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A "Zero Stage" for the Saturn IB Launch Vehicle

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THERE ARE MANY WAYS to increase the payload capability of liquid propellant launch vehicles that have been investigated by the Chrysler Corp. Space Division (CCSD) to increase the effectivity of the successful Saturn IB. The most basic method is to increase the energy available in any or all the vehicle stages through the use of higher energy fuels or by increasing the quantity of propellants available. Another method for increasing the payload capability of a vehicle involves the modification of existing hardware to increase the thrust of the lower stage or stages, that is, by modifying the original liquid propellant engines or augmenting thrust with parallel liquid engines or solid propellant motors. Finally, increased vehicle performance can be realized through the addition of an entirely new stage to the vehicle which is ignited either before or after the original vehicle stages. The discussion that follows considers this last method of upgrading vehicle performance -- the addition of an entirely new "Zero Stage" to the Saturn IB or "core" vehicle. The term "Zero Stage" is used here because the new stage is ignited before the first or "S-IB" stage of the Saturn IB vehicle. This new stage consists of four 120 in. solid propellant motors, developed for and presently used in the Titan III program, clustered symmetrically around the S-IB stage. These solid propellant or "Zero Stage" motors are ignited at liftoff, and power the vehicle above the dense lower atmosphere, thus allowing the basic Saturn IB to accelerate a larger payload to the required velocity for orbital injection.

BASIC SATURN IB VEHICLE

The basic Saturn IB vehicle (Fig. 1) consists of two stages, the S-IB and S-IVB, and the Instrument Unit (IU). This configuration is capable of placing approximately 40,000 lb of payload into a low (100 nautical mile) earth orbit.

S-IB OR FIRST STAGE - The S-IB stage makes use of tanks developed and qualified for the Redstone, Jupiter, and Saturn I vehicles. The engines for this stage, developed and proven for the Thor, Atlas, Jupiter, and Saturn I programs, have been uprated for additional thrust. By clustering eight of these uprated H-1 engines, a thrust of 1.6 million pounds is generated at launch, or liftoff, to propel the vehicle to an altitude of approximately 31 nautical miles.

The liquid oxygen (LOX) and kerosene (RP-1) propellants carried aboard the S-IB are stored in nine separate tanks (five LOX and four RP-1). Four hydraulically gimbaled engines permit flight path control as directed from the guidance system in the Instrument Unit. To separate the first stage

ABSTRACT

To meet the demands of increasing payload size and weight, and to fill the large payload gap between the Saturn IB and Saturn V, a number of methods of uprating the Saturn IB have been studied by NASA and Chrysler Corp. of providing increased payload capability is discussed in this paper. Four 120 in. United Technology Center UA-1205 solid propellant motors, originally developed for the Air Force Titan III program, are clustered around the S-IB first stage of the Saturn IB launch vehicle. These four solid propellant motors provide the total thrust for liftoff of the vehicle, with S-IB stage ignition occurring just prior to burnout and separation of the solid propellant motors. The term "Zero Stage" is applied to this added stage.

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phases. Phase 1 will be temporary and will consist of small bases with temporary occupancy capabilities deployed in several areas on the lunar surface. Sufficient time will be allowed between each lunar scientific mission or temporary base emplacement to allow for review of scientific data returned from a previous mission. Each base will be capable of supporting three men for approximately three months. The major scientific activity at such bases will be oriented toward the geosciences.

Phase 2 will be semipermanent and will consist of a buildup of capabilities (perhaps at a previous Phase 1 site) to support a base crew of approximately six men for six months to two years. Major scientific investigations will be oriented toward greater scientific depth in geosciences, with significant scientific investigations being conducted in astronomy and radiation.

Phase 3 is foreseen to be the establishment of a permanent base support scientific mission for 5-10 years or longer. The minimum base crew will be 12 men performing major investigations in astronomy along with significant geosciences investigations. However, much of the geoscience activity will be supported by smaller, less permanent bases established during the earlier phases. A limited number of scientific missions will also be made to other sites occupied intermittently and simultaneously with the permanent base occupation.

With appropriate extension upgrading and modification of hardware under development, it appears feasible to support the scientific missions presently envisioned by a maximum of six earth launches per year. The foregoing program assumes the general utilization of earth technology on the lunar surface. Logistics requirements may be significantly reduced by possible development of a technology specifically tailored to the lunar environment.

from the second, four solid propellant motors, each producing 36,000 lb of thrust for 1.5 sec, are employed as retrorockets. These motors are attached to the aft interstage (see the S-IVB stage in Fig. 1) between the first and second stages.

S-IVB OR SECOND STAGE - The S-IVB stage is powered by a single J-2 engine that burns liquid oxygen (LOX) and liquid hydrogen (LH_2) and provides a thrust of 200,000 lb. Control signals from the Instrument Unit are relayed to hydraulic gimbals on the J-2 engine, thus providing stage pitch and yaw control. Two auxiliary propulsion systems, located 180 deg apart on the circumference of the stage near the separation plane, furnish roll control and attitude control during coasting. After first-stage separation, the J-2 engine is ignited and the S-IVB propels the payload to the desired altitude. To insure that the propellant is settled in the bottom of the tanks during periods of weightlessness, three solid propellant ullage rockets are fired at the time of stage separation, just prior to ignition of the J-2 engine.

INSTRUMENT UNIT - The Instrument Unit (IU) houses the vehicle guidance and control systems and the flight instrumentation systems for the Saturn IB launch vehicle. Development and operational verification of this unit is a vital step in the overall Saturn/Apollo program since this unit, with minor modifications, will be used to control the Saturn V vehicle during the early phases of its lunar round trip. This unit serves as a structural link between the S-IVB stage and the payload or third stage. It contains the vehicle's electrical, guidance and control, instrumentation, measuring telemetry, radio frequency, environmental control, and emergency detection systems.

Inertial sensors and an inflight command link enable a variety of missions to be flown without system modification. The guidance computer features flexibility through extensive memory capacity and the ability to divide the memory between data and storage, as desired.

The guidance equations used by the guidance computer are based on path-adaptive guidance, a self-compensating concept, through which optimum trajectories can be flown for specific missions.

