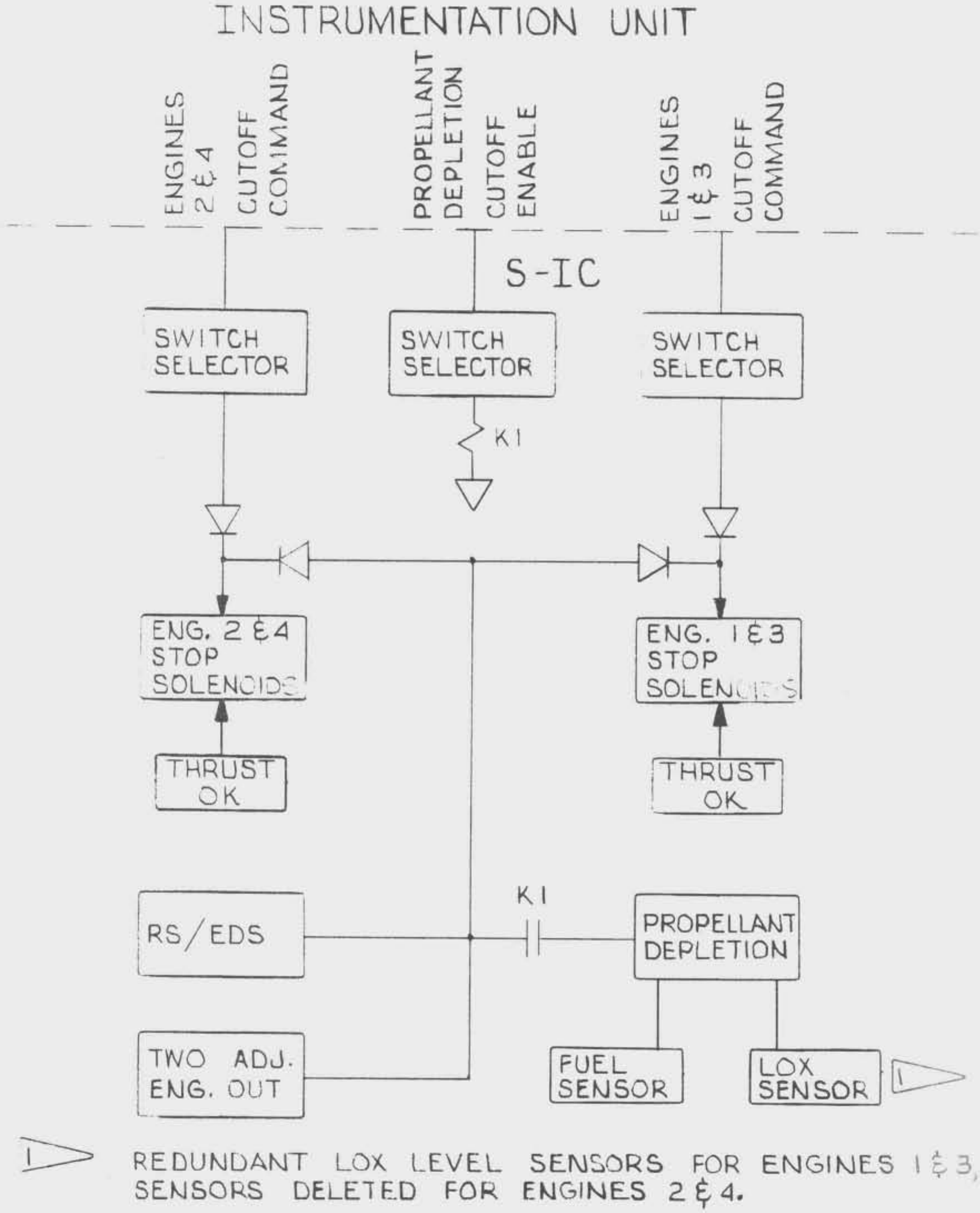


FIGURE 4.2.2.1-37 PROPOSED S-IC ENGINE CUTOFF FUNCTIONAL DIAGRAM

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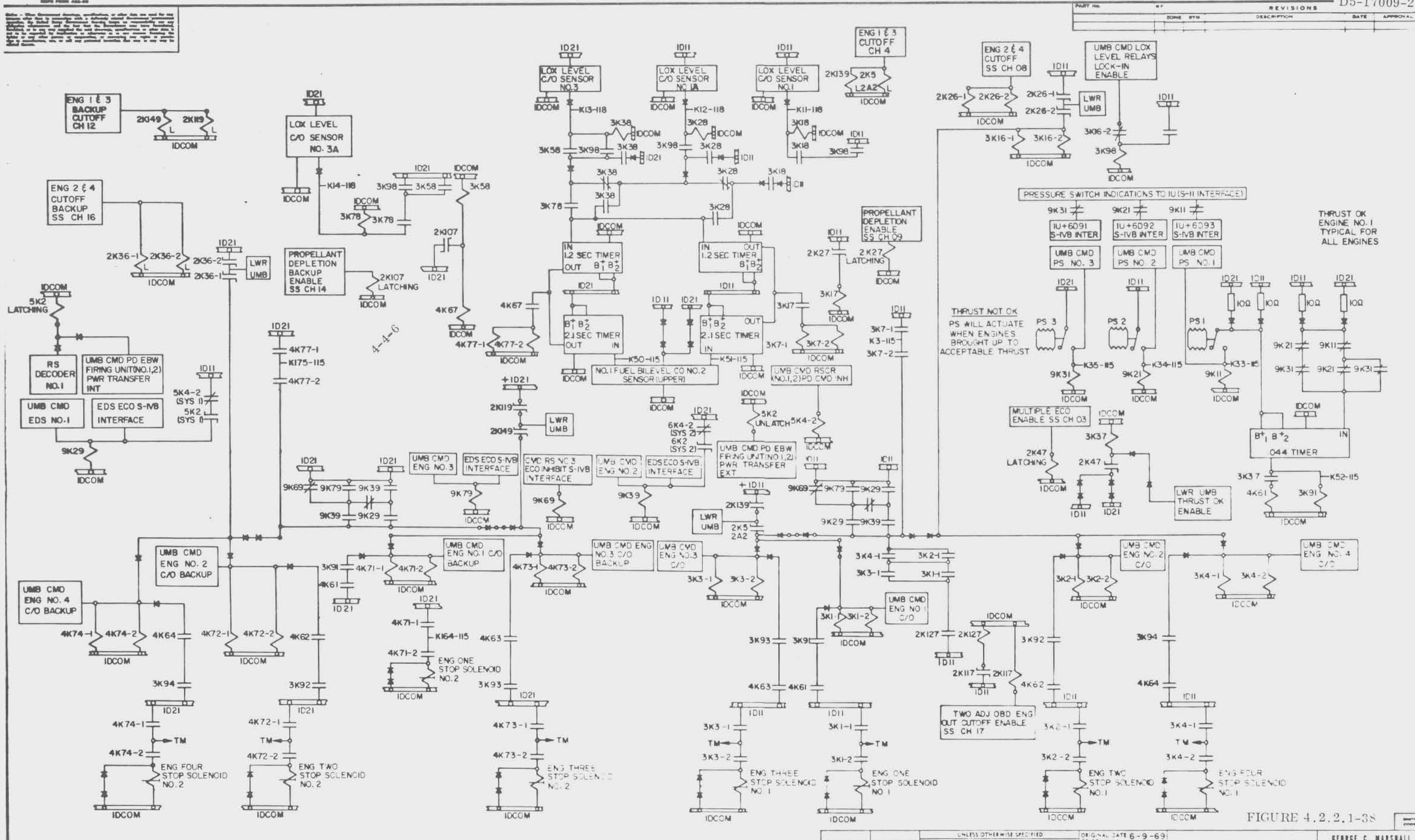

FIGURE 4.2.2.1-38 PROPOSED ENGINE CUTOFF FUNCTIONAL SCHEMATICS

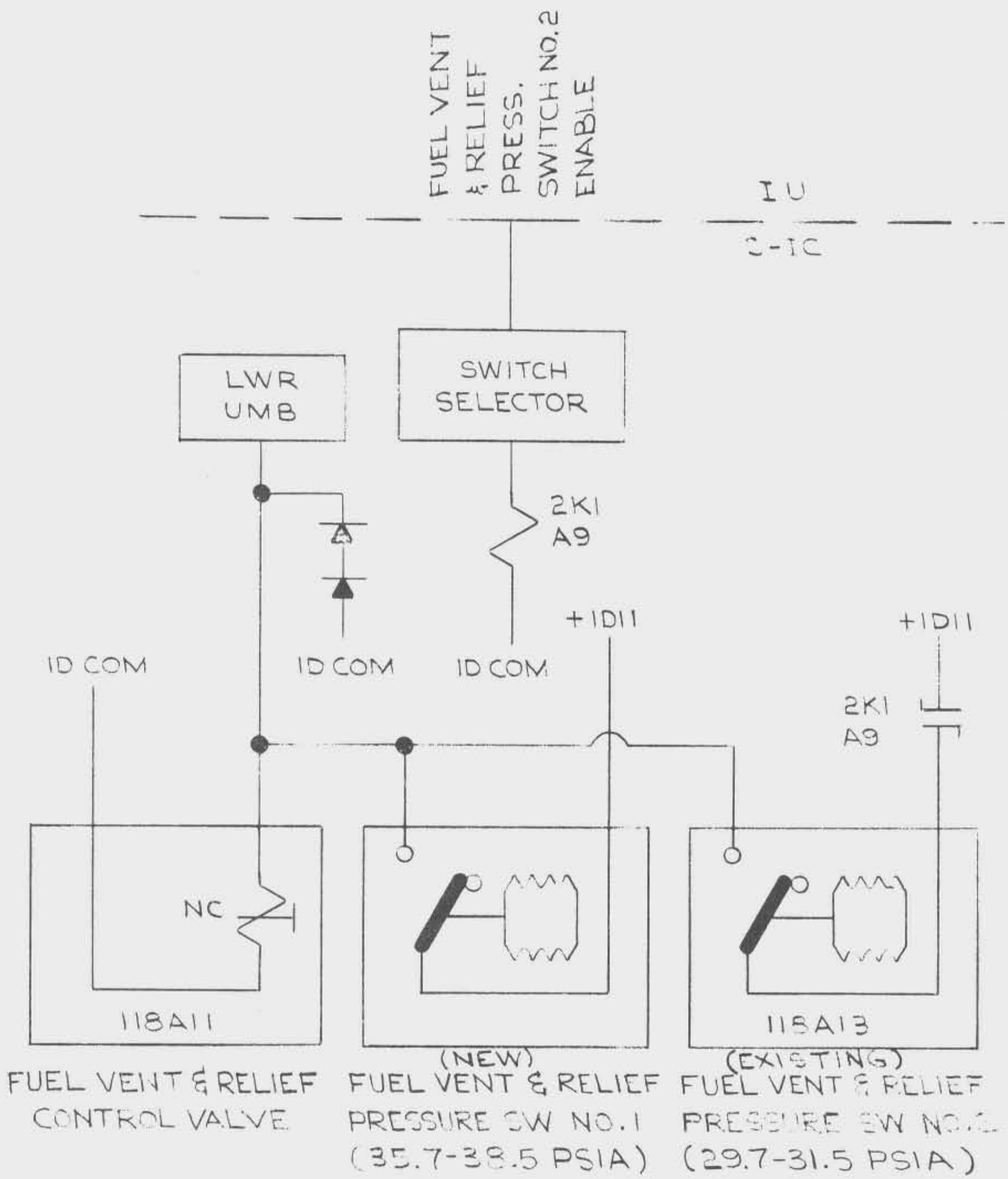

FIGURE 4.2.2.1-39 PROPOSED FUEL TANK VENT AND RELIEF PRESSURE SYSTEM

TABLE 4.2.2.1-II

INSTRUMENTATION UNIT CHANGES

SWITCH SELECTOR CHANNEL	FUNCTION	REMARKS
8	Engines 2 and 4 Cutoff Command	1 - Approx. 146 seconds or 4.68 "g" limit. 2 - Prior to possible propellant depletion.
16	Engines 2 and 4 Cutoff Command Backup	1 - Approx. 146 seconds or 4.68 "g" limit. 2 - Prior to possible propellant depletion.
9	Propellant Depletion Enable	Prior to possible propellant depletion.
14	Propellant Depletion Backup Enable	Prior to possible propellant depletion.
4	Engines 1 and 3 Cutoff Command	1 - Approx. 211 seconds or 4.68 "g" limit. 2 - Approx. 146 seconds or 4.68 "g" limit.
12	Engines 1 and 3 Cutoff Command Backup	1 - Approx. 211 seconds or 4.68 "g" limit. 2 - Approx. 146 seconds or 4.68 "g" limit.
13	Fuel Vent and Relief Pressure Switch No. 2 Enable	Approx. 50 seconds.

1 - Applicable for the normal engine cutoff sequence.

2 - Applicable for the reverse engine cutoff sequence.

4.2.2.1 (Continued)

4. Emergency Detection System

Design changes are not required to the Emergency Detection System.

5. Range Safety System

Design changes are not required to the Range Safety System.

6. Separation and Ordnance System

Separation and ordnance system components presently supplied with the S-IC stage for installation on the S-II stage, as shown in Figure 4.2.2.1-40, will be installed on the S-IVB stage, as shown in Figure 4.2.2.1-41. Interface cabling will be lengthened to mate with these components, as shown in Figure 4.2.2.1-42.

7. Propellant Loading System

The Propellant Loading System electronics design will not be changed. The fuel loading probe installation will be revised to lengthen the probe, to accommodate required fuel loading levels.

8. Measuring System

(a) Measurements

The Instrumentation System will be changed to deactivate 39 measurements and add 19 measurements. Measurements will be deactivated by deleting instrumentation and distributor wiring and stowing cabling. The additional measurements will require installation of transducers, zone boxes, and amplifiers, as listed in Table 4.2.2.1-III. The additional measurements will be effective for the first two stages only.

(b) Telemetry System

The Telemetry System design will not be changed. The Instrumentation Program and Components List will be modified to incorporate addition and deletion of measurements. Unused telemetry channels will be grounded by adding distributor wiring. The measurement program consists of a total of 292 measurements, 19 of these measurements are effective for the first two stages only.

SAT V
EBW FIRING UNIT, DETONATOR BLOCK,
FIRST PLANE SEPARATION SYSTEM

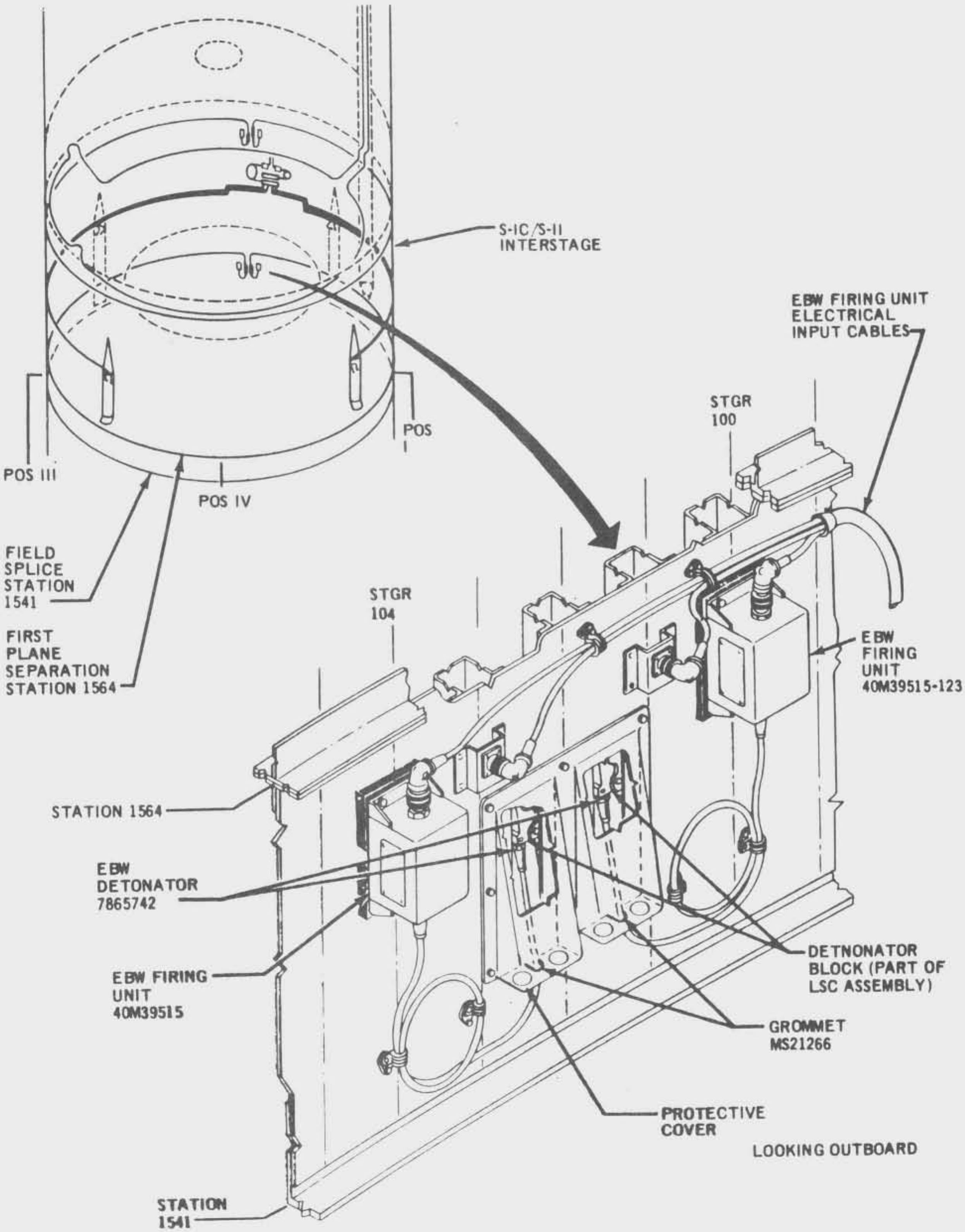

FIGURE 4.2.2.1-40 PRESENT S-IC SEPARATION AND ORDNANCE SYSTEM

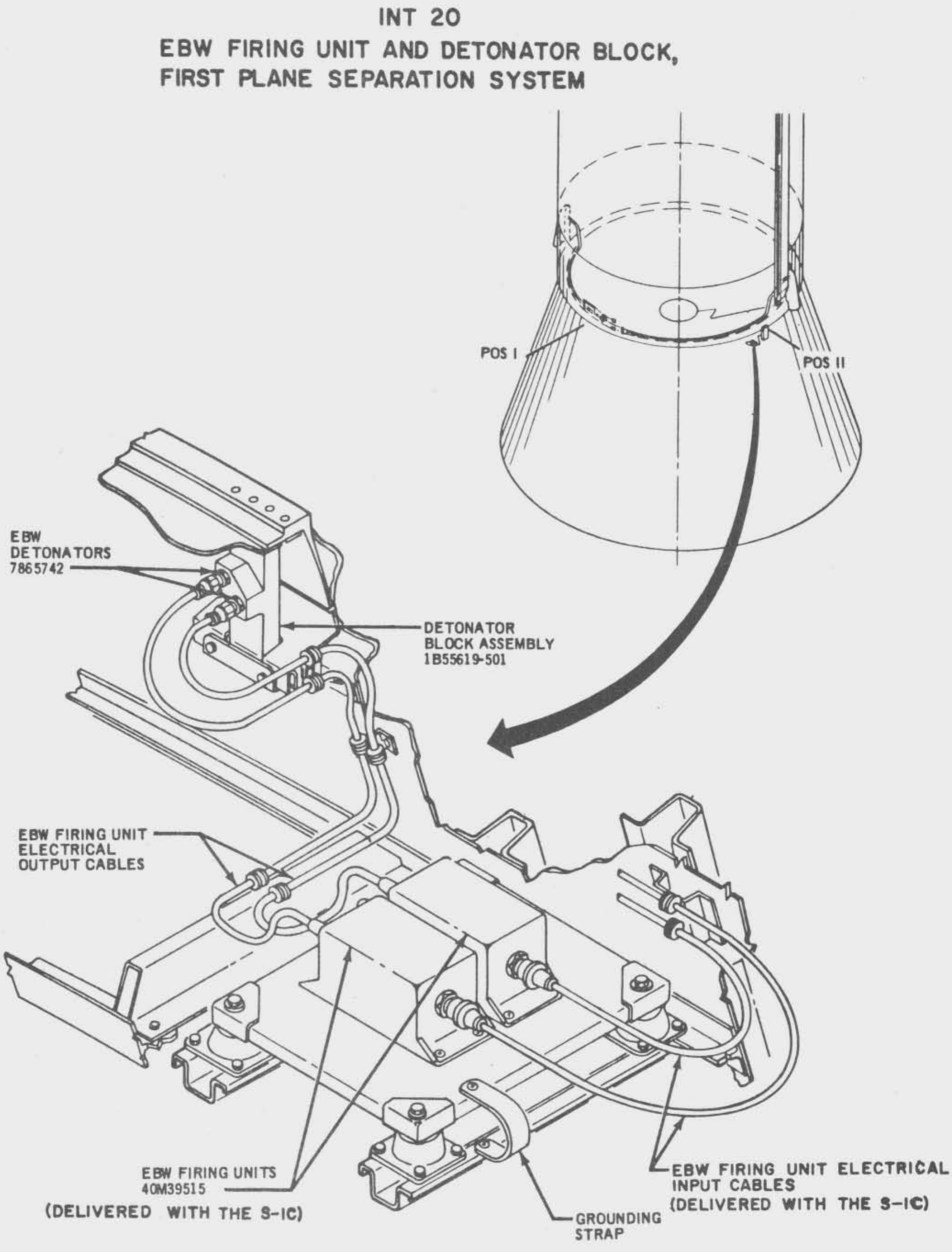

FIGURE 4.2.2.1-41 PROPOSED S-IC SEPARATION AND ORDNANCE SYSTEM

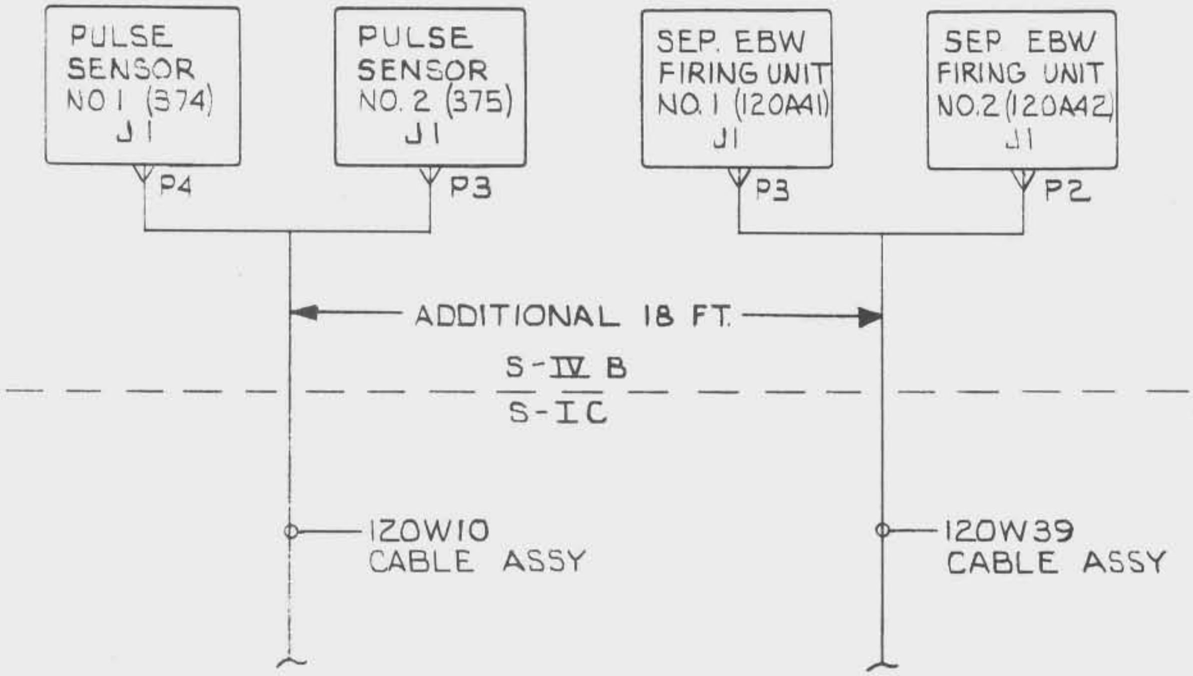

FIGURE 4.2.2.1-42

FIGURE 4.2.2.1-42 PROPOSED SEPARATION AND ORDNANCE SYSTEM CABLING CHANGE

TABLE 4.2.2.1-III

ADDITIONAL INSTRUMENTATION 1

MEASUREMENT/COMPONENT	INSTALLATION/PART NUMBER
C400-115	
Resistance Thermometer	60B72067-5
DC Amplifier	60B73113-63
C401-106 2	
Radiation Calorimeter	60B72065-1
DC Amplifier	60B73113-85
C402-106	
Thermocouple	60B71141-11
Zone Box	60B67608-3
DC Amplifier	60B73113-45
C403-115	
Thermocouple	60B71141-13
Zone Box	60B67608-1
DC Amplifier	60B73113-21
C404-106	
Thermocouple	60B71141-13
Zone Box	60B67608-1
DC Amplifier	60B73113-33
C61-106 2	
Radiation Calorimeter	60B72065-1
DC Amplifier	60B73113-85
C161-106	
Thermocouple	60B71141-13
Zone Box	60B67608-1
DC Amplifier	60B73113-33
C162-115	
Thermocouple	60B71141-13
Zone Box	60B67608-1
DC Amplifier	60B73113-21
E93-119 3	
Accelerometer	60B71058-1
	60B72192-11

TABLE 4.2.2.1-III

(Continued)

MEASUREMENT/COMPONENT	INSTALLATION/PART NUMBER
E92-117 Accelerometer	60B71057-1 60B72192-11
E82-115 Accelerometer	60B67121-1 60B72192-7
S117-118 DC Amplifier	4 60B73113-115
S119-118 DC Amplifier	4 60B73113-115
S121-118 DC Amplifier	4 60B73113-115
S123-118 DC Amplifier	4 60B73113-115
S125-118 DC Amplifier	4 60B73113-115
S127-118 DC Amplifier	4 60B73113-115
S129-118 DC Amplifier	4 60B73113-115
S131-118 DC Amplifier	4 60B73113-115

1 Measurements effective for first two stages only.

2 Requires GN_2 purge.

3 Requires heater blanket.

4 Presently installed back-up strain gage bridges are utilized for these measurements.

4.2.2.1 (Continued)

The present telemetry system utilizes frequencies in the 225-260 MHz band. The utilization of these frequencies by aerospace telemetry is on an interim basis, since this band is primarily allocated for military tactical communications. NASA has agreed, as documented in NMI 1052.111, to vacate the 225-260 MHz band by January 1, 1975. Aerospace telemetry will then utilize the 1435-1540 MHz or 2200-2300 MHz bands. The conversion from VHF to UHF is applicable to all aerospace telemetry and is not included in this study, since the problem is not unique to the INT-20 configuration.

9. Electrical Network Impact

(a) Main Power Distributor (115A1) 60B26411-13

The Main Power Distributor will not be changed.

(b) Sequence and Control Distributor (115A2) 60B62028-9

Approximately 23 wires will be added to and 6 wires deleted from the Sequence and Control Distributor. A 60B62100-5 Latching Relay Card Assembly will also be added.

(c) Propulsion Distributor (115A3) 60B62029-9

Approximately 19 wires will be added to and 17 wires deleted from the Propulsion Distributor.

(d) Timer Distributor (115A4) 60B62030-5

Approximately 8 wires will be added to and 9 wires deleted from the Timer Distributor.

(e) Measuring Distributor (115A7) 60B62032-9

Approximately 10 wires will be added to and 23 wires deleted from the measuring distributor. An additional 20 wires will be added for the first two stages only.

(f) Measuring Distributor (115A8) 60B62033-9

Approximately 33 wires will be added to and 47 wires deleted from the measuring distributor. An additional 8 wires will be added to the first two stages only.

4.2.2.1 (Continued)

(g) Thrust OK Distributor (115A9) 60B62295-5

Approximately 8 wires will be deleted from the Thrust OK Distributor.

(h) Cabling

The cabling design and installation will be changed to accommodate deactivation and addition of circuits and measurements and to interface with the S-IVB. The cabling change consists of stowing 29 cable branches, revising 11 cables to add wiring/connectors, lengthening 12 cable branches, and providing 4 new cables. The additional cable branches and cables are required for the first two stages only, except for cable assembly 118W16.

10. Additional Information

The identification of Electrical/Electronic Subsystems changes with INT-20 criteria is contained in Appendix A, Section 1.0. Section 2.0 of Appendix A contains technical support data, including an instrumentation program and components list, cable interconnection diagram, and electrical schematics. For a list of affected stage hardware - parts deleted, added, or revised and their applicable weights - see Appendix A, Section 3.0.

4.2.2.2 S-IC GSE/ESE

The Stage systems changes for INT-20 defined under paragraph 4.2.2.1 have the following impact on the Ground Support Equipment (GSE) and the Electrical Support Equipment (ESE):

a. Pneumatic Equipment

1. Oxidizer System Change Impact

(a) Pneumatic Console 65B23654

The GN_2 pressure drain orifice (A9936) will be replaced with one of increased flowrate.

(b) Pneumatic Checkout Rack 65B24090

A change to calibration requirements will be necessary to accommodate new calips switch setting.

(c) LOX bubbling calibration requirements will be changed.

2. Fuel System Change Impact

(a) Pneumatic Console 65B23654

The low fuel prepressurization orifice (A10113) will be replaced with one of increased flowrate.

(b) Pneumatic Console 65B23654

The GN_2 pressure drain orifice will be replaced with one of increased flowrate.

3. Auxiliary Systems Change Impact

(a) Pneumatic Console 65B23653

LOX dome and GG LOX purge orifice (A10134/A9800) will be changed and regulator calibrations requirements revised to decrease the flowrate to the stage.

(b) Engine Cocoon Thermal

Conditioning Purge calibration requirements will be changed.

b. Test and Checkout Equipment

1. Revise control room and umbilical patch distributors to enable checkout of new or revised stage systems.

4.2.2.2 (Continued)

2. Revise Ground Equipment Test Set (GETS) patch distributors to simulate new stage functions.

3. Revise 65B23959 - Pneumatic for S-IC test and checkout installation schematic to show new fuel tank pressure requirement.

4. Revise advanced electrical/mechanical schematics to reflect new GSE configuration.

5. Revise GETS schematics to reflect new stage/GSE configuration.

c. Handling Equipment

Revise 65B64037-1, "Environmental Protection - Transportation and Storage, S-IC Stage," and 65B64038-1, "Protective Cover and Plug Installation," to delete center engine system protective requirements.

4.2.3 S-IC/S-IVB Interface

The Saturn V/S-IVB aft interstage structure, as shown on Figure 4.2.3-1, is a truncated conical section 227.5 inches in length designed to mate the S-IVB and S-II stages in the normal Saturn V vehicle configuration. Hence, the forward and aft diameters of the structure are 260 inches and 396 inches, respectively, making it adaptable to mating with the S-IC stage. Means of accomplishing an S-IC/S-IVB mating are discussed in the following paragraphs.

4.2.3.1 Interface Configuration

Both the S-IVB aft interstage and the S-IC forward skirt are skin-stringer and frame type structures, the S-IVB having 144 aft interstage stringers and the S-IC having 216 forward skirt stringers. The interface areas of both structures are also similar in that they are made up of built-up rings consisting of inboard and outboard chords, webs, stiffeners and splices. The S-IVB interface ring has 132 ring stiffeners and 6 frame splices, and the S-IC interface ring has 96 ring stiffeners, 6 chord splices and 12 web splices. The S-IVB uses 288 3/8-inch diameter interface bolts on a 196.875 inch radius bolt circle and the S-IC uses 216-1/2-inch diameter bolts on a 197.17 inch radius bolt circle. These dimensional differences are illustrated on Figure 4.2.3-2. The S-IVB presently has three guide pin brackets which are not in the same circumferential locations as the three alignment pin receptacles on the S-IC stage.

The electrical disconnect panel located at the S-IVB separation plane (INT-20 Vehicle Station 1768) is approximately 18 feet forward of and displaced 90° circumferentially from the S-IC/S-II position. This requires that the cabling supplied with the S-IC stage be increased in length by approximately 36 feet. EBW firing unit mounting provisions near the S-IVB separation plane are directly forward of the corresponding S-II position. Hence, cabling supplied with these units on the S-IC stage need only be increased in length by approximately 18 feet.

Two basic schemes are proposed for accomplishing stage mating, a modified bolt hole pattern scheme for direct interface, and a scheme utilizing an adapter ring between the two stages. These schemes are illustrated in Figure 4.2.3-3, and discussed in the following paragraphs.

4.2.3.2 Modified Bolt Pattern - Direct Interface

View A-A of Figure 4.2.3-3 illustrates a modified bolt hole pattern scheme which permits a direct structural attachment of the S-IVB and S-IC stages. The interstage structure in this case is fabricated specifically for INT-20 application, having the new attach hole pattern and new alignment pin brackets. The new hole pattern cannot be used with an interstage already drilled in the standard Saturn V configuration. Otherwise the structure is typical Saturn V configuration. The attachment pattern is, however, compatible with the existing S-IC configuration, utilizing 130 of the 1/2-inch diameter bolts on the S-IC bolt circle radius. The remaining 28 bolts are 3/8-inch diameter, 18 of them in the S-IVB bolt circle radius and 10 on the

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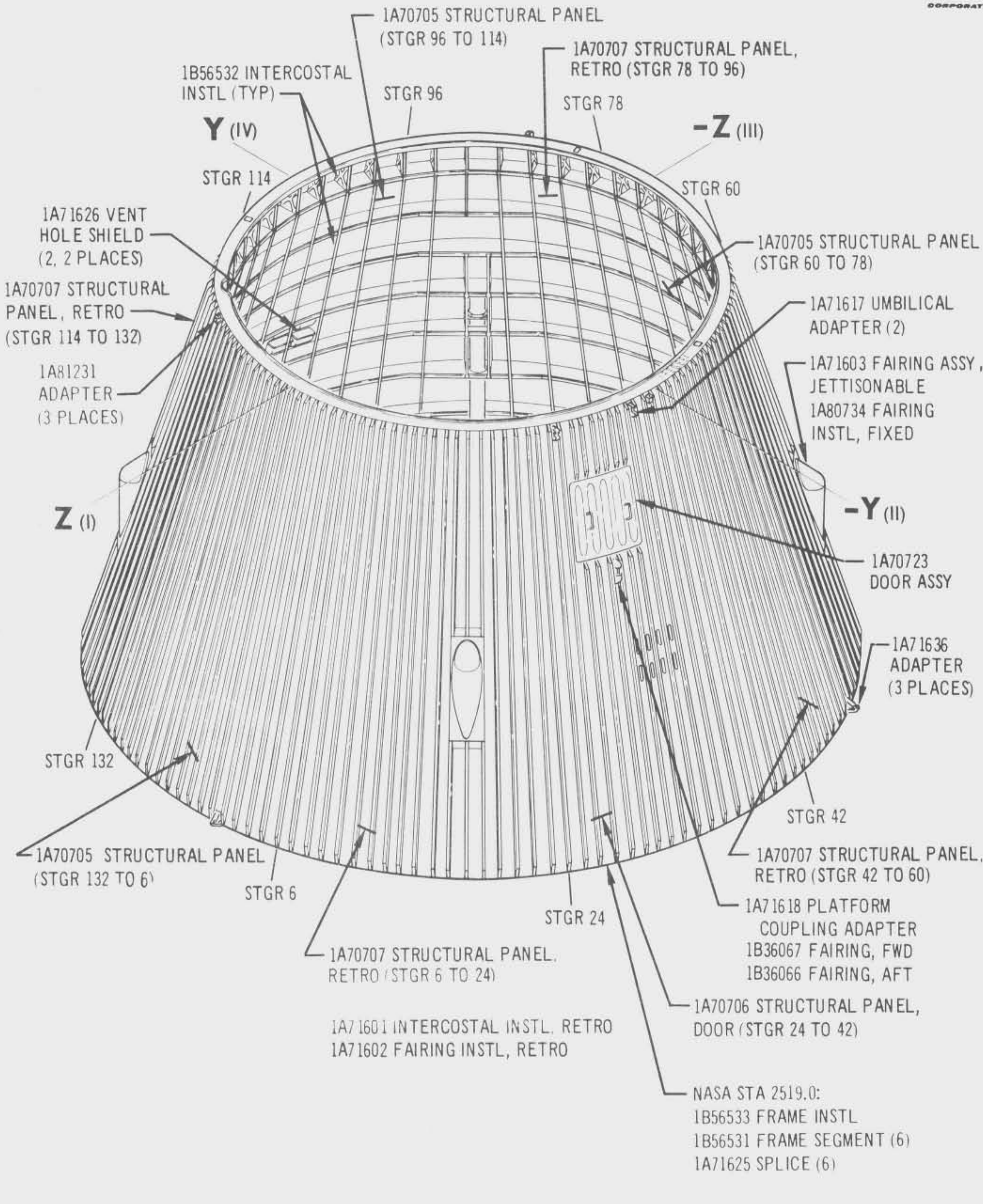

Figure 4.2.3-1. Saturn V/S-IVB Aft Interstage (1A-71604)

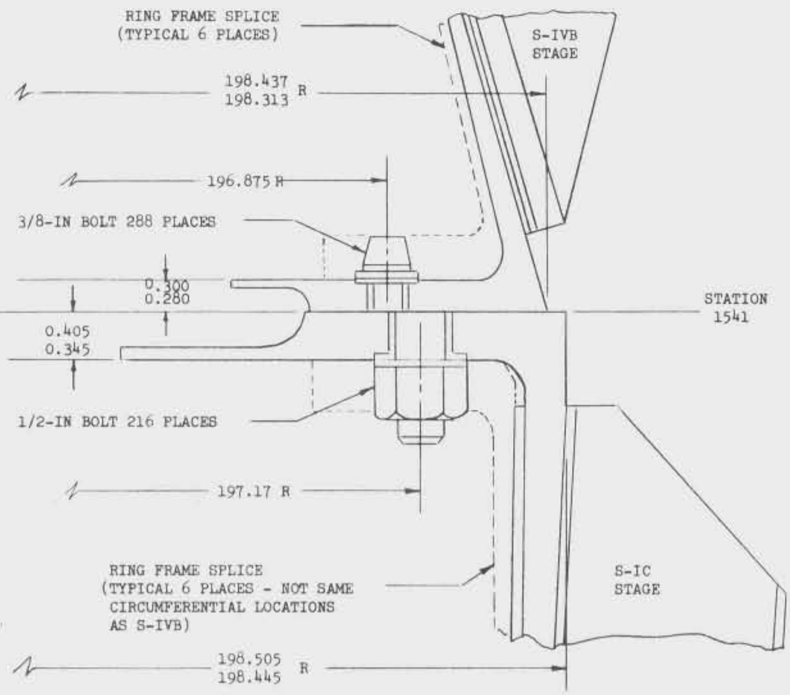

Figure 4.2.3-2. S-IC/S-IVB Interface Configuration

S-IC radius. Although 3/8-inch bolts have ample strength for INT-20 application, 1/2-inch bolts are retained at the 130 S-IC centers in order to avoid any change in S-IC stage handling fixtures. The 3/8-inch bolts are required to avoid interference with S-IVB ring stiffeners and splices.