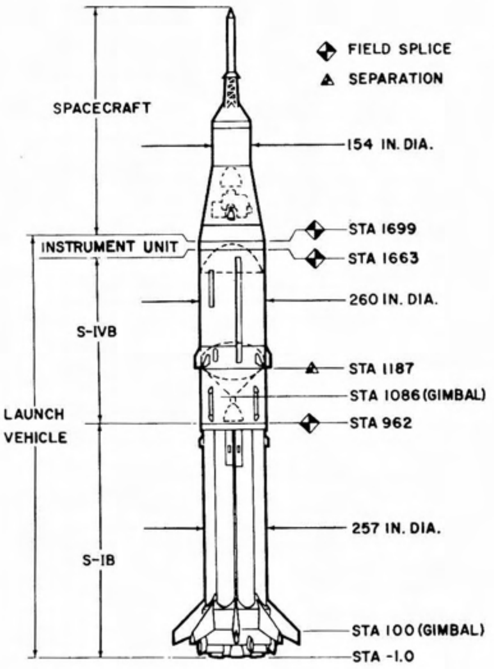


Fig. 1 - Saturn IB configuration

VEHICLE CONFIGURATIONS

GENERAL - Preliminary performance, propulsion, and structural tradeoff analyses performed by CCSD resulted in the selection of a Saturn IB/Zero Stage configuration consisting of a modified S-IB stage, four 120 in. diameter solid propellant rocket motors (SRM) mounted to the S-IB stage, a standard S-IVB stage, an instrument unit (IU), and Apollo modules or Centaur-plus-payload modules (Fig. 2).

The four SRM's will be mounted parallel to the S-IB stage and attached through separating joints to the outriggers of the thrust structure at the base of the stage and to the spider beam at the top of the stage.

ASTRIONICS SYSTEM - The astrionics system of the Saturn IB/Zero Stage vehicle is shown in a simplified block diagram (Fig. 3). The major portion of the equipment in this system is located in the IU. The astrionics system performs the functions of SRM TVC guidance and control, H-1 ignition, SRM staging, and SRM thrust termination in addition to those now performed.

The principal components of the original Saturn IB astrionics system requiring modification to perform the above functions for the Saturn IB/Zero Stage vehicle are the control computer networks and gains, the Launch Vehicle Digital Computer (LVDC) programs, and the S-IB switch selector (increased capacity). These system modifications will be discussed in the appropriate stage sections that follow.

OPERATIONAL SEQUENCE OF EVENTS - The following sequence of events has been selected for Zero Stage ignition and separation from the Saturn IB core vehicle. The four SRM's will be ignited on the pad and will be the only source of thrust through the first phase of flight. The S-IB

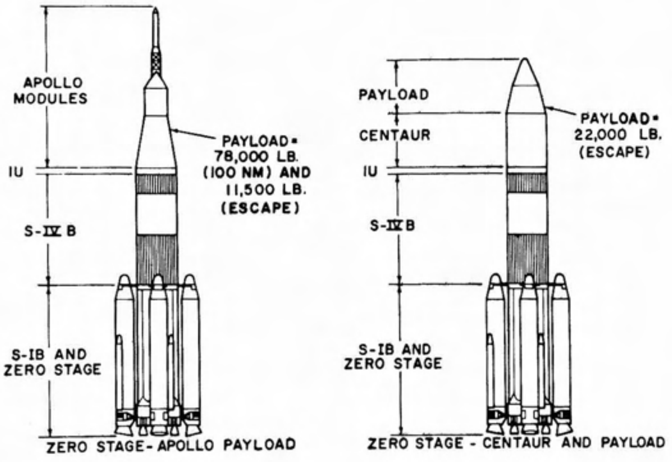


Fig. 2 - Vehicle configurations

stage H-1 engines will be ignited at approximately 108 sec after liftoff as the thrust of the SRM's nears termination and, at approximately 115 sec after liftoff, the SRM's will be separated from the S-IB stage. Figs. 4 and 5 illustrate the thrust characteristics of the Saturn IB/Zero Stage vehicle.

ENVIRONMENTS

TRAJECTORY PARAMETERS

Method of Analysis - In analyzing trajectories for the Saturn IB/Zero Stage vehicle, it was apparent that the ideal ignition-jettison sequence could be determined by performing a tradeoff between delaying ignition of the H-1 engines until SRM thrust has tailed off sufficiently to maintain axial acceleration at a reasonable level as the S-IB reaches full thrust, and igniting early enough to insure that S-IB thrust variations during thrust buildup are completed before the SRM's are jettisoned. The best time for jettisoning the SRM's was established as that point at which the axial acceleration of the SRM's alone became less than the axial acceleration of the total vehicle, since, from that point on, the SRM's degrade rather than improve vehicle performance. Using the above criteria, a specific ignition-jettison sequence was established.

With the sequencing thus established, various pitch programs were investigated to determine which shaping yielded maximum payload. The basic shaping procedure involved:

1. 15 sec vertical rise to allow launch pad clearance and any required roll maneuver.
2. 10 sec tilt-over at a constant pitch rate.
3. Gravity turn flight through the maximum dynamic pressure region and past SRM jettison to minimum angle-of-attack during these periods.
4. Pitch-up maneuver.
5. Constant attitude flight through S-IB burnout.
6. Constant pitch rate through S-IVB burning.

The pitch-up maneuver was varied in magnitude, duration, and time of initiation, and the pitch rates in the second and sixth steps above were adjusted to achieve the desired injection conditions. A limit on pitch rate of 1 deg/sec was applied to stay within Saturn IB engineering ground rules. The combination of rates and timing yielding maximum payload was determined and then used in establishing the reference trajectory.

Analysis Results - Using the above referenced weights, sequencing, and trajectory shaping techniques, a payload capability of 76,100 lb in a 100 nautical mile orbit was established for the manned Saturn IB/Zero Stage vehicle. This capability reflects an Apollo type payload with the launch escape system (LES) jettisoned after S-IVB ignition. This trajectory resulted in a maximum dynamic pressure of 388 psf and a maximum axial acceleration of 3.82 g. Figs. 6-9 present the time histories and pertinent parameters for this trajectory. The payload capability of the Saturn IB/Zero Stage configuration for a non-man-rated mission was established as 78,000 lb.

In order to allow rapid evaluation of payload increments resulting from small changes in vehicle parameters (thrust, structural weight, and so on), payload-exchange ratios were

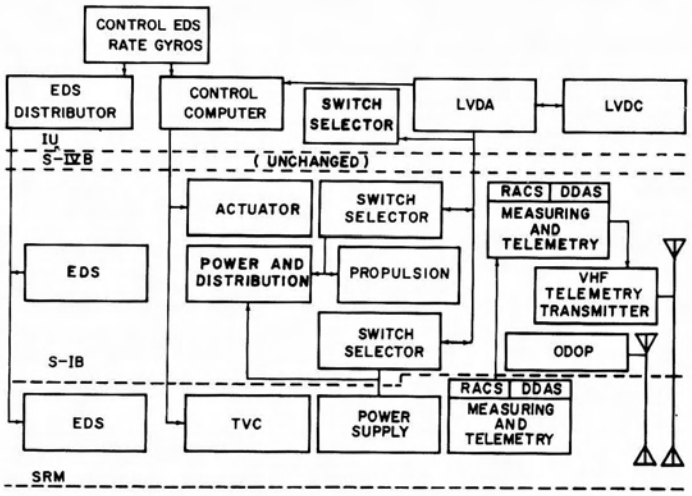


Fig. 3 - Saturn IB/Zero Stage astrionics system

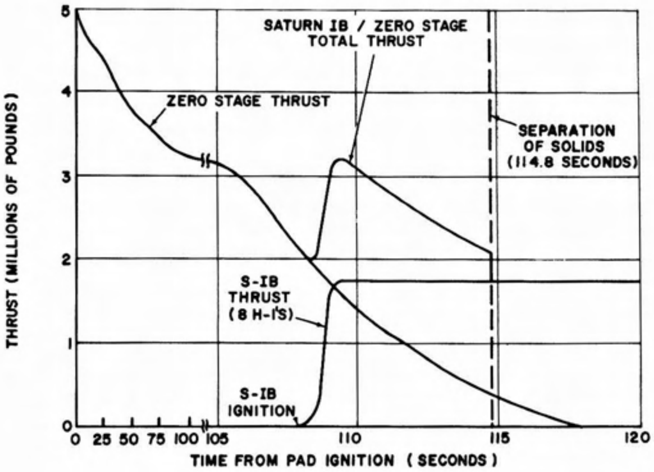


Fig. 4 - Thrust trace for Saturn IB/Zero Stage vehicle

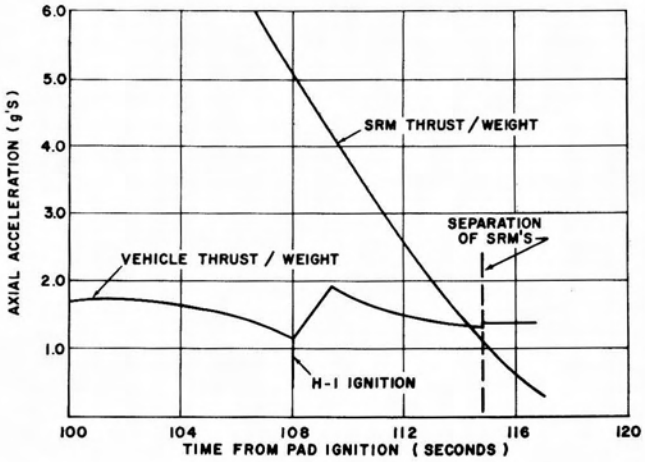


Fig. 5 - Acceleration trace for Saturn IB/Zero Stage vehicle

so determined. Using the reference trajectory as a base, variations in significant vehicle parameters were analyzed to determine their effect on payload. Variations in one direction only were investigated; the partials obtained were assumed to be applicable in either direction. The partials obtained are presented in Table 1. The figures presented in the "partials" column of this table indicate the change in payload resulting from a unit change in the vehicle parameter.

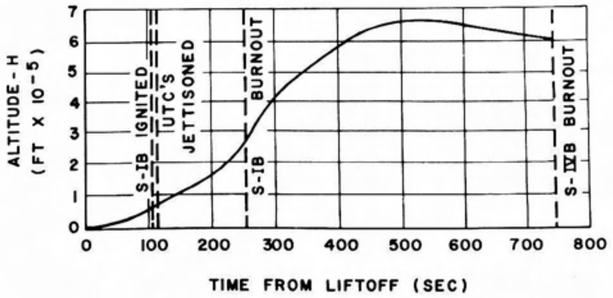


Fig. 6 - Altitude versus liftoff time

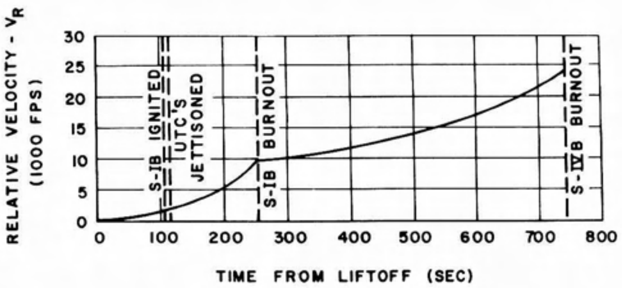


Fig. 7 - Relative velocity versus liftoff time

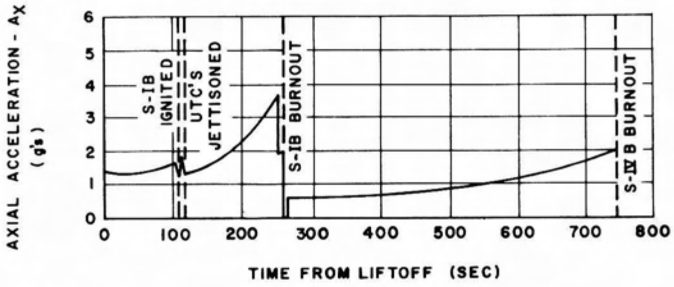


Fig. 8 - Axial acceleration versus liftoff time

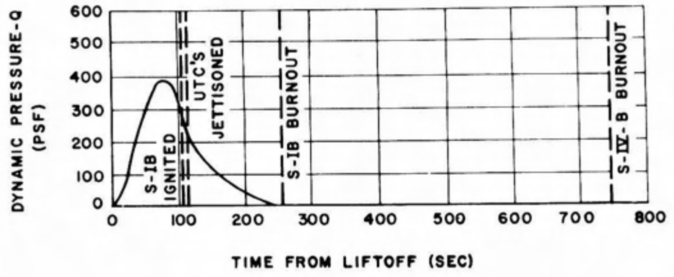


Fig. 9 - Dynamic pressure versus liftoff time

THERMAL ENVIRONMENT - Detailed thermal analyses were performed on the Saturn IB/Zero Stage vehicle in two primary areas, the S-IB base region and tank walls, since both the convective heating (dependent upon the flow fields and shock impingement areas produced by the four SRM's) and radiant heating to these areas are affected by the addition of SRM's to the S-IB stage.