Tests conducted on the S-IVB/S-II joint have shown that the attach bolts themselves are not critical, but that failure occurs in the S-IVB attach angle along the bolt line at an average load of 1692 lb per inch. With the S-IVB bolt spacing of 4.30 inches the load per bolt was 7300 lb, well below the allowable of 17,600 lb for the 3/8 dia. bolts used.

The maximum tension load defined for the INT-20 vehicle is only 201 lbs/inch, occuring at max $q\alpha$. Dynamic tension loads were not provided. However, although possibly conservative, the joint was checked for the same dynamic rebound loads used for the Saturn V, or 731 lb per inch ultimate. The spacing that results from skipping every third S-IC bolt yields a span loading of 8.60 inches or 6300 lb per bolt. Applying the 1.16 hard point factor used with the Saturn V joint gives a design ultimate load of 7300 lb, which essentially matches the proven capability of the S-IVB joint. It should be noted, however, that a substantial increase in joint strength will be realized by the fact that all of the highly loaded bolts are 0.295 in. closer to the S-IVB attach angle than in the tested configuration. Only the 18-3/8-inch bolts that are in the splice areas remain on the S-IVB bolt circle, and the highest bolt load in this area is 5200 lb.

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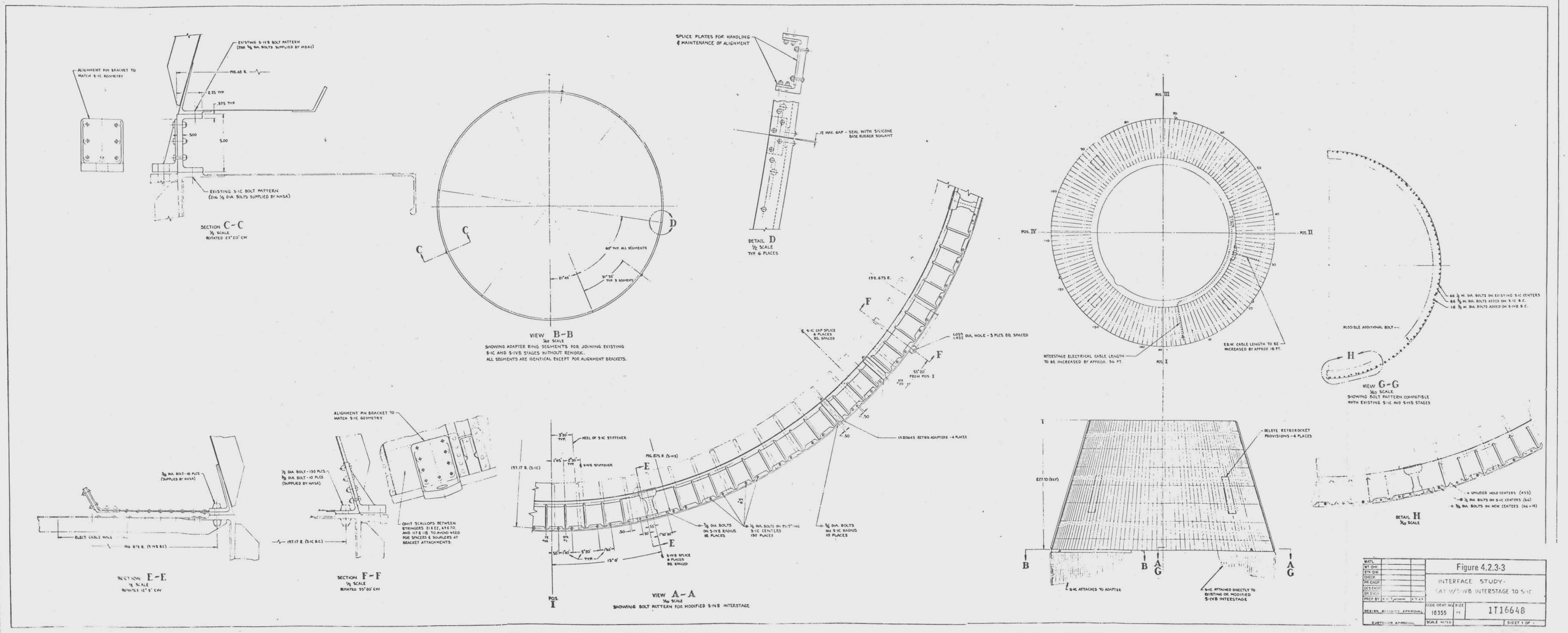

Figure 4.2.3-3
INTERFACE STUDY-
SAT V/S-IVB INTERSTAGE TO S-IC

These relative bolt positions are more clearly shown in Sections E-E and F-F of Figure 4. 2. 3-3. Also shown in Section E-E is the position of an S-IC electrical feedthrough hole relative to the S-IVB aft frame. Although the frame overlaps half of the hole, it is understood that the hole was only used for R&D versions, and that presently the electrical cables are routed around the attach frame.

4. 2. 3. 3 Adapter Ring Configuration

View B-B of Figure 4. 2. 3-3 illustrates a method of structurally mating the two stages which requires no rework of either stage, but utilizes both existing bolt hole patterns. This is accomplished by use of an adapter ring which makes the transition between the two structures. The ring has a channel cross-section approximately 5-inches deep, allowing adequate clearance for all bolt installations, including horizontal splices.

The ring will be fabricated from six equal segments, spliced together and then drilled. Removable alignment pin brackets will be bolted to the outer face at three locations. The ring then could be installed on the S-IVB aft interstage at any convenient facility, although it is envisioned to be factory installed, using master tooling to assure the most accurate positioning. The interstage-adapter ring would then be shipped as a unit.

Ground handling and transportation will be affected to only a minor degree by either of the interface schemes. The 0. 295-inch increase in bolt circle radius will require elongating the holes in the 8 hold down brackets on the transportation dolly to make it compatible with Saturn V or INT-20 interstages. With the adapter ring attached, the weather protection covers will be slightly different to allow for the additional 5-inches. This requires very minor adjustment, since the cover is more or less tailor made for each interstage at time of shipment. The adapter ring weight is not a significant factor for shipping or handling.

4. 2. 3. 4 Retrofit Scheme

Whereas view A-A of Figure 4. 2. 3-3 defined a bolt pattern compatible with a predrilled S-IC stage but required an undrilled S-IVB stage, view G-G shows a direct interface bolt arrangement suitable for the case where both stages are already drilled. This arrangement could be used for the retrofit case, i. e., retrieval of a Saturn V stage from storage for adaptation to the INT-20 configuration.

The pattern shown utilizes 66 of the existing S-IC 1/2 in. dia. holes and 6 existing S-IVB 3/8 in. dia. holes; these of course must be duplicated in the matching assembly. In addition, 78 holes in new locations are required in each stage; 12 of these must be 3/8 dia. and on the S-IVB bolt circle since they are in the splice areas, but the other 66 are on the larger S-IC bolt circle and could be either 1/2 or 3/8 dia. Since the load problem is one of flange strength, not bolt strength, 3/8 dia. bolts are used to take advantage of their lower cost and easier installation.

Spacing restrictions imposed by the presence of S-IVB holes causes an increase in span loading over that achieved in the direct interface design in 14 of the 150 bolts. For 12 of these bolts the load increase is less than 4%, which should be well within the increased capability provided by the improved bolt position, i. e., nearer the heel of the attach angle. The load increase on the other two bolts is 10%. If further analysis or test cannot verify the capability to carry this load, or reduce the assumed loads, then two additional S-IVB bolts could be installed. Although they would be undesirably close to existing S-IC holes, the loading on these bolts would only be 4200 lb and the S-IC flange can be backed up by a heavy doubler if necessary.

The alignment brackets will be the same as those used with the modified interstage as shown in Section F-F, but their attachment will require use of a tapered filler and doubler to accommodate the scallop pattern on the existing attach angle.

The simpler, and preferred, means of adapting a Saturn V stage to INT-20 configuration on a retrofit basis is through use of the adapter ring concept. In this case, no further drilling would be required in either stage; the ring would mate the two stages utilizing their existing interface bolt patterns.

4. 2. 3. 5 Interface Configuration Selection

Both of the previously described interface options for the baseline INT 20 - the direct interface conept and the adapter ring concept - are technically feasible, although the adapter ring concept would probably be less likely to pose coordination problems and would provide more program flexibility. For retro-fit purposes, only the adapter ring appears practical, as it avoids having to generate new and/or revised tooling to accomplish a re-drill of the interface rings, which would be an especially difficult task to accomplish on a completed S-IC stage with engines installed.

To assist in evaluation, tooling and cost trade investigations were made on the two configurations. As described in Section 5. 3. 3, some new tooling is required for each. For the direct interface (new bolt hole pattern) a new control master, two new transfer gages and a new drill plate would have to be made. For the adapter ring, a new stretch form die and trim fixture, and new assembly/drill jig would be required. Other minor tools would be required for the alignment brackets and splice plates. From a tooling/manufacturing standpoint the adapter ring concept is preferred, for, being an off-line operation it avoids interference with the concurrent Saturn V production by not requiring intermittent drill plate changes. Hence, a potential source of error is removed.

Further, the results of the cost-trade investigation (Section 5. 6. 3) indicate that with concurrent Saturn V production, the adapter ring total program cost is less than that for the direct interface up to a quantity of 22 vehicles. The trade point for program costs without concurrent Saturn V production is 17 units. Although a higher recurring cost per unit results for the adapter ring, the much higher development costs involved with generating a new master gage and transfer gages for the direct interface concept result in higher program costs.

4.2.3.5 (Continued)

Thus, all things considered - manufacturing approach, retrofit capability and economics - the adapter ring is the recommended S-IC/S-IVB interface concept.

4.2.3.6 S-IC/S-IVB Interface Effects on Astrionics System

a. Sequencing Subsystem Requirements

The Flight Control Computer (FCC) requires an S-IC burn mode signal replacement for one presently interlocked through the S-II stage.

b. Sequencing Subsystem Implementation

Figure 4.2.3.6-1 illustrates the modification in the S-IC stage and interstage wiring to power the S-IC burn mode and remove the S-IC burn command upon S-IC staging. The following characteristics justify the change:

1. Powering of S-IC burn mode is similar to Saturn V.

2. Removes power from S-IC burn mode command. Presently this is done by switching to S-II burn mode with K34 relay set by the Switch Selector. This same Switch Selector function on INT-20 will be used for one of the S-IC Engine Cant removal commands and S-II burn mode will be entered momentarily after S-IC cutoff and prior to staging only to break the S-IC burn mode latch internal to FCC.

3. Power removal at separation will also switch TM measurements to S-IVB mode.

4. This change has no hardware impact on IU or S-IVB.

5. This change has only a minor impact on the interstage and S-IC.

6. No impact on software beyond the nominal for mission-to-mission flight tapes.

SAT V - S-II MEAS. STAGE SEPARATION (DI-10)

INT-20 - S-IC MEAS. STAGE SEPARATION (DI-10)

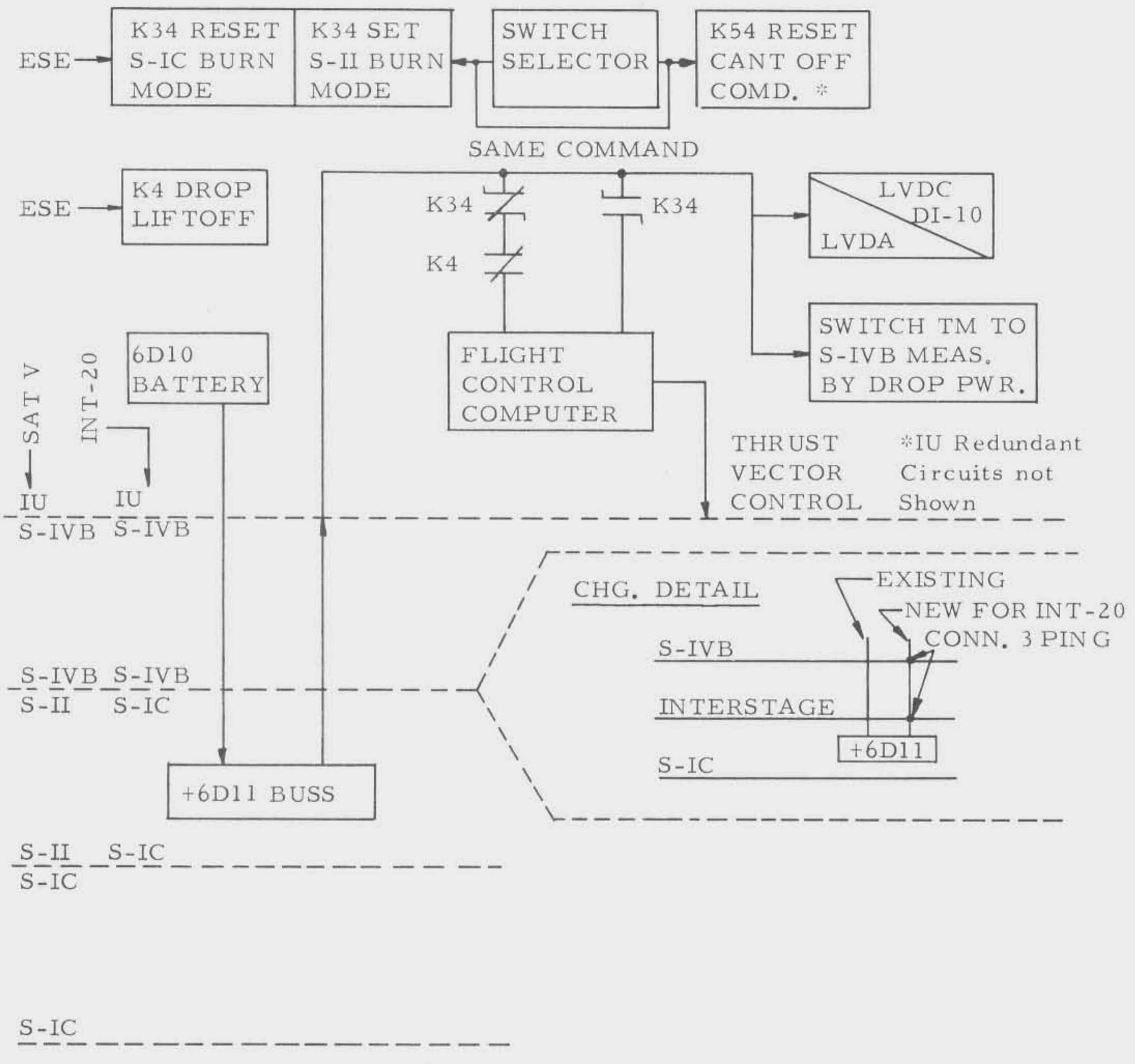

FIGURE 4.2.3.6-1 S-IC BURN MODE POWERING INTERLOCKED WITH S-IC STAGE

4.2.4 S-IVB Stage and GSE/ESE Impact

4.2.4.1 Baseline Stage Configuration

The S-IVB stage configuration recommended for INT-20 vehicle use is the Saturn V version, as shown in Figure 4.2.4-1 (reference Section 3.1.2). This version was designed and built to perform a two-burn mission on the standard Saturn V LOR vehicle. First burn inserts the partially loaded stage and its payload into low Earth orbit, for a coast period of up to 4-1/2 hours. The S-IVB is then re-ignited to insert the payload on a translunar trajectory, and following burnout, provides up to two hours of attitude control. The Saturn IB/S-IVB stage, on the other hand, performs in only a single burn mission to low Earth orbit. Hence, in addition to greater structural load carrying capability, the Saturn V/S-IVB stage possesses a number of additional systems associated with the engine restart and increased coast periods.

Since the baseline mission for the INT-20 vehicle requires only a single burn of the S-IVB stage into orbit, restart capability is not required, and a Saturn IB type of propulsion system would suffice. The baseline INT-20/S-IVB stage will be derived by accomplishing certain in-line changes to, or deletions of, Saturn V stage unused systems. These changes will eliminate potential problems or provide operational simplicity, and improve reliability. The deletions will also result in reduced stage recurring cost, but will not be so extensive as to preclude the relatively simple addition of restart capability if required for future alternate missions. The alternate INT-20/S-IVB stage configuration, however, discussed subsequently in Section 4.3.3, is derived through more extensive changes and/or deletions, and does not retain the flexibility for simple addition of restart capability.