An SRM exhaust plume contains approximately 38% alumina particles (by weight). These particles are larger and at higher temperatures than the carbon particles in the H-1 engine plumes. The temperature of these particles ranges from 2370-5070 F for particle sizes from 1.58-7.90 microns in diameter, respectively. This range is due to the slower cooling rate (thermal lag) of the larger particles as referenced to the gaseous temperatures. Knowing temperatures and particle cloud effective emissivities, the radiant heating rates to any area of interest on the vehicle can be determined as a function of time using standard radiation equations with the appropriate view factor. During SRM burn, the radiant heat from the four exhaust plumes will provide a varying heat input to the entire aft end of the vehicle, with the most severe heating occurring in the base region. The outer heat shield areas nearest the four SRM's were found to have the maximum view of the SRM exhaust plumes and will therefore receive the maximum radiant heat input. The magnitude of these radiant heat fluxes will be relatively constant at approximately 8 Btu/ft^2-sec, a relatively small value when compared to the existing S-IB stage heat shield design criteria. Fig. 10 shows the maximum S-IB stage heat shield incident radiation environment for the Saturn IB/Zero Stage configuration compared with the maximum radiation flux for the standard Saturn IB con-

Table 1 - Performance Partials

Stage	Parameter (x)	Partials
Zero	W_{Dry}	-0.0404 lb/lb
S-IB	Thrust	0.0176 lb/lb
S-IB	I_{sp}	368.04 lb/sec
S-IB	W_{pc}	0.0098 lb/lb
S-IB	W_{Dry}	-0.2024 lb/lb
S-IVB	Thrust	0.1475 lb/lb
S-IVB	I_{sp}	252.2 lb/sec
S-IVB	W_{pc}	-0.0018 lb/lb
S-IVB	W_{Dry}	-1.0 lb/lb
All	Drag	-16.7 lb/%

figuration. These radiant environments are plotted as a function of altitude; however, they can be converted to time for any desired trajectory.

Prior to SRM separation, the convective environment to the aft portion of the vehicle will be greatly influenced by the alteration of external flow caused by the SRM's and associated plumes. Immediately following S-IB stage H-1 engine ignition, and prior to SRM separation, the base heat shield convective environment should increase because of the plume interactions and flow reversal. During this short time interval, the combined convective and radiant heat inputs from both the H-1 and SRM engine plumes should remain below the maximum value of the existing S-IB stage heat shield design criteria. The integrated heat load to the S-IB stage base region, between the time of Zero Stage ignition and S-IB stage engine cutoff, will, however, approach that of the design criteria because of the longer time interval involved, approximately 142 sec versus approximately 252 sec. This longer interval allows for additional heat soakage through the thermal protection and, as a result, a few selected areas such as the S-IB J-ring may require additional insulation. The H-1 engine nozzles may also be a program area since each engine will be exposed to two SRM plumes. Prior to H-1 engine ignition, no coolant will be supplied to these engine nozzles, and the resulting radiant environment may produce critical heating of the unshielded H-1 engine fuel tubes.

Preliminary indications are that no problems will result from recirculating convective heating along the booster tanks prior to S-IB H-1 engine ignition. The SRM plumes will not intersect in the 0-70,000 ft altitude range of the Saturn IB/Zero Stage vehicle; therefore, no recirculating gases will be present. However, following H-1 engine ignition and prior to and during SRM separation, the H-1 and SRM plumes will intersect. During this span of approximately 7 sec, a recirculation convective flow field may be established. The possibility that recirculation may occur, and the potential effects on the booster tanks and tail shrouds, will require additional investigation.

Heating rates to the interstage will be increased due to the impingement of a bow shock which will be present at the nose of the SRM's. As shown in Fig. 11, this increase produced by shock impingement, coupled with the longer heating time of the Saturn IB/Zero Stage vehicle trajectory, will result in an integrated aerodynamic heating load to the interstage that is more than double the heat load presently expected on the standard Saturn IB vehicle. Additional heating to the S-IB stage may occur as a result of flow in the region between the SRM's and S-IB stage. However, before conditions in this area can be effectively evaluated, pressure tests of the Saturn IB/Zero Stage configuration must be performed. Preliminary analyses indicate that this heat load will not be critical, although the aerodynamic heating load produced by the Saturn IB/Zero Stage configuration and trajectory will be more severe, in general, than that calculated for the standard Saturn IB configuration and maximum heating trajectory.

ACOUSTIC ENVIRONMENT - The jet noise and aerodynamic noise effects on the Saturn IB/Zero Stage Apollo configuration were analyzed and are presented in Figs. 12-14. Fig. 12 presents the acoustic environment as external overall sound pressure level (OASPL) versus vehicle station (0-2300 in.). Figs. 13 and 14 present the power spectral density distribution expressed in 1/3 octave band sound pressure level versus frequency for jet noise for two vehicle stations, station 54 and station 1698. Specification values for the basic Saturn IB vehicle have been shown for comparison purposes.

Two points are worthy of note:

1. Saturn IB/Zero Stage OASPL is lower than Saturn IB design OASPL.

2. Saturn IB/Zero Stage OASPL contains more acoustic energy in the low frequency range than Saturn IB design specifications.