For the baseline INT-20/S-IVB stage, the following Saturn V stage systems will be changed.

a. Repressurization System. The ambient repressurization system, which is a backup system on the Saturn V stage, is deleted in its entirety. The primary system, utilizing an O_2H_2 burner in conjunction with three cold helium bottles, remains as is except that the burner is not installed and the lines to the burner are capped off.

b. Auxiliary Propulsion System. The two APS ullaging engines (one per unit) are not installed, propellant lines are capped, and electrical connectors coiled and stowed.

c. LH_2 Continuous Vent System. The continuous vent system in the forward skirt is disconnected by removing a bellows assembly and capping off the open ports.

d. Plume Impingement Curtain. The retrorocket plume impingement curtain installation in the aft skirt area is deleted, since S-IC stage rather than S-IVB stage retrorockets will be used for stage separation.

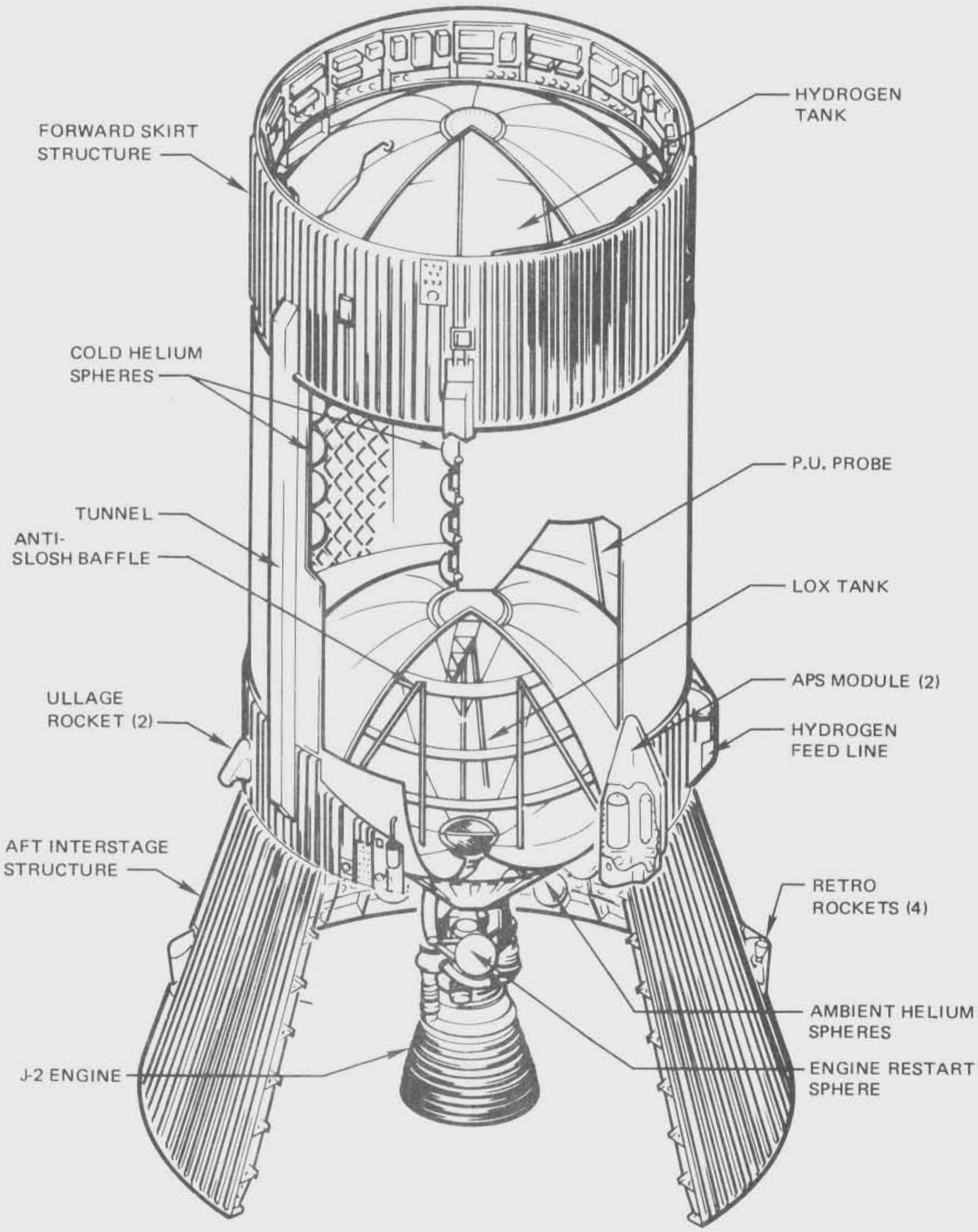

Figure 4.2.4-1. Saturn V/S-IVB Stage Configuration

e. Thermo-conditioning Duct. The thermo-conditioning duct, which maintains the pneumatic bottle (No. 6) temperature, is capped off.

f. Other changes involve relocation of taps on the PU system bridge ratio transformer to accomplish the mission-required propellant loading and off-loading of the APS system to reflect attitude control requirements. Appropriate instrumentation in conjunction with the removed systems is also deleted.

4.2.4.2 Baseline Interstage Configuration

The standard Saturn V aft interstage structure, previously described in Section 4.2.3 (see Figure 4.2.3-1), is built up from eight structural panels. Four of these panels, located 90° apart from each other, contain the necessary fittings, intercostals and fairings for housing the retro-rockets normally employed to effect S-IVB/S-II stage separation. Of the other four panels, three are plain and one contains an access door.

Since S-IC retrorockets will be used for S-IVB/S-IC stage separation, the standard S-IVB retrorockets may be deleted. Two options are available to accomplish this. In the first case, the retrorockets and their attendant ordnance system would merely not be installed, leaving all provisions as is. In the second case, all provisions for the rockets would be deleted during production by constructing the interstage from seven plain panels and the one door panel. In either case, the costs of the retro-rockets themselves would be saved.

A cost trade investigation was made on the two options, as reported in Section 5.6.3. It was determined that the non-recurring costs involved in effecting either change would be somewhat minimal, and practically the same. Neglecting the cost savings for the retrorockets, recurring program costs for the first case (merely not installing the rockets) would be constant, i.e., no change. In the second case, however, a cost saving per unit would result due to the reduced number of parts and fabrication time involved in manufacturing a plain interstage. Thus it was recommended that all retrorocket provisions be deleted on INT-20 interstages.

4.2.4.3 Propulsion System

The S-IVB stage employs a single J-2 engine, gimballed to provide pitch and yaw control during powered flight. Liquid oxygen and liquid hydrogen propellants are burned at a nominal 5:1 (oxidizer: fuel) weight mixture ratio to provide a nominal vacuum thrust of 205,000 lb. The engine provides a specific impulse of 426 seconds, and has a 27.5:1 nozzle area ratio. In the normal Saturn V/S-IVB stage configuration, the engine is restartable (one restart). The single burn requirement of the INT-20 configuration will not require any modifications to the engine itself. It is proposed, however, that the stage pneumatic sphere be connected to the J-2 control helium sphere. This simple modification would provide more pneumatics for engine burn as well as any safing operation which may be required.

The suggested deletions/modifications to the Saturn V/S-IVB stage propulsion system place the INT-20/S-IVB baseline stage in satisfactory condition for a single burn mission. The deletion of the continuous vent system, ambient repressurization system and O_2H_2 burner represent a significant reduction in hardware. These deletions are noted schematically in Figure 4.2.4-2. The operation of the remaining stage systems is briefly discussed in the following paragraphs.

a. LH_2 Pressurization System

The LH_2 tank is prepressurized with ground helium and subsequent to liftoff the LH_2 tank ullage pressure remains at the tank relief level. This results in satisfactory ullage pressure conditions being present for engine start. During stage burn, the LH_2 tank ullage is pressurized by a tap-off from the J-2 LH_2 injector through a pressurization control module. The ullage pressure is therefore controlled by ullage pressure sensing switch/pressurization control module interaction. The sizing of the pressurization control orifice is such that no pressurization cycles are expected. Over control capability does exist should the ullage pressure decrease to 28 psia. Step pressurization is effected approximately 300 seconds into the S-IVB burn, whereby the normal flow orifice and a secondary orifice are open which assures a maximum ullage pressure level. This system is unaffected by the baseline stage modifications.

b. LOX Pressurization System

The LOX tank is prepressurized with ground helium and subsequent to liftoff the ullage pressure normally decreases very slightly. This pressure decrease is not significant and very satisfactory engine start conditions are therefore present.

Helium stored in spheres mounted in the LH_2 tank supply the pressurant for the LOX tank. In the interest of efficiency, the pressurant helium is increased in internal energy by passing some pressurant through the J-2 heat exchanger. Controlled mixing is then employed to produce the desired ullage pressurant energy level. Nine spheres are mounted in the LH_2 tank to provide this LOX tank pressurant. Since a single burn mission requires only 6 spheres, the pressurant supply is adequate, and the pressurization capability is not affected by the baseline modifications.

c. Tank Venting Systems

The LOX and LH_2 tanks employ vent and relief valves, relief valves, and nonpropulsive vent (NPV) systems. The maximum pressure which a tank can attain is controlled by its vent and relief valve. Redundancy is afforded by a parallel relief valve with both valves venting into the NPV ducting. The isolation of the propulsive (continuous) vent system does not affect the tank venting capability.

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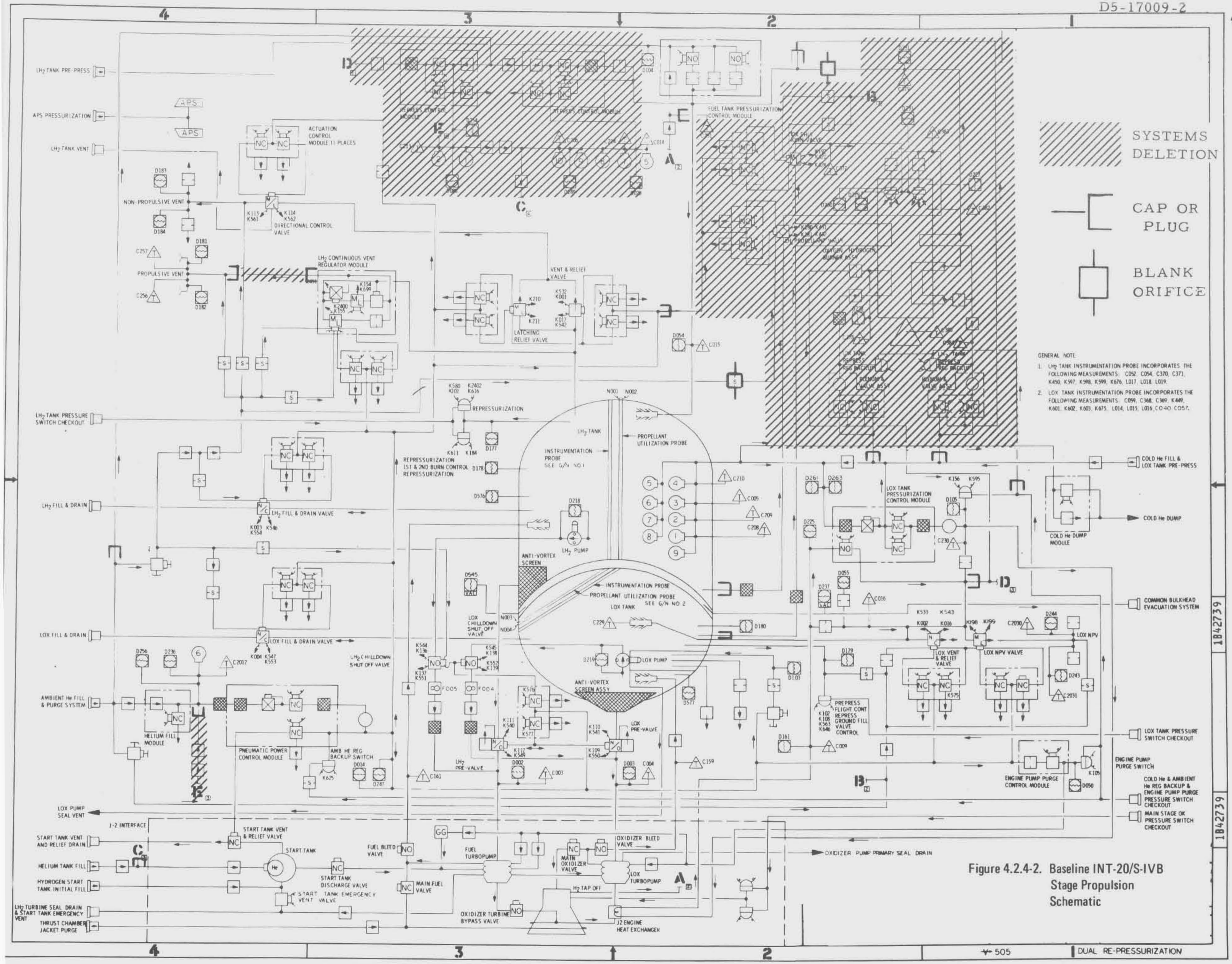

Figure 4.2.4-2. Baseline INT-20/S-IVB Stage Propulsion Schematic

d. Feed Duct and Engine Preconditioning

The provision of the required NPSH is directly influenced, up to engine start, by the operation of the LOX and LH_2 recirculation chilldown systems. These systems force the respective sub-cooled propellants through the main feed duct, J-2 turbomachinery and then back to the respective propellant tanks. These systems are not affected by the modifications necessary to attain the baseline configuration.

e. Pneumatic Control System

Helium provides a pressure to operate all S-IVB stage pneumatically-operated values. Helium is supplied from a sphere precharged at 3,100 ± 100 psia. The pneumatic control module filters and regulates the helium pressure to 495 ± 25 psia for use in the actuation control modules. Stage pneumatics are unaffected by the modifications required by the baseline configuration.

f. Auxiliary Propulsion System (APS)

The standard Saturn V/S-IVB APS component arrangement is pictured on Figure 4.2.4-3. The attitude control engines of the APS provide three axes control for the vehicle during the coast phase and roll control during stage burn. The ullage engines, normally employed to provide propellant settling during the restart sequence, are deleted for INT-20 application, schematically illustrated on Figure 4.2.4-4. The elimination of these engines does not affect the performance capability of the remaining APS engines.

As a result of attitude control, maneuvering and ullage requirements during orbital coast and restart, the Saturn V/S-IVB APS normally carries approximately 500 pounds more propellant than a Saturn IB/S-IVB APS. These additional propellants are not required on a Saturn IB type, single burn mission; hence, INT-20/S-IVB APS units could be off-loaded approximately 80%. No redesign or modification would be required as a result of the off-loading. It would be accomplished by revising the appropriate APS loading procedures.

4.2.4.4 Electrical System

Due to basic mission similarities, the INT-20/S-IVB stage electrical requirements are very nearly the same as for a Saturn IB/S-IVB stage. Thus, system modifications required for INT-20 will be generally limited to the differences between a Saturn V/ and Saturn IB/S-IVB stage. These modifications will consist of propellant utilization (PU) system mixture ratio changes, coiling and stowing of unused wire harnesses and minor interface revisions.

As a result of differences in mission profile, Saturn V and Saturn IB propellant loading and propellant utilization requirements differ significantly. For Saturn V missions, the fuel tank is loaded as full as is

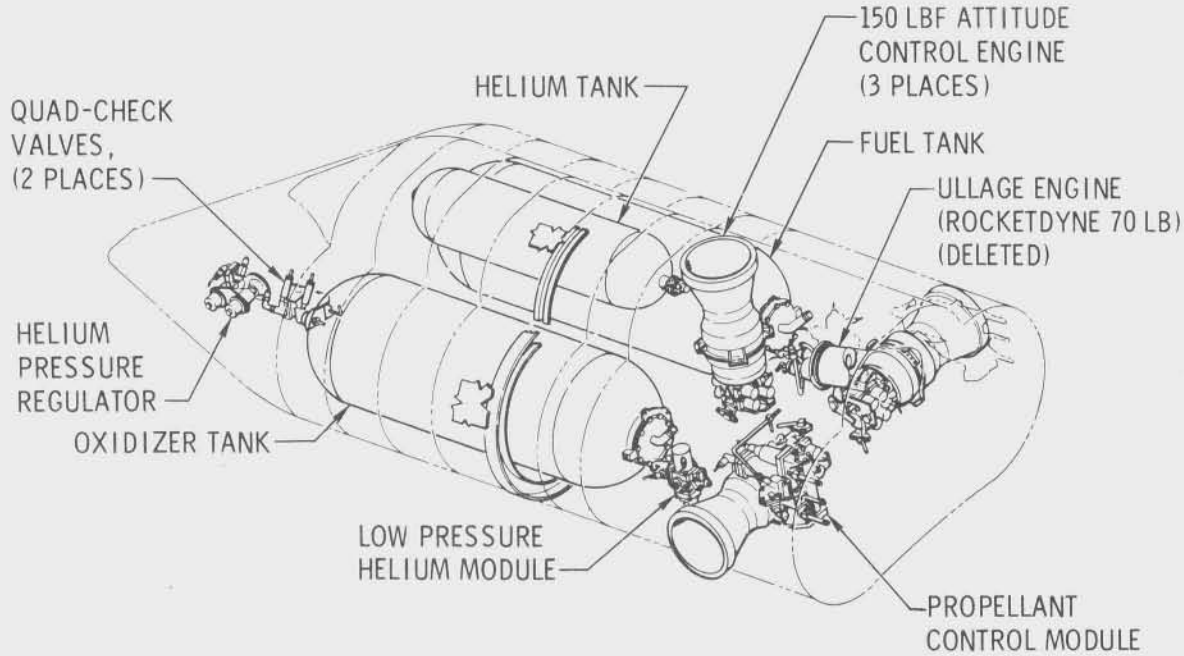

Figure 4.2.4-3. APS Component Installation

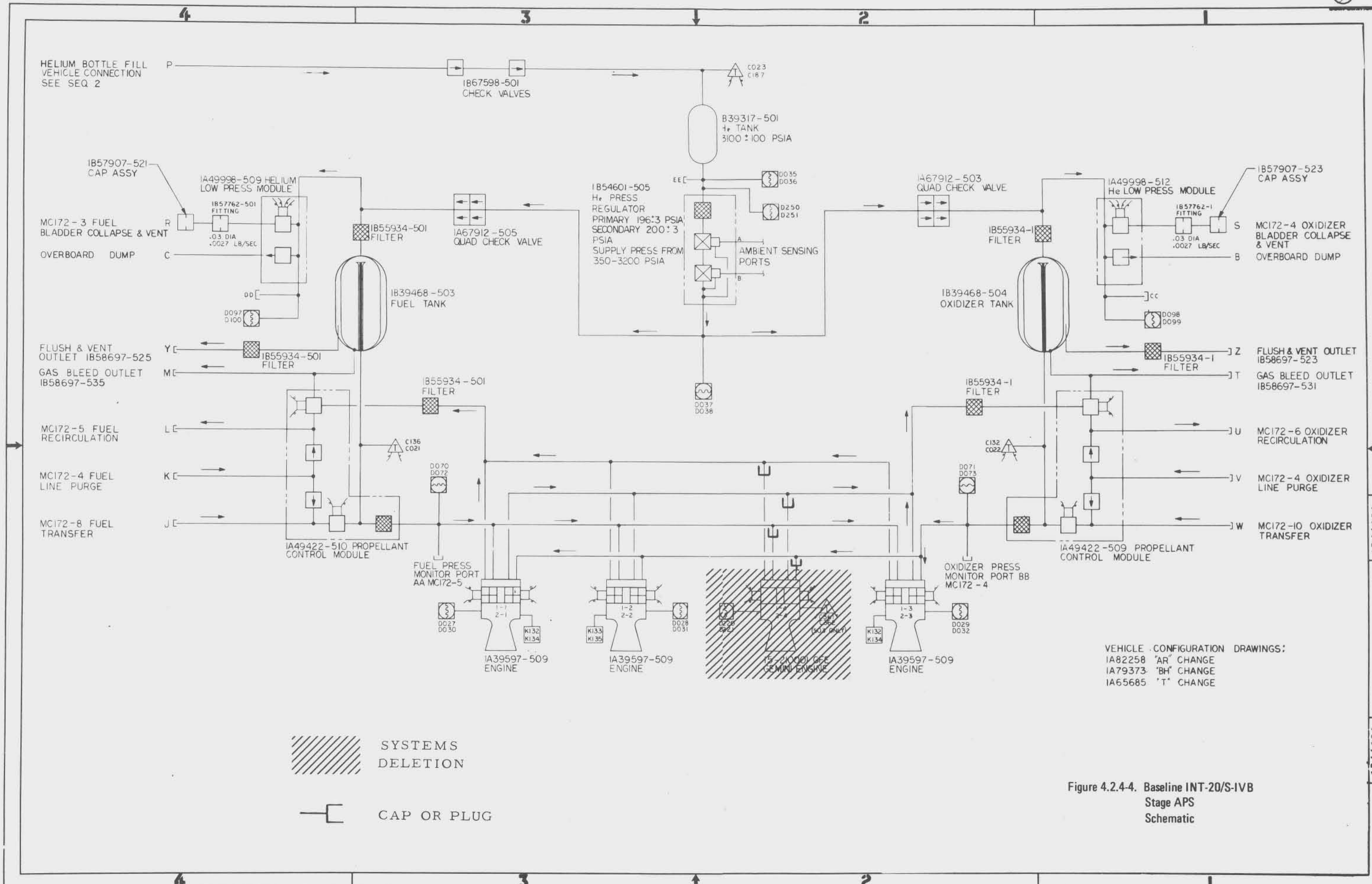

Figure 4.2.4-4. Baseline INT-20/S-IVB Stage APS Schematic

practical, the PU system is biased for orbital boiloff prior to restart, the reference mixture ratio (RMR) must be changed for first and second burn, and a low engine mixture ratio must be maintained during second start. For Saturn IB type missions, the fuel tank is off-loaded to achieve optimum payload for low Earth orbit and a single RMR and bias are maintained. Similarily, for INT-20 application, bias and mixture ratio changes will not be required. These differences will be resolved by relocating taps on the PU electronics assembly bridge ratio transformer. This modification has been successfully performed in the past.

A number of wire harnesses will have branches coiled and stowed or will be removed entirely to be compatible with system deletions. Generally, where the two-burn flexibility is to be maintained, the affected connectors are coiled and stowed rather than removed. The exception to this is for the ambient repressurization system, which is permanently deleted. Thus, 3 wire harnesses are reworked with the resulting deletion of 31 wires. For the remainder of the baseline INT-20/S-IVB electrical system a total of 32 branches of 9 wire harnesses will be coiled and stowed, and 12 wire harnesses will be deleted, as itemized on Table 4.2.4-I. These harnesses provided power and control for the O_2H_2 burner, APS ullage engines, continuous vent system and ambient repressurization system. Included also is the instrumentation for these systems. Table 4.2.4-II provides a list of the Saturn V/S-IVB stage telemetry measurements deleted.

The Instrument Unit (IU)/S-IVB interface will be compatible without modification for the INT-20 vehicle. The S-IVB/S-IC interface will differ from the S-IVB/S-II interface by the addition of one wire in the interstage. This wire will provide a talkback to remove power from the S-IC burn mode command in the LVDC, and switch TM measurements to the S-IVB mode.

Table 4.2.4-I

SAT-V/S-IVB WIRE HARNESS REVISIONS FOR INT-20

Delete ambient repressurization system	Rework 3 wire harnesses total 31 wires removed
Delete O_2H_2 burner	Coil and Stow 17 connectors total in 3 wire harnesses
Delete APS ullage engines	Coil and Stow 2 connectors in 1 wire harness.
Disconnect continuous vent system	Coil and Stow 4 connectors in 1 wire harness
Delete Instrumentation for above	Coil and Stow 9 connectors total in 4 wire harnesses, Remove 8 wire harnesses.
Delete 4 retrorockets	Remove 4 wire harnesses.

Table 4.2.4-II (page 1 of 2)

SAT-V/S-IVB TELEMETRY MEASUREMENTS DELETED FOR INT-20

C014-403	Temp	- He Repress Sphere No. 5 gas
C206-403		He Repress Sphere No. 10 gas
C214-403		He Repress Sphere No. 7 gas
C256-409		Fuel Tank Continuous Vent 1
C257-409		Fuel Tank Continuous Vent 2
C378-403		O_2/H_2 Burner LOX Tank Press Coil Outlet
C379-403		O_2/H_2 Burner LH_2 Tank Press Coil Outlet
C382-403		O_2/H_2 Burner Chamber Dome
C2034-403		O_2/H_2 Burner Dome No. 2
D020-403	Press	- Fuel Tank He Bottle Repress
D088-403		LOX Tank Repress Spheres
D181-409		Fuel Tank Continuous Vent 1
D182-409		Fuel Tank Continuous Vent 2
D220-414		Ullage Control Chamber No. 1-4
D221-415		Ullage Control Chamber No. 2-4
D227-403		O_2/H_2 Burner Chamber Dome
D228-403		O_2/H_2 Burner LOX Tank Press Coil Outlet
D231-403		O_2/H_2 Burner LH_2 Tank Press Coil Outlet
D249-403		Fuel Tank Bottle Repress Backup Meas
D254-403		LOX Tank Repress Sphere Backup Meas
K154-411	Event	- Relief Over-Ride Shut-off Vlv, Cont Vent LH_2 Cl
K155-411		Orf. SOV Cont Vt LH_2 Tk-Cl
K180-404		He Heater LH_2 Vlv Full-Cl
K181-404		He Heater LH_2 Vlv Full-Op
K182-404		He Heater LOX Vlv Full Cl
K183-404		He Heater LOX Vlv Full Op
K192-403		O_2/H_2 Burner LOX Man Shutdown Valve Ind Open
K195-404		Repress Sys. Ambient Mode
K427		O_2/H_2 LOX Shutdown Open
K428		O_2/H_2 LOX Shutdown Closed

Table 4.2.4-II (page 2 of 2)

K431		O_2/H_2 Prop Vlv Open
K432		O_2/H_2 Prop Vlv Closed
K699		LH_2 Tk V & R Vlv Closed
K2400		LH_2 Vent Ori Bypass Closed
M010-411	Volt -	Fuel Boiloff Bias Signal
M073-404		Helium Heater Spark Exciter No. 2
M074-404		Helium Heater Spark Exciter No. 1

The Saturn V/S-IVB requires many command functions from the IU to provide inflight command capability. Many will not be required for the INT-20 vehicle. There is a total of 30 of these commands which are not required as itemized in Table 4.2.4-III, which may be deleted by software changes in the IU.

Table 4.2.4-III (page 1 of 2)

SAT-V/S-IVB SWITCH SELECTOR COMMANDS--SPARE ON INT-20

Channel 3	LOX Tank Repress Control Valve Enable - On
4	LOX Tank Repress Control Valve Enable - Off
17	PU Valve Hardover Position-On
18	PU Valve Hardover Position - Off
26	O_2/H_2 Burner Fuel Valve and LOX Shutdown Valve Open Pilot Valve - On
32	2nd Burn Command - On
33	2nd Burn Command - Off
34	PU Fuel Boiloff Bias-On
35	PU Fuel Boiloff Bias-Off
36	Stage Repress System Mode Selector (ambient)
37	Stage Repress System Mode Selector (Cryogenic)
39	LH_2 Repress Control Valve-On
42	70 pound Ullage Eng Comm No. 1-On
43	70 pound Ullage Eng Comm No. 1-Off
60	O_2/H_2 Burner Fuel Valve and LOX Shutdown Valve Close Pilot Valve-On
61	O_2/H_2 Burner Fuel Valve and LOX Shutdown Valve Close Pilot Valve-Off

Table 4.2.4-III (page 2 of 2)

Channel 72	O_2/H_2 Burner Fuel Valve and LOX Shutdown Valve Open Pilot Valve-Off
74	O_2/H_2 Burner LOX Propellant Valve Close-On
75	O_2/H_2 Burner LOX Propellant Valve Close-Off
81	LH_2 Repress Control Valve-Off
84	LH_2 Tank Continuous Vent Valve Close-On
85	Voting Circuit Enable-On
86	Voting Circuit Enable-Off
87	LH_2 Tank Continuous Vent Valve Close-Off
89	O_2/H_2 Burner LOX Propellant Valve Pilot Valve-On
90	O_2/H_2 Burner LOX Propellant Valve Pilot Valve-Off
101	70 pound Ullage Engine Comm No. 2-On
102	70 pound Ullage Engine Comm No. 2-Off
111	LH_2 Tank Continuous Vent Valve Open-On
112	LH_2 Tank Continuous Vent Valve Open-Off

4.2.4.5 Ordnance System

Two S-IVB stage ordnance systems are affected for INT-20 vehicle use, the retrorocket ignition system and the separation system. Since S-IC stage retrorockets will be employed for stage separation, the S-IVB retrorockets and retrorocket ignition system will be deleted.

Mounting provisions for EBW firing units for S-IVB stage separation are located just aft of the separation joint (INT-20 vehicle station 1768) in the forward end of the interstage. For typical Saturn V application, the units for this installation are furnished with the S-II stage, mated electrically to the S-II with ample harness length for launch configuration installation. Similarily, for INT-20 vehicle application, firing units will be furnished with the S-IC stage, mated electrically and stowed forward on the stage for installation in the S-IVB. An additional eighteen feet of electrical input cable (over that normally required for S-IC/S-II installation) will be needed for the INT-20. The firing units furnished with the S-IC stage have 48-in. output cables. S-IVB mounting provisions are designed for 30-in. cables; thus, additional clamps will be required to accommodate the slack.

The remaining S-IVB stage ordnance systems--the propellant dispersion system, the ullage rocket ignition system and the ullage rocket jettison system - remain unchanged for INT-20 vehicle use.

4.2.4.6 Control System

The standard Saturn V/S-IVB flight control system requires no significant modification for INT-20 application. Pitch and yaw control during powered flight will be provided by J-2 engine gimballing. Powered flight roll control and coast-attitude control (if required) would be provided by the APS units. Standard stage separation methods will be employed, and the separation transient investigation (Section 4.1.4.4) indicated the control system could handle the expected transients.

4.2.4.7 Environmental Control Systems

The various Saturn V/S-IVB stage environmental control systems, both for ground hold and in-flight, require no changes for the INT-20 configuration.

In the forward skirt-IU compartment area, ground hold environmental control consists of a gas purge (air and nitrogen) to minimize the possibility of an explosive gas atmosphere. The purge gas is supplied from a purge duct mounted within the IU. Temperature control of electronic equipment is obtained not only by controlling purge gas temperature, but by circulating heat-transfer fluid (methanol/water/corrosion inhibitor) through the cold plates on which equipment is mounted. These cold plates are connected in parallel with the IU cold plates, and both receive their fluid supply from the IU. During ground hold, a GSE operated heat exchanger in the IU maintains fluid temperature, while during flight, the fluid is pumped in a closed loop through an ice sublimator for maintenance of temperature. The cold plates are also covered on both in-board and out-board faces by a low emissivity, aluminized mylar radiation shield to reduce heat loss to the LH_2 tank forward dome.

In the aft skirt-aft interstage compartment, purge gas during ground hold is distributed by a purge manifold circling the S-IVB aft dome near the aft skirt attach flange. The gaseous nitrogen not only eliminates explosive gas mixture, but with proper temperature control, thermally conditions such equipment as the electronics, auxiliary propulsion system, ambient helium bottle and hydraulic accumulator-reservoir. In-flight environmental control is by passive methods, i.e., specifying the proper surface finish and/or insulation of the panel-mounted electronic equipment in the area.

4.2.4.8 Stage Analyses

Various analyses were performed in order to provide data for proper evaluation of the S-IVB stage adequacy/suitability in the INT-20 configuration. Subsequent paragraphs report on these investigations.

a. Thermodynamics

A thermodynamic analysis was performed to determine the S-IVB stage structural temperatures resulting from the boost through the atmosphere. The analytical techniques and assumptions used for the investigation were identical for those used in Saturn V/S-IVB stage aero/thermodynamic analyses. The INT-20/S-IVB analysis was based on the convective heating environment data, i.e., film coefficients and recovery temperatures, as presented in Section 4.1.3.2.

The S-IVB stage structure investigated included the forward skirt, aft skirt, aft interstage and the LH_2 tank sidewall. Three conditions were considered for each of the skirt/interstage structures: (1) uninsulated structure; (2) structure insulated with 0.01-inches of Korotherm; and (3) structure insulated with 0.02-inches of Korotherm. Further, to provide data necessary to evaluate protuberance heating effects, each of the above conditions were analyzed for heating factors (h/ho) of 1.0 (undisturbed flow), 1.5 and 2.0.

The skin-stringer configurations used in the analysis are shown on Figure 4.2.4-5. Three stringer locations were considered, as shown. Since the temperature histories for these three locations were for any given condition quite similar, only one temperature curve (the most critical) is shown for the stringer on the structural heating curves.

The results of the analysis are presented on Figures 4.2.4-6 through 4.2.4-14, which give temperature histories for the forward skirt, aft skirt and aft interstage. No separate curves is shown for the LH_2 tank side-wall; the temperature for that structure was determined to be