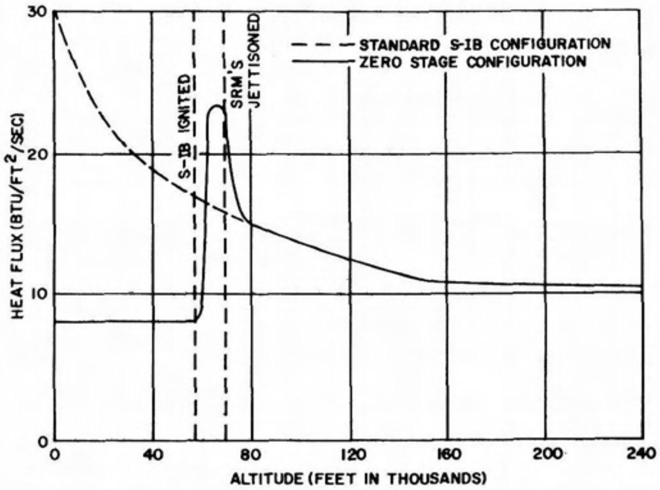


Fig. 10 - S-IB stage heat shield maximum radiation environment

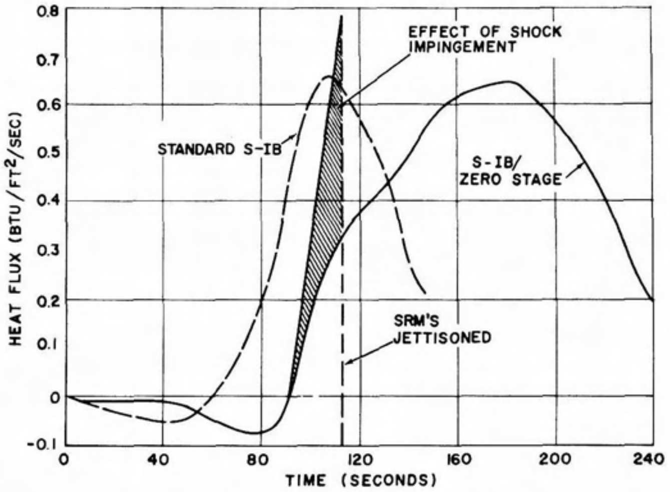


Fig. 11 - S-IB forward interstage aerodynamic heating environment

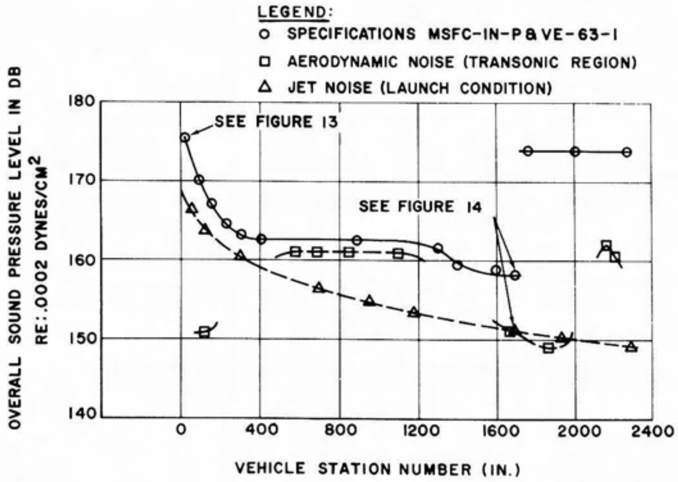


Fig. 12 - OASPL acoustical envelopes for Zero Stage with Apollo payload

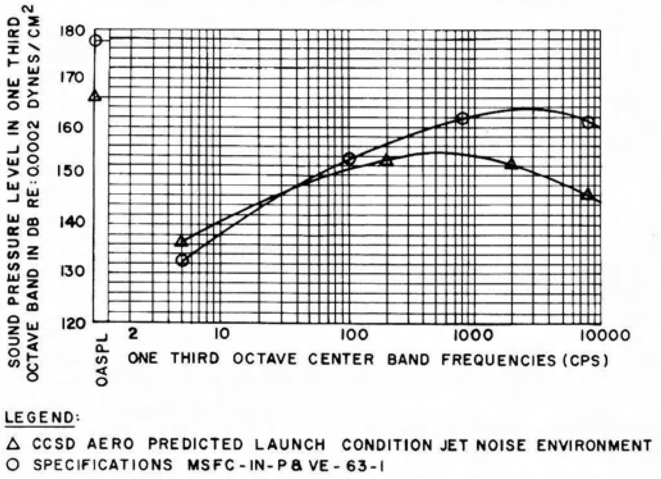


Fig. 13 - Power spectral density, Zero Stage jet noise, station 54

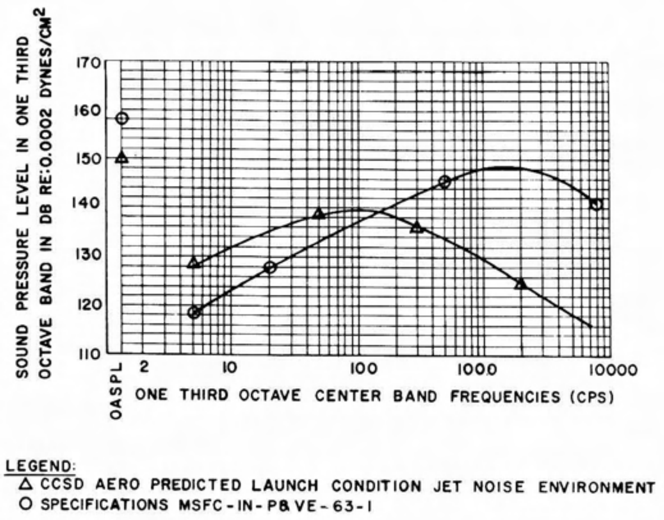


Fig. 14 - Power spectral density, Zero Stage jet noise, station 1698

AERODYNAMIC ENVIRONMENT - The aerodynamic characteristics of the Saturn IB/Zero Stage vehicle with various payloads (Apollo and Centaur) were determined and are presented in terms of total static stability in Figs. 15-18.

The total normal force gradient C_Z and center of pressure (CP/D) for the Apollo and Centaur Zero Stage configurations and center bodies alone (strap-ons ejected) are presented as a function of flight Mach number in Figs. 15 and 16. The total drag coefficients of these configurations were determined for the two flight conditions and are presented in Figs. 17 and 18. These conditions are: launch configuration with four SRM's (power on) and S-IB (power off), and ejection of the SRM's. The values of total drag are determined at $\alpha = 0$.