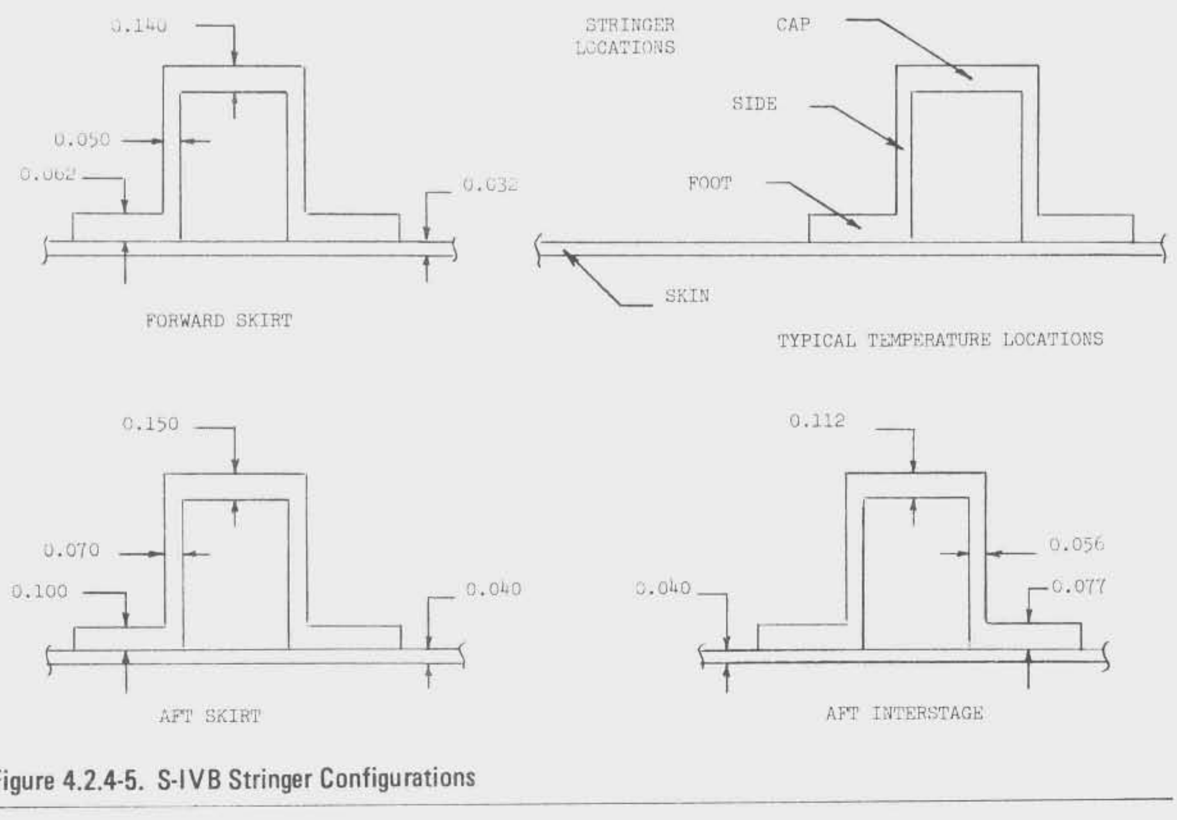

Figure 4.2.4-5. S-IVB Stringer Configurations

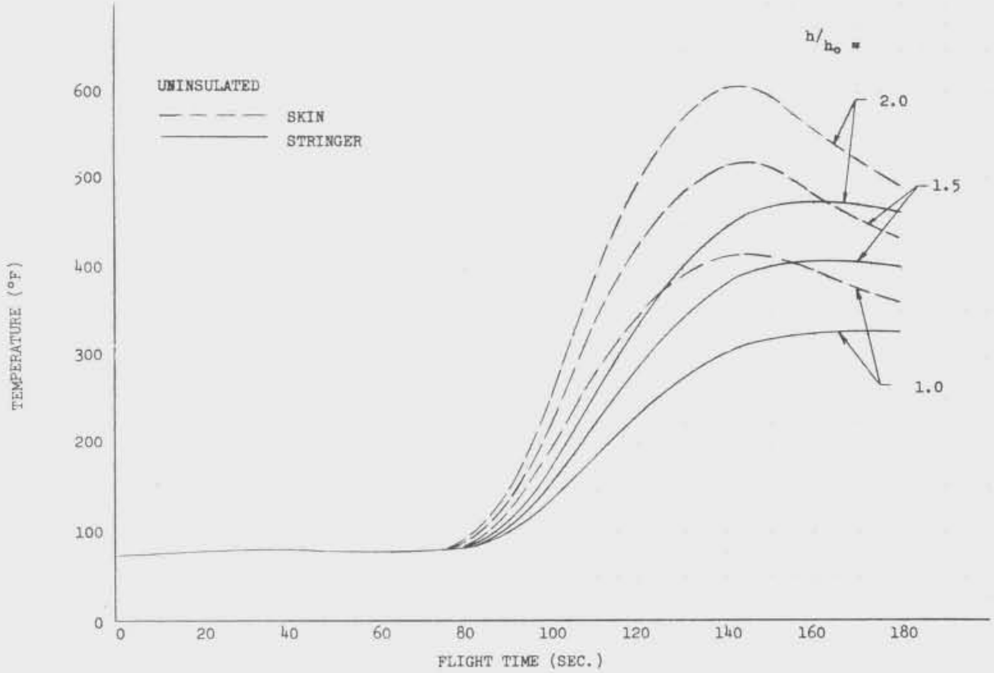

Figure 4.2.4-6. S-IVB Forward Skirt Temperature History

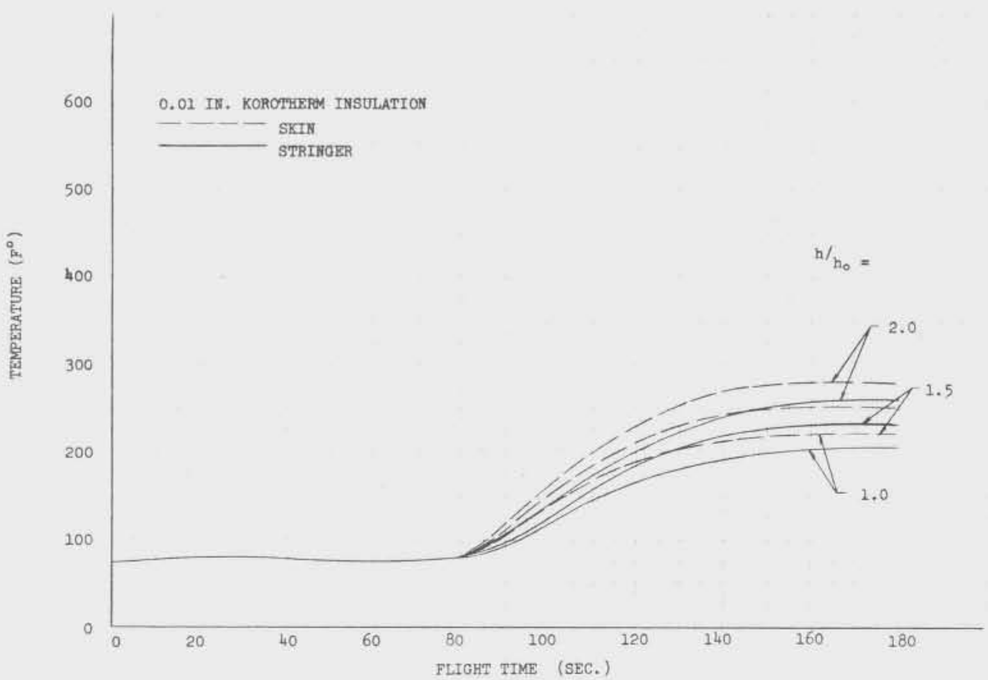

Figure 4.2.4-7. S-IVB Forward Skirt Temperature History

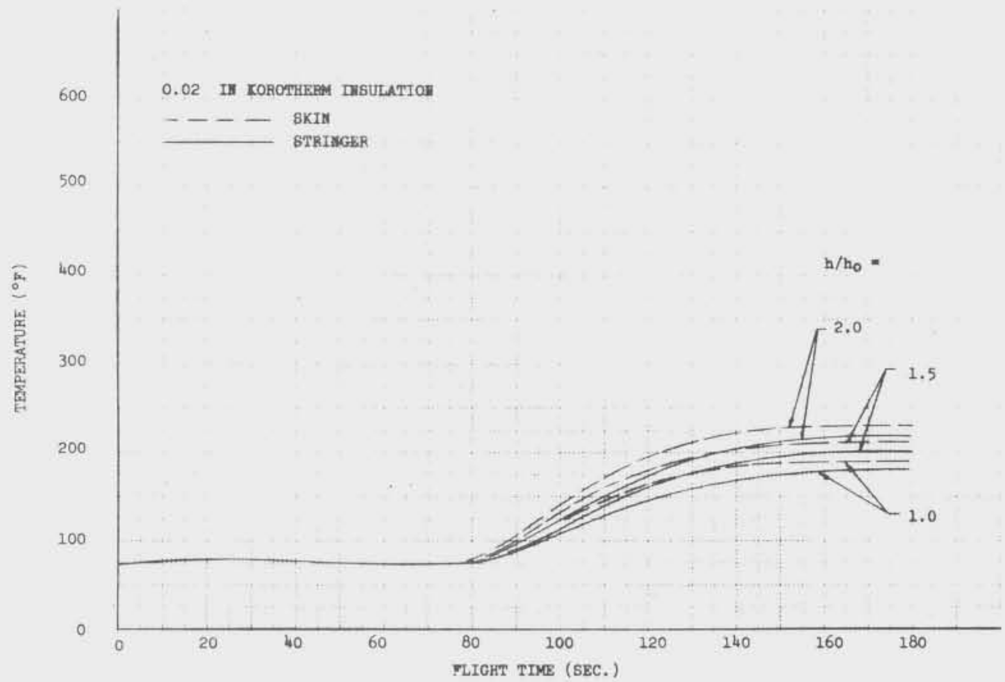

Figure 4.2.4-8. S-IVB Forward Skirt Temperature History

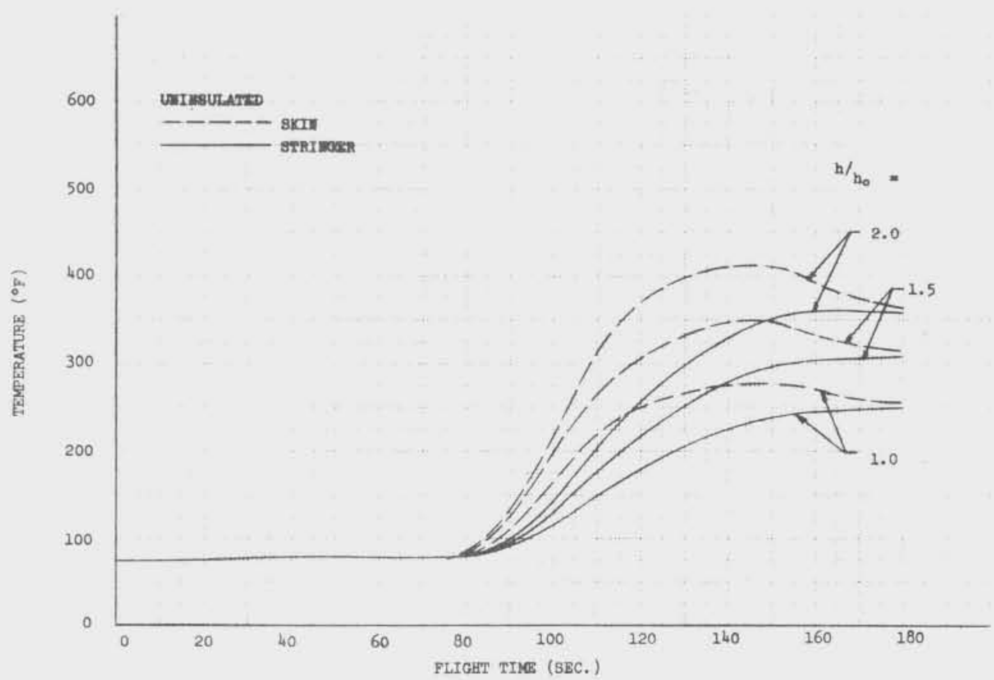

Figure 4.2.4-9. S-IVB Aft Skirt Temperature History

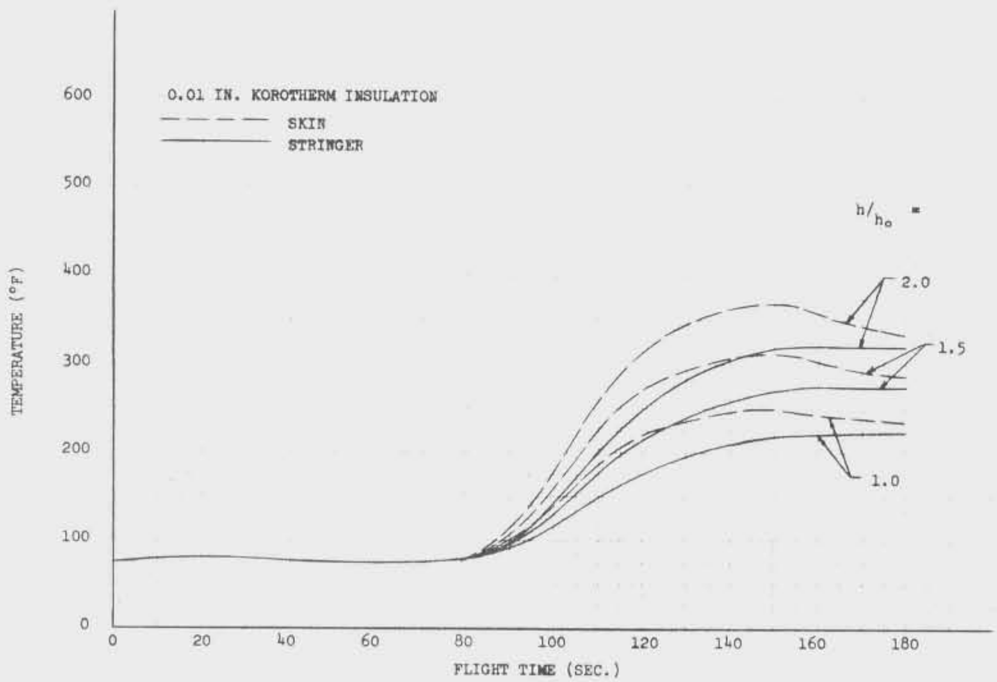

Figure 4.2.4-10. S-IVB Aft Skirt Temperature History

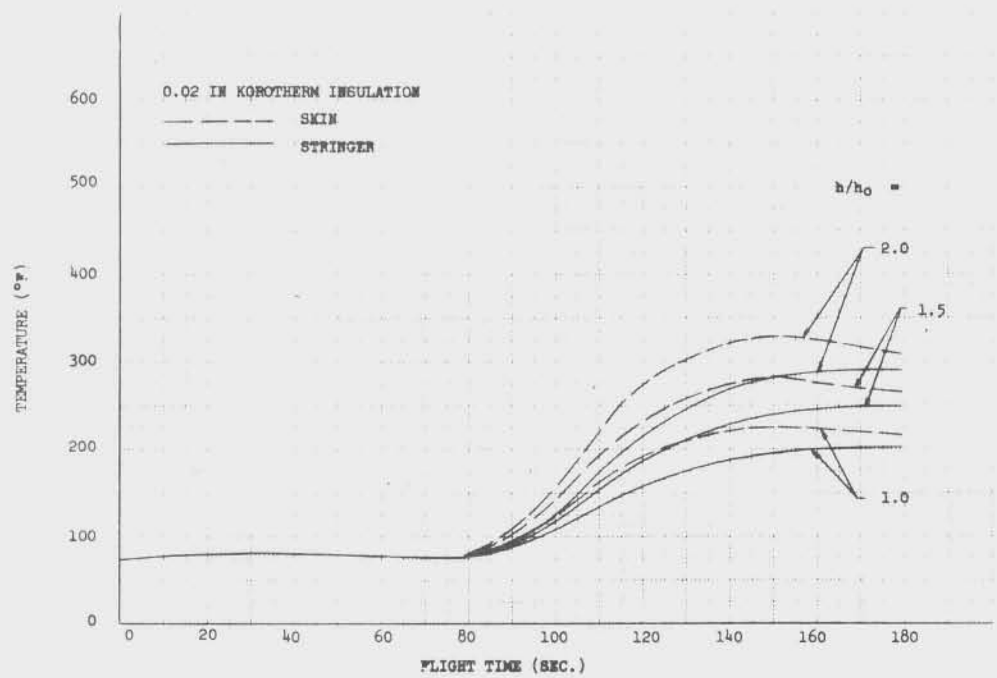

Figure 4.2.4-11. S-IVB Aft Skirt Temperature History

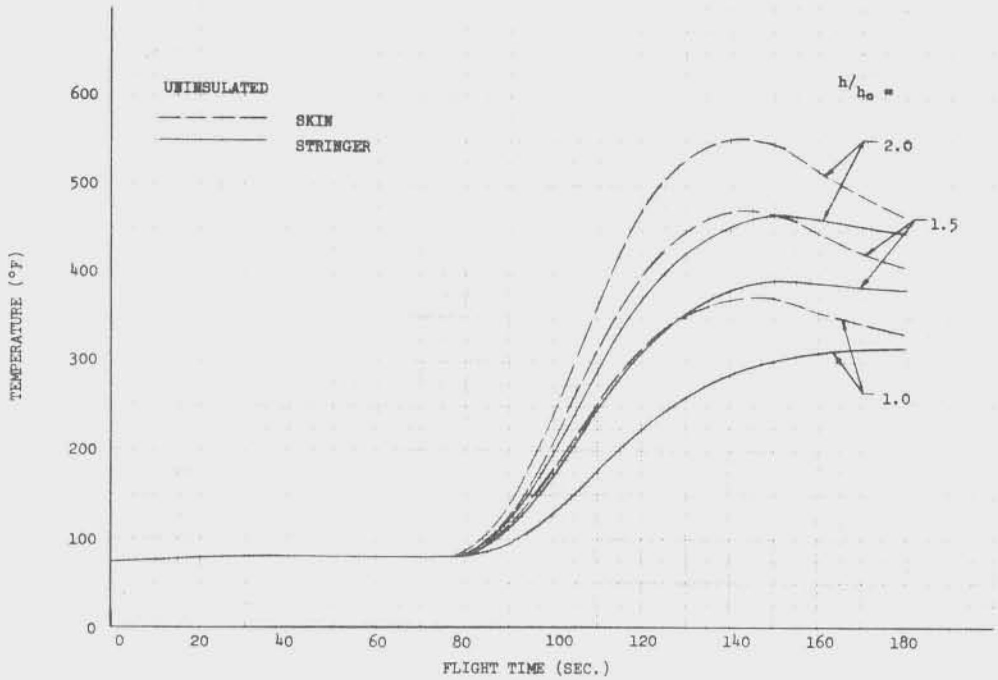

Figure 4.2.4-12. S-IVB Aft Interstage Temperature History

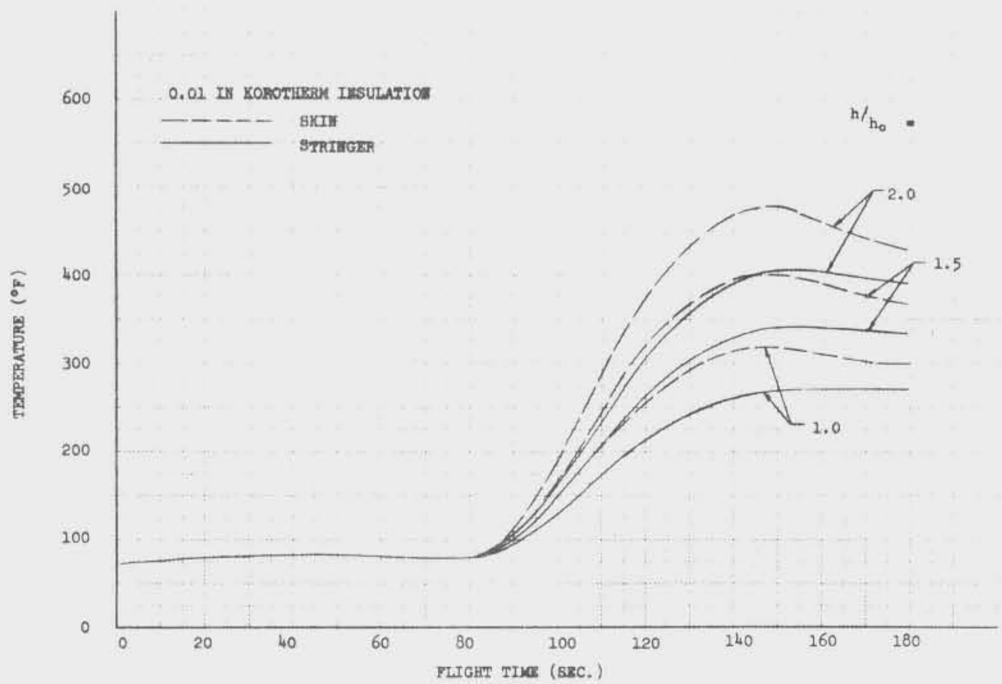

Figure 4.2.4-13. S-IVB Aft Interstage Temperature History

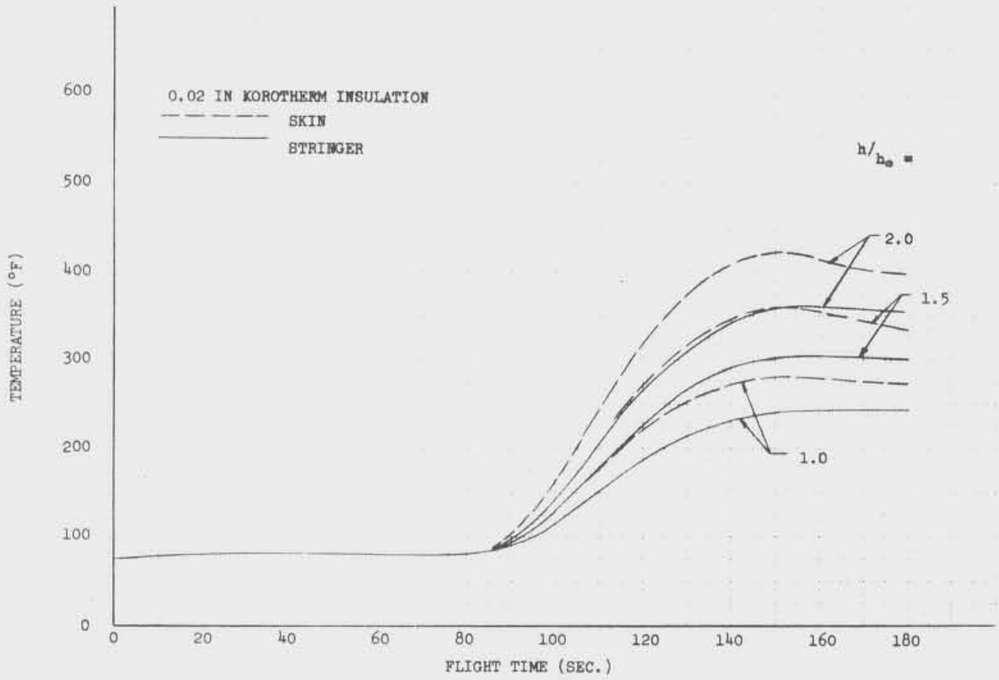

Figure 4.2.4-14. S-IVB Aft Interstage Temperature History

approximately one-half the forward skirt skin temperature (h/ho = 1.0). Peak temperatures in all cases are reached in the 140 - 180 second flight time period, well past the region of maximum dynamic pressures. Hence, structural capability degradation must be accounted for in checking maximum acceleration flight loads.