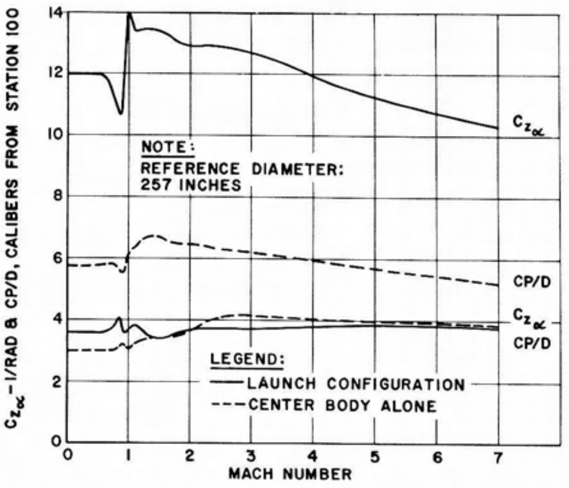


Fig. 15 - Normal force gradient and center of pressure, Centaur payload

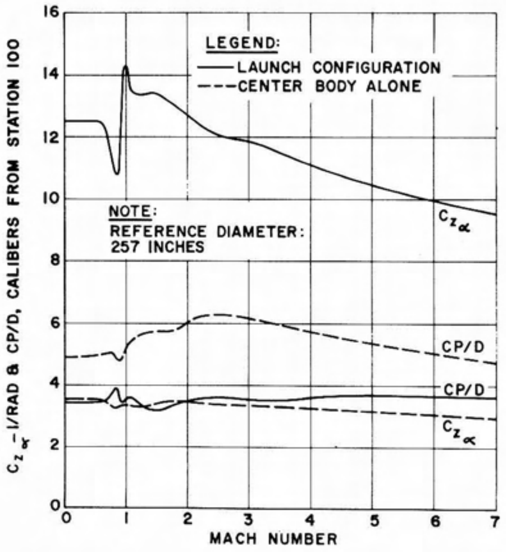


Fig. 16 - Normal force gradient and center of pressure, Apollo payload

CONTROLLABILITY

During the first phase of flight, the Zero Stage thrust vector control (TVC) system will be used to control the vehicle. After the H-1 engines on the S-IB stage are ignited and before SRM separation, control will be switched to the Saturn IB core vehicle guidance and control system.

A preliminary rigid body response study conducted on the Saturn IB/Zero Stage configuration indicated no controllability problems associated with this configuration. For this study, the vehicle response to winds in the high dynamic pressure region of flight was investigated, and no dispersions in vehicle parameters were considered. Since the area of maximum dynamic pressure will be encountered before S-IB H-1 engine ignition, vehicle control using the SRM TVC system was studied in detail.

Synthetic wind profiles were developed using the 95 percentile wind envelopes, 99 percentile wind shear envelopes, and 9 meter/sec gusts. The magnitudes of the shears and gusts were reduced by 15% as an approximation to a root-mean-square (rms) value for their individual contributions. Drift minimum control gains were used, based upon an undamped natural frequency of 0.2 cps and 70% critical damping.

Fig. 19 presents envelopes of peak values of thrust vector deflection angles (β) and angles of attack (α) in response to the synthetic winds at various times in the maximum dynamic pressure region.

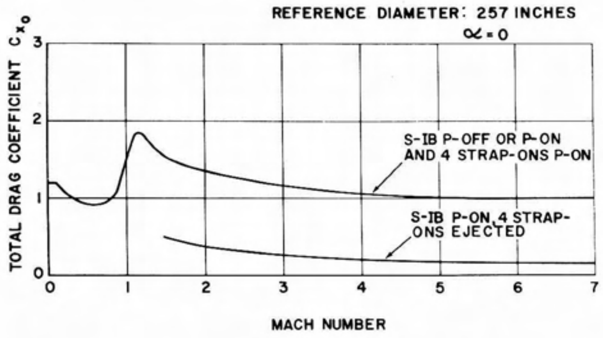


Fig. 17 - Total drag coefficient, Apollo payload

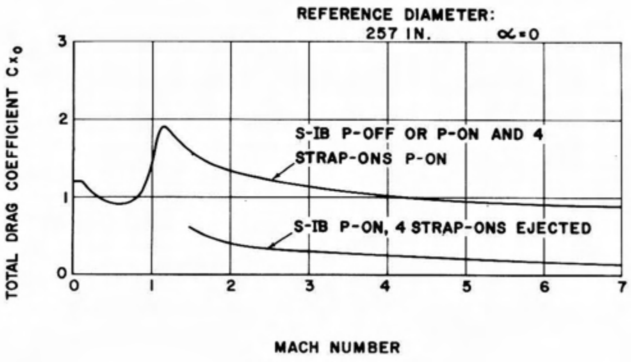


Fig. 18 - Total drag coefficient, Centaur payload

In the portion of Saturn IB/Zero Stage flight investiga the maximum thrust vector deflection angle required was 4.7 deg and the maximum angle of attack encountered was 10.7 deg, resulting in a maximum value of 0.465 psi-rd as compared to 0.63 psi-rd for the standard Saturn IB. The maximum thrust deflection requirement of 4.7 deg is well within the 8 deg capability of the SRM TVC system.

An additional study of SRM tailoff TVC requirements, performed to account for variations in SRM burn, indicated a maximum deflection requirement of 3 deg at the point where the S-IB H-1 engines have developed sufficient thrust to control the vehicle. This investigation considered the extreme case when one motor was at the 3 sigma limit of manufacturing tolerance and at a temperature of 70 F rather than 80 F. This deflection angle requirement is also well within the 8 deg capability of the solid motor TVC system.

VEHICLE EXTERNAL LOADS

Since the Saturn IB/Zero Stage vehicle will not be restrained by hold-downs on the pad (as is the case for the standard Saturn IB vehicle), the design loads for this vehicle were based on the static loading that will occur on the pad and on the overall loading that will occur in flight. The critical external loading condition for the Saturn IB/Zero Stage vehicle occurs at the point of maximum dynamic pressure (max q) in flight. A bending moment diagram is shown in Fig. 20. Note that the Saturn IB/Zero Stage flight loads are lower than those encountered by the standard Saturn IB vehicle.

SRM SEPARATION

An analysis of SRM separation was performed to determine:

1. The number and location of translation rockets required to insure successful separation of the four SRM's from the S-IB stage.
2. The motion of the SRM's from the beginning of the jettison sequence (firing of the translation rockets) until the SRM's clear the Saturn IB core vehicle.

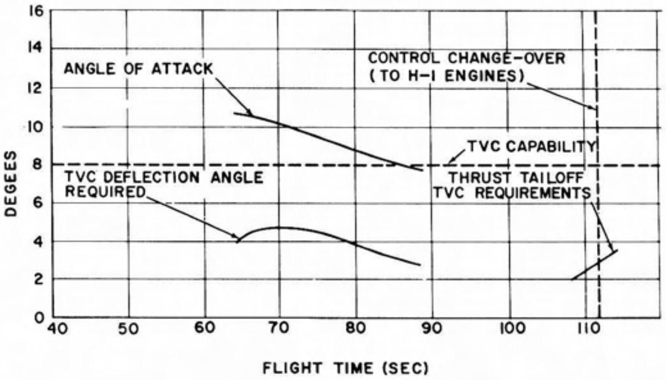


Fig. 19 - Angle of attack and TVC deflection angle requirements

The four SRM's are mounted parallel to the S-IB stage and attached, through separating joints, to the thrust structure outriggers at their bases and to the spider beam at their forward ends. The SRM's will be ignited at liftoff and separated from the S-IB stage after the H-1 engines have ignited and after the axial acceleration of the SRM's alone has become less than the axial acceleration of the Saturn IB core vehicle (approximately 114.8 sec after liftoff).

SRM separation will be accomplished by severing the support structure at the separation joints with explosive bolts and firing the translation motors mounted on the forward and aft ends of the SRM's. The path followed by the SRM's after separation is essentially one of controlled free fall, with the translation motors insuring correct separation motion and preventing interference between the SRM's and the S-IB stage.

The translation motors are mounted so that their exhaust plumes will not impinge on the surface of the S-IB stage when they are fired. This is accomplished by mounting the translation motors in pairs, one on each side of the centerline of the SRM and symmetrically about the centerline. This type of mounting also minimizes spin during separation.

Three motion analysis ground rules were established for this analysis:

1. The translation motors must produce the required thrust for a minimum of 3 sec (maximum time required to complete separation).
2. Linear displacement of the center of gravity of each SRM, away from the S-IB stage, must be 610 in. This linear displacement will preclude any possibility of interference between each SRM and the S-IB stage.
3. Angular displacement of the longitudinal axis of each SRM, away from the S-IB stage, must be 30 deg. This angular displacement will satisfy the requirement that the forward end of each SRM move away from the S-IB stage faster than the aft end (not so fast, however, that the SRM's will cartwheel).

The analysis led to the following conclusions. The translation motor thrust necessary for correct separation of the SRM (disregarding aerodynamic forces) is 13,200 lb at the forward translation motor attach point and 24,600 lb at the aft translation attach point. The larger thrust of the aft translation motors is required to offset the radial vector of the SRM tailoff thrust. The type of translation motor used for this is Unit 306, Rocket Motor Manual, SPIA/MI, Vol. II. There will be two sets of translation motors mounted on the forward end of the SRM and four sets mounted on the aft end (a total of 12 motors per SRM), providing 11,600 and 28,800 lb of radial thrust, respectively.

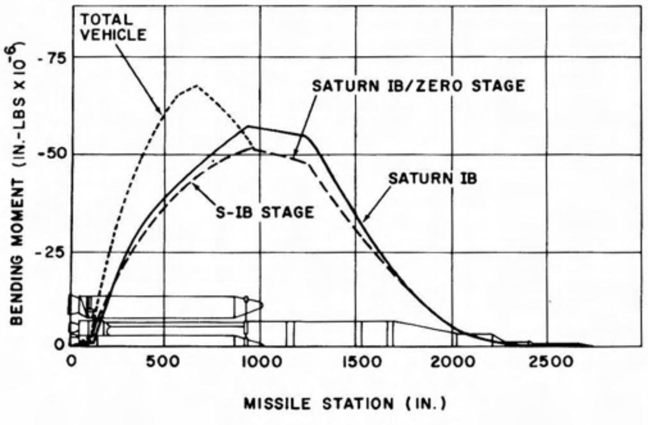


Fig. 20 - Bending moment diagram

Using the prescribed number of rockets, the separation sequence will be completed in 2.6 sec. At the end of this period, SRM axial displacement will be 503 in., yaw plane displacement will be 425 in., pitch plane displacement will be 171 in., and total SRM angle of attack will be 22 deg, 0 minutes.

Fig. 21 illustrates the movement of one SRM away from the S-IB core vehicle during separation. The Saturn IB/Zero Stage vehicle will be executing a gravity turn as the SRM's separate. Since the aerodynamic forces under these conditions are low compared to applied forces, it was assumed for this analysis that the launch vehicle angle of attack was zero. This simplifies the analysis in that the motions of all SRM's are identical.

VEHICLE WEIGHTS AND MASS CHARACTERISTICS

Table 2 is a comparative listing of Saturn IB/Zero Stage vehicle weights and baseline Saturn IB weights. The table shows an increase in payload capability of 38,600 lb and a total vehicle weight increase of 2,105,440 lb of which 2,066,164 lb represent the four SRM's.

The dry stage weight increases to the basic S-IB stage are itemized in Table 3. Removing the fins and engine skirts reduced the weight of the stage by 4900 lb. Incorporating this weight reduction, the total weight increase to the dry S-IB stage is 3544 lb, which includes 2009 lb of SRM attachment hardware.

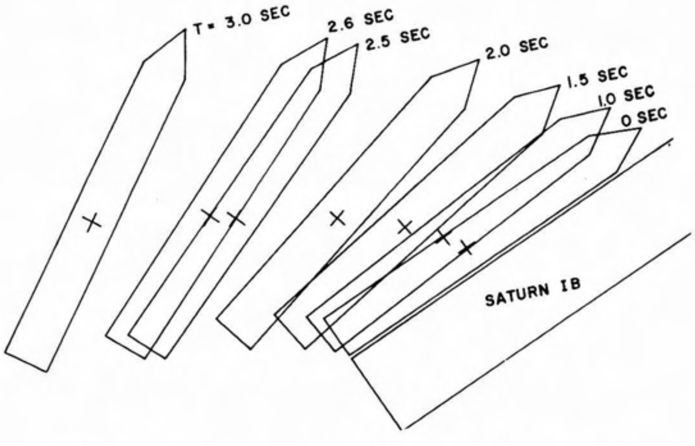


Fig. 21 - Saturn IB/Zero Stage separation sequence

VEHICLE STAGE MODIFICATIONS

GENERAL - The discussion that follows covers the hardware modifications required to the stages of the standard Saturn IB vehicle to arrive at the Saturn IB/Zero Stage launch vehicle. Although the majority of significant changes must be made on the S-IB stage, there appear to be no major engineering problems involved.

The standard United Technology Center UA-1205 SRM's will require some structural modification in the aft skirt and nose cone areas, and the translation motor arrangement will be modified for the Saturn IB/Zero Stage application. Some electrical modifications will also be required to align the control axes of the SRM TVC system with the flight axes of the Saturn IB core vehicle.

Minor structural changes will be required to the interstage (S-IB/S-IVB). The S-IVB stage and Instrument Unit used for the Saturn IB/Zero Stage application have bee designed to reflect Saturn V loading and performance requirements and will be used unchanged.