Comparison of INT-20 vehicle structural loads and thermal environment with S-IVB stage structural capability indicated that the thermal configuration of the Saturn V/S-IVB stage would be satisfactory for the INT-20/S-IVB stage (see following paragraph, 4.2.4.1-b). The maximum predicted temperatures for both configurations are compared in Table 4.2.4-IV. These temperature differences are not large enough to effect any changes in the stage systems for environmental control.

For the INT-20/S-IVB forward skirt, insulation will only be required in the areas of protuberance heating, i.e., the main and auxiliary tunnels, telemetry antennas, and range safety antennas. Approximately 10 pounds of Korotherm TC-320 is required. Similarly, approximately 16 pounds of Korotherm is used on the aft skirt, insulating protuberance areas of the APS (including roll rocket plume effects), LH_2 chilldown return fairing, LH_2 chilldown pump, LH_2 fill and drain line fairing, ullage rockets and LH_2 feed line.

Table 4.2.4-IV

S-IVB SKIN/STRINGER TEMPERATURE COMPARISON
(h/ho = 1.0)

Structure	S-IVB/INT-20	S-IVB/SAT. V
Forward Skirt Skin	417°F	389°F
Forward Skirt Stringer	329°F	320°F
Aft Skirt Skin	277°F	258°F
Aft Skirt Stringer	249°F	235°F
Aft Interstage Skin*	319°F	330°F
Aft Interstage Stringer*	274°F	320°F

*Insulated with 0.01-inches Korotherm

The aft interstage will require approximately 0.01-inches of insulation over the entire surface area, with additional insulation in the wake areas of the aft skirt protuberances. In the areas of the retrorockets, however, which are heavily insulated in the Saturn V configuration, some insulation saving will result due to the rocket's deletion. The resulting Korotherm weight required is approximately 250 pounds.

A minor amount of insulation is used to protect the main and auxiliary tunnels (forward ends), the ullage rocket fairings and the chilldown return line fairing.

b. Structural Capability

The INT-20 baseline vehicle structural loads as presented in Section 4.1.6 were used to develop combined compression and tension load envelopes for comparison with Saturn V/S-IVB stage structural capability. The load envelopes are shown in terms of N_c and N_t, which are combined loads in pounds per inch of circumference for compression and tension, respectively, and are computed for any vehicle station as follows:

$$N_{c_{ult}} = F.S._{ult} \left[\frac{P}{2\pi R} + \frac{M}{\pi R^2} \right] - p_{min} \frac{R}{2}$$

$$N_{t_{ult}} = F.S._{ult} \left[-\frac{P}{2\pi R} + \frac{M}{\pi R^2} + p_{max} \frac{R}{2} \right]$$

where

$F.S._{ult}$ = Ultimate factor of safety, 1.4 for manned flight

P = Axial load, including aerodynamic drag

M = Bending moment

R = Shell radius

p = Net pressure across shell wall (applicable to tank shell only)

When establishing net pressure across the tank wall, ullage pressure, head pressure and ambient conditions were all taken into account. Saturn V/S-IVB stage tank pressure schedules, as presented in Table 4.2.4-V, were used for the INT-20/S-IVB stage.

Table 4.2.4-V

S-IVB STAGE TANK PRESSURE SCHEDULES

LH_2 Tank	Sat. V (503 & Subs) Pressure Range (psia)
Pre-pressurization	28 - 31
First Burn Flight Control	28 - 31
Second Burn Flight Control	28 - 31
Repressurization	28 - 31
Vent and Relief Range	31 - 34
Back up Relief Range	31 - 34
LOX Tank	
Pre-pressurization	38 - 41
First and Second Burn Flight Control	38 - 41
Repressurization	38 - 41
Vent and Relief Range	40.5 - 43.5
Back up Relief Range	42.5 - 45.5

The results of these combined load calculations are illustrated on Figures 4.2.4-15 and 4.2.4-16, which present the S-IVB stage tension and compression load distributions, respectively. As shown by the first figure, S-IVB stage tension load capability is more than adequate to withstand applied tension loads from INT-20 application. Since critical tension loads derive from the maximum $q\alpha$ flight condition, no structural temperature increase is appropriate (structure at room temperature). Dynamic tension loads were not provided for INT-20. They would not, however, be expected to be significantly greater than those for the standard Saturn V; hence, stage tension load capability would be adequate.

Figure 4.2.4-16 presents the S-IVB stage compression load distribution for two conditions, maximum $q\alpha$ and peak acceleration. The load capability curves are for the appropriate temperature condition, i.e., room temperature at time of maximum $q\alpha$ and elevated temperature at peak acceleration. The temperatures shown for the latter case are stringer temperatures, and are for uninsulated structure on the forward and aft skirts and structure insulated with 0.01 inches of Korotherm on the aft interstage. As is shown on the figure, S-IVB stage compression capability is adequate for the proposed INT-20 use.

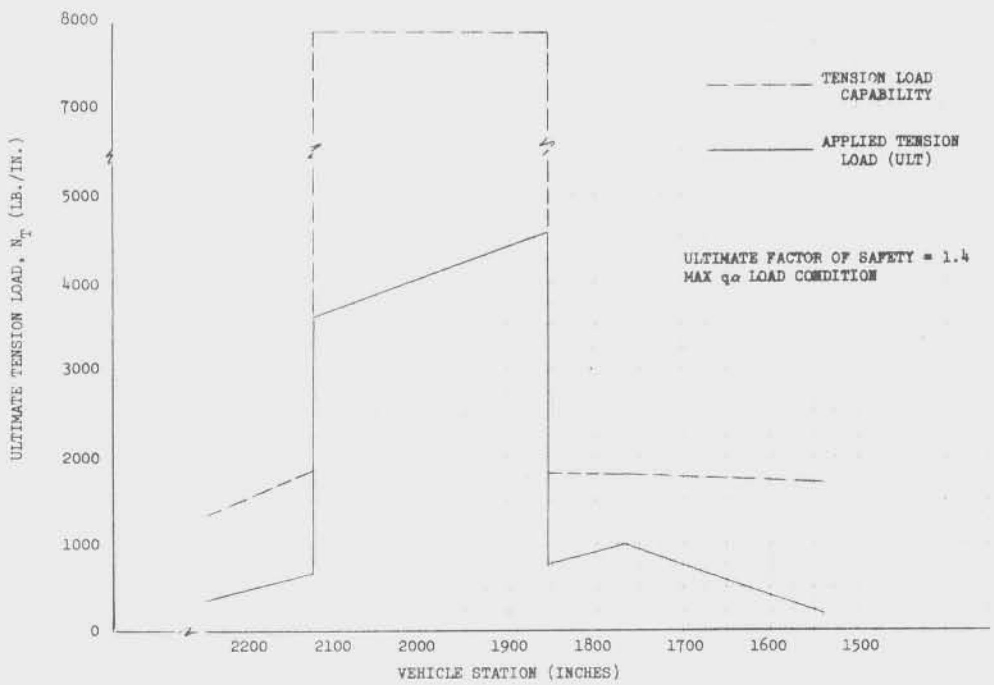

Figure 4.2.4-15. S-IVB Stage Tension Load Distribtuion

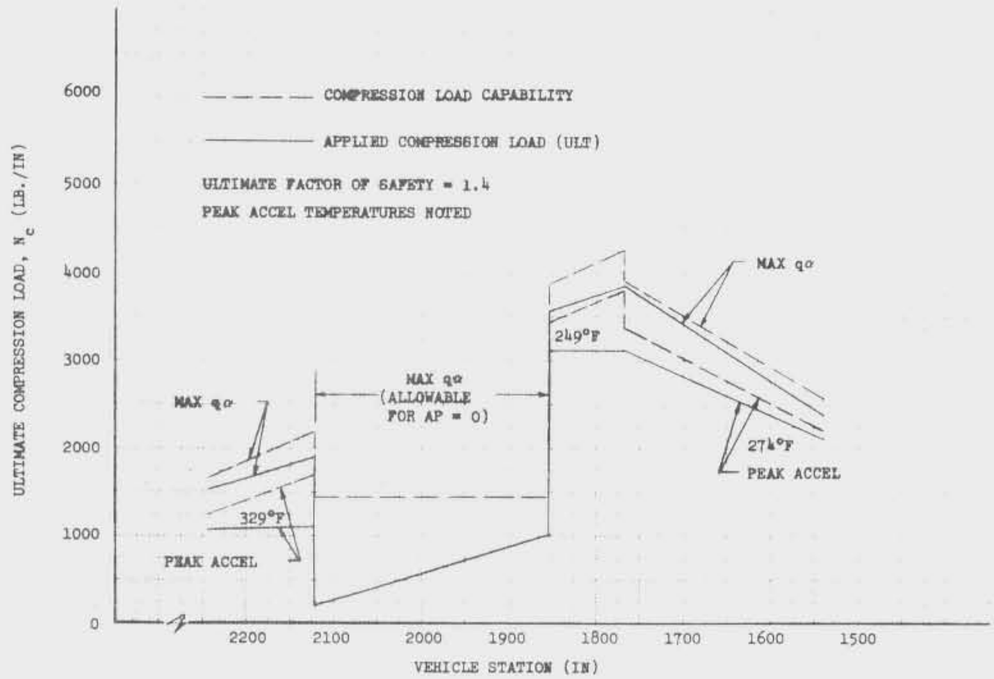

Figure 4.2.4-16. S-IVB Stage Compression Load Distribution

c. Acoustic Environment

Data obtained on the S-IVB stage during Saturn V flights indicate that the dynamic levels during liftoff on some S-IVB critical components were higher than previously predicted (these components have been subsequently requalified to the higher levels). Since the dynamic levels on an S-IVB flown as the second stage on the S-IC booster are estimated to be about 25% higher than the levels on the Saturn V/S-IVB, it is anticipated that some components would need requalification. A brief evaluation was performed based on projections of the Saturn V acoustic and vibration data. The results of the evaluation indicate that approximately ten components and/or subassemblies might require requalification. Section 5. 2. 3 itemizes these selected porbable requalification items, and discusses an attendant requalification program.

Existing Saturn V flight data (AS-501 and -502) were extrapolated in order to predict the acoustic environment for the INT-20 launch vehicle. Figure 4. 2. 4-17 illustrates the variation of sound pressure level with Saturn V vehicle station in the two octave bands of primary interest. These two bands were selected because vibration levels resulting from acoustics in these bands come closer to exceeding qualification levels than in other portions of the spectrum for the liftoff case.

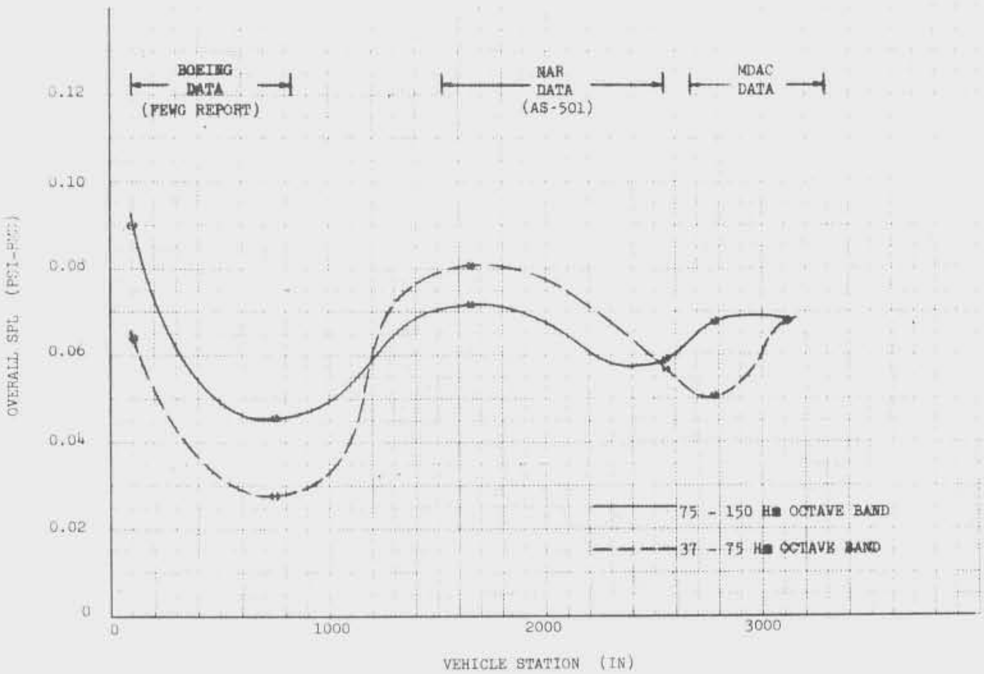

Figure 4.2.4-17. Saturn V Liftoff Sound Pressures Vs Station

The sound pressure levels appear to have a peak and valley characteristic which indicates a peak near vehicle station 1650, which would be in the S-IVB aft skirt-interstage area in the INT-20 configuration. The S-IVB forward skirt environment can be expected to be about one dB lower than the aft skirt based on extrapolation. These peaks may be due to direct radiation from the deflected portion of the exhaust plume.

Octave band spectra are shown on Figure 4. 2. 4-18 for the expected INT-20 liftoff levels. S-IVB levels from AS-501 and -502 are shown for reference. The INT-20 levels were derived from NAR aft skirt data by an adjustment to bring them to the same statistical level as the Boeing and MDAC data. The levels were then reduced by one dB to adjust them to the expected four-engine S-IC environment.

The forward skirt spectra are assumed to resemble the aft skirt levels except for the effects of molecular absorption. Figure 4. 4. 2-17 indicated that the difference in levels at the low frequencies was slightly more than one dB. Higher frequencies roll off slightly faster due to molecular absorption.

The most significant level increase is five dB in the 37-75 Hz octave band. Hence, the probable requalification for some critical components. Basic structure will probably not require requalification because acoustic tests of the S-IVB structure have been performed at levels which are adequate to cover the predicted INT-20 levels of frequencies where structural panel resonance exist.

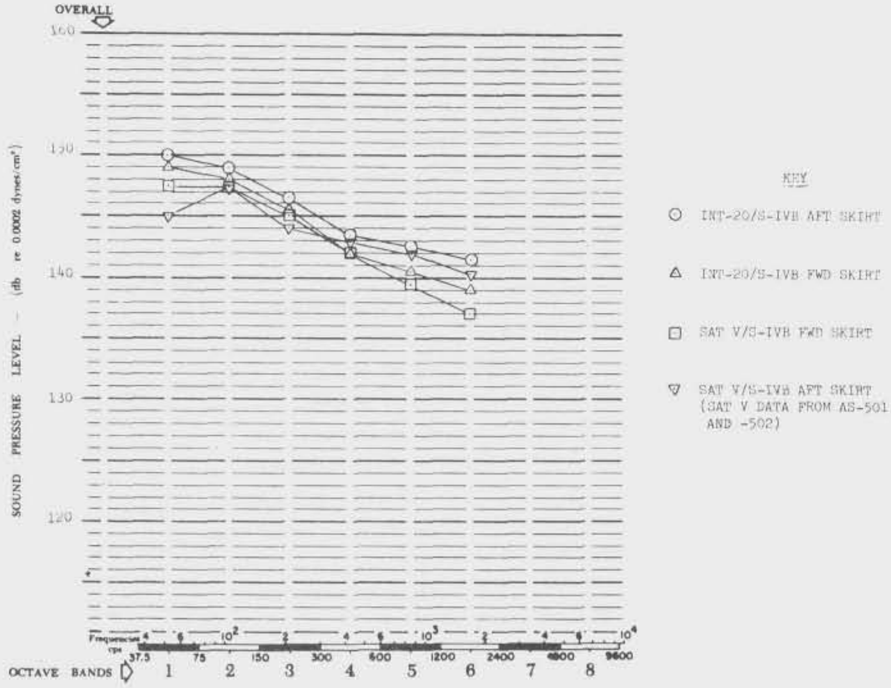

Figure 4.2.4-18. INT-20 Expected SPL at Liftoff

d. Baseline Stage Weights

1. Weight Breakdown

A detailed dry stage weight breakdown for the INT-20/S-IVB baseline stage configuration is presented in NASA format in Table 4. 2. 4-VI. The reference configuration was designated as Saturn V Vehicle -511.

The interstage weight summary is given in Table 4. 2. 4-VII. Included in the category Interstage Structure (W3. 13) is the weight of the new adapter ring structure recommended for S-IC/S-IVB mating. The table also reflects the deletion of the retrorocket cases and all mounting provisions, and the retrorocket propellant (Service Items).

2. Weight Substantiation

The substantiation for the weight changes reflected in Tables 4. 2. 4-VI and 4. 2. 4-VII is presented below.

STAGE WEIGHT SUBSTANTIATION

W3.18 Heat & Flame Protection

Delete retrorocket plume impingement curtain installation	-115 lb
Change to W3.18	-115 lb

W4.7 Fuel System

Delete (5) ambient helium bottles and plumbing	-665 lb
Change to W4.7	-665 lb

W4.8 Oxidizer System

Delete (2) ambient helium bottles and plumbing	-266 lb
Change to W4.8	-266 lb

W4.9 Cryogenic Repress System

Delete O_2H_2 burner	- 56 lb
Change to W4.9	- 56 lb

W6.8 Telemetry & Measuring System

Delete Telemetry Measurements	- 63 lb
Change to W6.8	- 63 lb

W6.16 Auxiliary Propulsion System

Delete (2) Ullage Engines	- 23 lb
Change to W6.16	- 23 lb

INTERSTAGE WEIGHT SUBSTANTIATION

W3.13 Interstage Structure

Add new 5-inch deep adapter ring	+503 lb
Delete (4) existing retrorocket panels and replace with plain structural panels	-104 lb
Delete existing retrorocket intercostals	-204 lb
Change to W3.13	+195 lb

W3. 18 Heat & Flame Protection

Delete Additional External insulation around retrorockets	- 32 lb
Change to W3. 18	- 32 lb

W6. 17 Separation System

Delete entire separation system - 4 retro-rockets, fairings, support structure, ordnance	-727 lb
Change to W6. 17	-727 lb

4. 2. 4. 9 Stage GSE

a. Mechanical GSE

No modifications to existing stage equipment will be required for the INT-20/S-IVB. Minor changes on the aft interstage transportation dolly and weather protection cover will be required due to the attached adapter ring (see Section 4. 2. 3. 3).

b. Propulsion GSE

Propulsion GSE changes will not be required for checkout of the INT-20/S-IVB stage.

c. Electrical GSE

No electrical GSE changes of any significance will be required for INT-20/S-IVB stage checkout. Minor modifications in the form of software changes and patching will be effected to accommodate systems deletions and measurement program reduction.

TABLE 4.2.4-VI

INT-20/S-IVB BASELINE STAGE DRY WEIGHT SUMMARY

NASA Second Generation Breakdown		S-IVB-511 Reference Stage (lbs)	INT-20/S-IVB Baseline Configuration (lbs)
W3.3	Propellant Container	8,933	8,933
W3.6	Forward of Tanks	1,242	1,242
W3.8	Aft of Tanks	1,816	1,816
W3.9	Thrust Structure	774	774
W3.10	Fairings and Associated Structure	197	197
W3.15	Paint and Sealer	104	104
W3.18	Heat and Flame Protection	182	67
W3.0	Structure	13,248	13,133
W4.1	Engine and Accessories	3,572	3,572
W4.6	Purge System for Chilldown	272	272
W4.7	Fuel System	1,573	908
W4.8	Oxidizer System	1,264	998
W4.9	Cryogenic Repressurization System	310	254
W4.10	Stage Control System Hardware	284	284
W4.0	Propulsion System	7,275	6,288
W6.1	Equipment and Instrumentation Structure	430	430
W6.2	Environmental Control System	231	231
W6.5	Control System Electronics	116	116
W6.8	Telemetry and Measuring System	1,165	1,102
W6.10	P. U. System	175	175
W6.11	Electrical System	829	829
W6.12	Range Safety System	69	69
W6.15	Pneumatic System	298	298
W6.16	Auxiliary Propulsion System	855	832
W6.17	Separation System	117	117
W6.18	Ullage System	212	212
W6.20	Systems for Total Vehicle	91	91
W6.0	Equipment and Instrumentation	4,588	4,502
WAD	Stage Dry Weight	25,111	23,923
Change from S-IVB-511 Baseline		0	-1,188

Table 4.2.4-VII

INT-20/S-IVB INTERSTAGE WEIGHT SUMMARY

NASA Second Generation Breakdown		S-IVB-511 Reference Interstage (lbs)	INT-20/S-IVB Interstage (lbs)
W3.13	Interstage Structure*	5678	5873
W3.15	Paint and Sealer	49	49
W3.18	Heat and Flame Protection	523	491
W3.0	Interstage Structure	6250	6413
W6.2	Environmental Control System	17	17
W6.8	Telemetry and Measuring System	15	15
W6.12	Range Safety System	2	2
W6.17	Separation System	727	0
W6.20	Systems for Total Vehicle	10	10
W6.0	Equipment and Instrumentation	771	44
WBD	INTERSTAGE DRY	7021	6457
	SERVICE ITEMS	1062	0
	INTERSTAGE AT GRD. IGN.	8083	6457

*Includes new adapter ring.

4.2.5 Astrionics Systems Adaptation

Modifications to the electrical/electronic systems of the S-IC, the S-IVB and the Instrument Unit are required for adaptation to the INT-20 application.

4.2.5.1 S-IC Stage Astrionics

Changes to the S-IC stage electrical/electronic systems primarily envolve sequence and control, network cabling, and measurements. Network cabling will be "tied-back" for circuit deactivation wherever practical for ease of reversibility. Changes are described in Appendix A-1 and affect the following areas or systems:

a. S-IC/S-IVB interface.

b. Sequence and control.

c. Separation and ordnance system.

d. Measurements.

e. Electrical network distributor wiring.

4.2.5.2 S-IVB Stage Astrionics

System modifications to the S-IVB stage are described in Section 4.2.4.4 and generally consist of propellant utilization (PU) system mixture ratio changes, coiling and stowing of unused wire harnesses and minor interface changes.

4.2.5.3 IU

a. IU Electrical Interface Effects

The IU Interfaces were investigated using applicable Interface Control Documents and the IU schematics for possible effects of the INT-20 baseline booster and payload. The results of this study have shown it unnecessary to change any interface or supporting hardware within the IU. The rationale behind this is discussed in subsequent paragraphs.

1. Spacecraft

For this study it has been groundruled that the standard Saturn V IU/S-C interface will remain as is. Four items, however, deserve some discussion.

The baseline INT-20 vehicle has no Launch Escape System, therefore, no requirement for the Q-Ball Assembly. This unit is powered from the IU and monitored by IU TM. Its absence can be treated in the way the system exists now after Launch Escape System jettison in normal Saturn V boost. The Q-Ball power-on command is done from ESE and power-off from the Switch Selector. The circuitry can be left spare and no power-on issued from ESE.

The Saturn V IU/S-C interface contains three wires called LV Engine Cutoff No. 1, No. 2, and No. 3. These wires are normally at +28V from the spacecraft and hold open relay contacts which when closed cause engine cutoff to be sent to all stages. It is therefore necessary that these relays be energized to prevent engine cutoff or that the function be disabled in some other way. It is possible to prevent engine cutoff by removal of the EDS Engine Cutoff Enable Timer and by deletion of the Switch Selector function called "LV Engines EDS Cutoff Enable" which also enables the cutoff. Again we are assuming the IU/S-C interface will remain as is and, therefore, no effort is required.

A similar situation is the Saturn V IU/S-C interface contains two wires called IU Command System Enable A and B. These wires are at +28V from the spacecraft and hold closed relay contacts which enable the Digital Command System to interrupt the Launch Vehicle Digital Computer. An alternate method is a Switch Selector function called "IU Command System Enable". The Switch Selector could, therefore, be used prior to liftoff to enable the Digital Command System in the absence of the spacecraft. No hardware change is required in this function with or without the present IU/S-C interface.

With the IU/S-C interface open, the functions of spacecraft control of Saturn, which is a backup to the IU Guidance System and

4.2.5.3 (Continued)

the Emergency Detection System that is integral with spacecraft functions, will not be operable.

No other problem areas exist at the IU/S-C interface and the remaining wires can be left open as spare with no effect on the IU performance.

2. S-IVB

The IU/S-IVB interface can be left as is with no change required. Wiring that was studied for possible impact was associated with: S-II Stage, S-IC center engine, and O_2H_2 Burner Malfunction. It was found that without exception the present wiring could be left spare with no effect on the IU performance as required by the INT-20 baseline booster.

b. IU Subsystem Effects - Baseline Vehicle

1. Guidance and Control Subsystem

(a) Hardware

(1) Requirements

The current functional requirements for the Saturn V FCC are:

Control loop compensation.
Signal mixing.
Gain program implementation.
Control mode determination.

The compensation and signal mixing duties will not change in adapting to an INT-20 configuration. However, additional requirements will exist with respect to gain program and control mode functions because there is no S-II stage and the mandatory S-IC stage control gain switch associated with cutoff of two outboard engines.

There are presently two S-IC switchpoints available. Preliminary control system analysis of this stage indicated that with the early outboard engine cutoffs at least three gain changes would be required in order to maintain acceptable stability margins. Therefore, the S-IC stage switchpoint requirement for the INT-20 will be listed as four (the additional one to allow increased system complexity as required without a further hardware impact). The analysis also indicated that there is a potential stability problem

4.2.5.3 (Continued)

if the gain switch that is to occur simultaneously with the g-level engine cutoff happens before the thrust begins to decay. With nominal hardware/software performance there should be no concern, but it is an area that warrants further investigation (i.e., determine the maximum amount of time difference between the occurrence of the two events before problems insue and then devise techniques to safeguard against the time lapse in obtaining that level).

The control mode function is altered in that with the elimination of the S-II stage, provisions must be made to insure that upon completion of the S-IC staging, the S-IVB stage control system is activated.

(2) Implementation

In order to provide the four S-IC switchpoints in a manner that would produce minimum impact on the present S-V configuration, two presently unused switchpoints will be utilized. The IU networks provide the FCC interface with nine switchpoints. The first six are presently used and the last three are terminated at the FCC interface. Therefore, two of these will be routed to the S-IC filters. This will require four wires to be added to the FCC cable harness and Motherboards 6 and 7 to be redesigned. All S-IC filters are located on Motherboards 6 and 7.

Since the present Saturn V configuration has an internal latching arrangement for the S-IC stage and the only initiation of the S-II burn signal will release the latch, and investigation was made to determine if a redesign of the Switching Control Board and Switching Circuit "C" would be required. It was determined that an S-II burn signal will be initiated on the INT-20 vehicle as part of the S-IC engine cant removal circuitry (the same switch selector function is used for both). Thus the S-II burn mode will be entered momentarily after S-IC cutoff and prior to staging. Therefore, this S-II burn signal will be used to release the internal latch and there will be no redesign of the FCC required.

4.2.5.3 (Continued)

(b) Software and Related Activities

(1) Requirements

The baseline INT-20 mission was evaluated to determine impact on flight software and related activities (flight program verification and guidance dynamics analysis). The baseline mission is direct injection to a 100 N.M. circular orbit with a fixed launch azimuth.

In determining impact of the INT-20 vehicle, techniques currently used for LOR mission flight software development served as a reference. Since the baseline INT-20 mission is essentially contained in the LOR mission, software changes are minimal. Those software changes which are required result from: (1) elimination of the S-II stage and its associated discretes and time base, (2) the requirement to perform S-IC two engine shutdown based on a g-limit and to stage S-IC cutoff with a g-limit test, and (3) mission simplification which permits elimination of certain software routines. If the digital control option is selected, significant changes to flight software will be required.

Because of software changes for the INT-20 mission, program verification will be impacted. Simulators currently used for the LOR mission must be modified and the new program logic must be verified.

Impact of the INT-20 on guidance dynamics is very minimal since these studies deal with flight phases using closed-loop (IGM) guidance. Thus, only guidance dynamics during the S-IVB burn need be studied, and this burn is quite similar to that of the LOR mission.

(2) Flight Program Modifications

The baseline INT-20 mission can be accommodated within the framework of the present LOR program by: (1) deleting portions of the LOR program which are not applicable, (2) making data changes of the usual mission-to-mission type, (3) adding a small amount of logic to the program, and (4) making significant program changes only if the digital control option is exercised.

4.2.5.3 (Continued)

Flight program modifications are summarized in Table 4.2.5.3-I. Where specific requirements for INT-20 are lacking, as for example in the case of the digital command system or telemetry, it has been assumed that flexibility inherent in the LOR program will hold changes to data alone.

Flight functions, which either involve logic changes or are deserving of special note, will now be discussed further.

Cutoff of S-IC Engines

Current mission planning for the INT-20 calls for shutdown of two of the S-IC engines before acceleration reaches 4.68g's and for stage cutoff at fuel depletion. The flight program will issue the cutoff signal for two engine shutdown and will provide a backup signal for S-IC cutoff. Logic changes will be required to the program in order to properly set these signals.

The two engine shutdown signal will be generated by the flight program so that acceleration does not exceed 4.68 g's. This can best be accomplished by inserting equations and logic to compute the time at which the g-limit will be reached. The cutoff signal will then be based on this time. By computing the time from measured acceleration, the S-IC stage will be used more effectively in low performance cases than is possible with a preset cutoff time. The changes required to the flight program are straight forward.

S-IC stage cutoff will also be done utilizing the same program computing the time at which the g-limit will be reached. Again computing the time from measured acceleration, the S-IC stage will be used more effectively in off nominal performance cases than is possible with a preset cutoff time. In both cases two redundant cutoff commands will be issued via the S-IC switch selector making the cutoff implementation fully redundant.

Further logic changes to the program will be required to protect against a single S-IC engine out. Two discretes are being added to isolate an engine out to either the engine pair 1 and 3 or the engine pair 2 and 4. Thus, if an engine out occurs prior to two engine shutdown, the program will

TABLE 4.2.5.3-I. SUMMARY OF FLIGHT PROGRAM MODIFICATIONS

PROGRAM FUNCTION	TYPE OF MODIFICATION
Ground Retargeting	Delete
Variable Launch Azimuth	Delete
Accelerometer Processing	Data Change
Boost Navigation	Data Change
Boost Guidance	Data Change
S-IVB Cutoff	No Change
Orbital Navigation	Data Change
Orbital Guidance	Data Change
Telemetry Acquisition and Loss	Data Change
S-IVB Restart	Delete
Time Bases	Minor Logic Changes
Discretes	Minor Logic Changes
Interrupts	Data Change
Attitude Control	Without digital control -data changes With digital control -major additions of logic and equations
Switch Selector Processing	Data Changes
Digital Command System	Data Changes
Telemetry & Data Compression	Data Changes

4.2.5.3 (Continued)

determine which pair of engines is affected. When the g-limit is attained, the program will command the affected pair to shutdown. This prevents shutting down two good engines in a single engine out case. Logic changes to the program are minimal.

Boost Guidance

For first stage guidance, only data changes will be required for the baseline mission. Guidance for the single engine out case also is assumed to involve only data changes.

Second stage guidance, using a brief attitude freeze followed by IGM, can also be provided by data changes. For this purpose, the "abort-to-orbit" logic already in the LOR program will be used.

Time Bases

All time bases past Time Base 2 will be affected in some measure by the use of an INT-20 vehicle. Those changes currently identified involve only minor logic changes to the program.

Time Base 3 will govern S-IVB burn rather than S-II burn in the INT-20. This time base will terminate with S-IVB cutoff and Time Base 4 will begin. Thus, orbital flight will be governed by Time Base 4. Use of the other LOR time bases is not defined at this time for the INT-20. Modification to these time bases will be required; however, changes are expected to have a minor impact on flight software.

Discretes

Addition of the two S-IC engine out discrete inputs will require modification of program logic. This modification has previously been discussed under "Cutoff of S-IC Engines".

Attitude Control

Attitude control software is impacted only if the digital control option is exercised. In this case, a significant amount of logic and equations will be added to the program. See Section 4.3.4 for further discussion of the proposed digital control system.

4.2.5.3 (Continued)

Switch Selector Processing

The modified sequences required by the INT-20 can be provided by changing switch selector data tables. The capability to insert a computed time for S-IC two engine cutoff into the switch selector table is currently present in the LOR program.

(3) Flight Program Verification

Flight program changes are minimal and fall within the normal mission-to-mission changes.

(4) Guidance Dynamics Analysis

Guidance dynamics analysis for the INT-20 will be limited to the S-IVB stage burn. From a guidance dynamics viewpoint, this burn will be bracketed by the Saturn V and the Saturn IB S-IVB burns. Thus, only mission-to-mission type changes will be required for guidance dynamics analysis.

2. Electrical Subsystem

(a) Requirements

There are no new requirements placed on the IU Electrical Subsystem as a result of internal change to the IU. There is slight reduction in requirements for +28V power from the 6D10 battery which is used to power busses in other stages to allow the stages to send discretes to the IU with IU power. The removal of the S-II stage causes this small change in requirement.

(b) Implementation

There is no change in the IU Electrical Subsystem resulting from implementing the INT-20 baseline booster and payload. The S-II associated wiring will be left spare and system will look the same as it does on Saturn V after the S-II is staged.

3. Instrumentation and Communication Subsystem

(a) Requirements

There is a reduced requirement for measurements because of S-II stage and Q-Ball deletion.

4.2.5.3 (Continued)

The AS-511 baseline and min-mod approach dictate the use of the present telemetry system on AS-511 as opposed to the Saturn IB system.

(b) Implementation

There are six measurements associated with the Q-Ball Assembly. These can be left spare but in all probability will actually be used for the Q-Ball.

S-II stage associated measurements are:

8-Actuator Position.
8-Valve Current (each actuator).
3-Discrete measurements indicating:

S-II Stage Separation.
EDS Manual S-II/S-IVB Separation Sequence Start.
S-II Burn Mode.

The Actuator Position and Valve Current measurements are time shared channels also used for the S-IC stage, therefore, removal would at best save some switching circuits.

The discrete measurements because of changes outside the IU will in some cases change their names. S-II Stage Separation will become S-IC Stage Separation. EDS Manual S-II/S-IVB Separation Sequence Start will become EDS Manual S-IC/S-IVB Separation Sequence Start. S-II Burn Mode will be unchanged.

The I&C hardware does not require any change. It is recommended that the slight amount of unrequired hardware be left as spare.

4. Environmental Control System

(a) Requirements

There are no ECS modifications required to support the INT-20 mission. The electronic components remain essentially unchanged, therefore, temperature control requirements are unchanged. The environments in which the IU must operate have been investigated and found to be compatible with ECS capabilities. These environments include acceleration and a dynamic heating during boost, and orbital heating.

4.2.5.3 (Continued)

5. Structure Subsystem

(a) Description of Present Structure and Function

The IU structure is a honeycomb composite cylindrical section, 260 inches in diameter and 36 inches high. The structure is manufactured in three segments. The total shell thickness is 0.95 inch. The inner face sheet is .020 inch thick 7075-T6 aluminum; the outer face sheet is .030 inch thick 7075-T6 aluminum. Extruded channel sections at the upper and lower interfaces introduce the load from adjacent stages. Bonded brackets and pads on the inner surface facilitate the mounting of Guidance and Control Components and Environmental Control System Equipment. The structure has a load carrying bolted access door and umbilical panel. The specific IU structure configuration which will be considered for this study will be the Saturn V, or 500 series, IU. The structure differs from the Saturn IB, or 200 series, primarily in that the Saturn V has external cork insulation, a pad of vibration damping compound in the ST-124 area, and different antenna mounting provisions. The Saturn V configuration has a higher in-flight load carrying capability at End Boost Condition by virtue of the external insulation.

(b) INT-20 Baseline Loads and Environment and Design Criteria

The interface loads were determined by The Boeing Company and presented in the Reference 3.1.3.6-1 document, which was subsequently updated by Reference 4.2.5.3-1. For access door installation, a 1.0 Factor of Safety should be used for the loads imposed at the IU interface due to the 95% March Wind Condition (Table 6-I of Reference 3.1.3.6-1). Deflection at the access door opening is the major consideration for access door removal and installation.

A comparison between the baseline INT-20 lower interface loads and the present IU structural capability is shown in Table 4.2.5.3-II for the various load conditions of concern. The lower interface loads are always worst case loading in the IU structure when peaking loads are neglected.

The table illustrates the baseline Saturn V IU structure is capable of withstanding the required loads. The payload weight (above IU) for the INT-20 baseline vehicle is given at 132,026 lbs. The Saturn V payload weight (above IU), including Launch Escape System, is