S-IB STAGE

Structural Design

Geometry - The four SRM's composing the Zero Stage will be attached to the S-IB stage of the core vehicle by trusses on the SRM aft skirt extending to the thrust structure of the S-IB, and by struts on the SRM nose cone extending to the spider beam of the S-IB. (All thrust and torque on each SRM is transmitted by the aft skirt truss supports; see Figs. 22 and 23). The nose struts transmit the overturning kick load and a portion of the side loads (Figs. 24 and 25).

The vehicle is supported on the launch pad by 12 pedestals, three per SRM (Fig. 26). Before Zero Stage ignition, the four innermost pedestals will be removed and the re-

Table 2 - Vehicle Weight Summary

Item	Baseline Weight, lb	Saturn IB/Zero Stage Weight, lb
Total Vehicle	[1,311,208]	[3,416,648]
Payload	35,500	74,100
Launch Escape System	8,055	8,055
Vehicle Instrument Unit	2,550	2,550
S-IVB at Ground Ignition	258,572	258,572
S-IB/S-IVB Interstage	6,701	7,350
S-IB at Ground Ignition	(999,830)	--
S-IB/Zero Stage at Ground Ignition	--	(3,066,021)
Dry Stage	85,600	89,144
Solid Rocket Motors (4), GFE	--	2,066,164
Propellant and Service Items	914,230	910,713

Table 3 - H-1 Engine Performance Parameters

Thrust (Sea Level), lb	1,600,000
Thrust (Vacuum), lb	1,793,672
Nominal Effective Gimbal Angle	4° 45'
Outboard Engine Gimbal Angle, deg	6 (nominal)
Inboard Engine Gimbal Angle, deg	3 (nominal
Maximum Gimbal Angle, deg	8
Exit Diameter, in./engine	45.068
Thrust Correction for Altitude	$T \approx 205K + (14.7 - P_A)\ 1637$
Max 3σ Vacuum Thrust Variation Between Engines, lb	2,250 (est)
I_{sp} (Sea Level), sec	263.9
I_{sp} (Vacuum), sec	295.7
Nominal Effective Cant Angle for All Engines	4° 45'

ining pedestals will support the vehicle at the SRM pedestal support fittings.

Internal Loads - Two factors greatly affect the internal loading of the Saturn IB/Zero Stage vehicle. First, the propellant and LOX tanks are full when the vehicle experiences maximum dynamic pressure, and secondly, the thrust loads are introduced at the end of the outriggers instead of the H-1 engine attach points. Since the bending moments used to analyze the vehicle are similar to those experienced by the standard Saturn IB vehicle (Fig. 20), and since the thrust loads being applied at the ends of the outriggers are similar to the rebound loads experienced by the standard Saturn IB vehicle, the combination of these loads plus the SRM kick loads result in only slightly higher internal loading.

Figs. 27-30 present a comparison of S-IB stage and S-IB/Zero Stage internal loading measured as each vehicle experiences maximum dynamic pressure during flight. This occurs at liftoff plus 64.6 sec for the Saturn IB, and at liftoff plus 80 sec for the Saturn IB/Zero Stage.

Modifications - The fuel tank fittings on the standard S-IB spider beam will be modified to accept the struts from the SRM nose cones. This modification as well as a number of other minor changes are noted in Fig. 31. These changes can be accommodated using existing forgings.

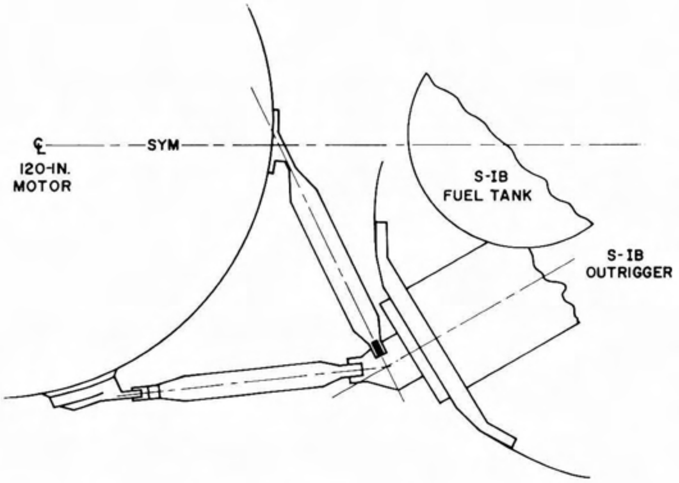


Fig. 22 - Truss geometry (view looking down)

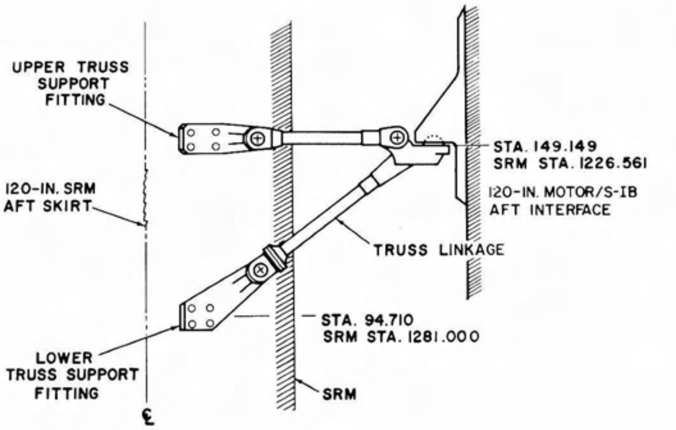


Fig. 23 - True view of truss geometry

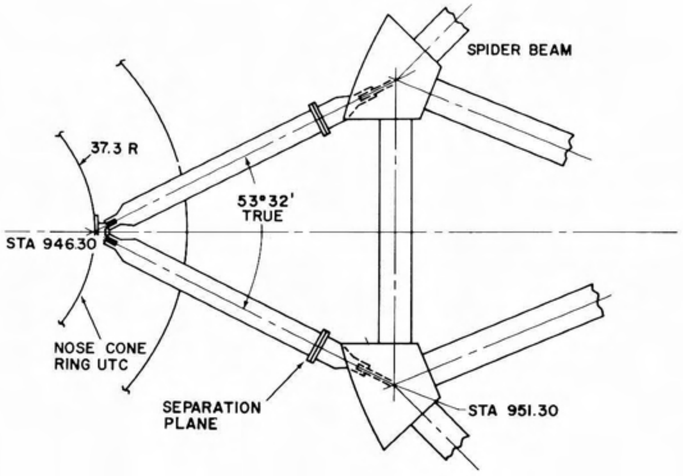


Fig. 24 - Forward support structure

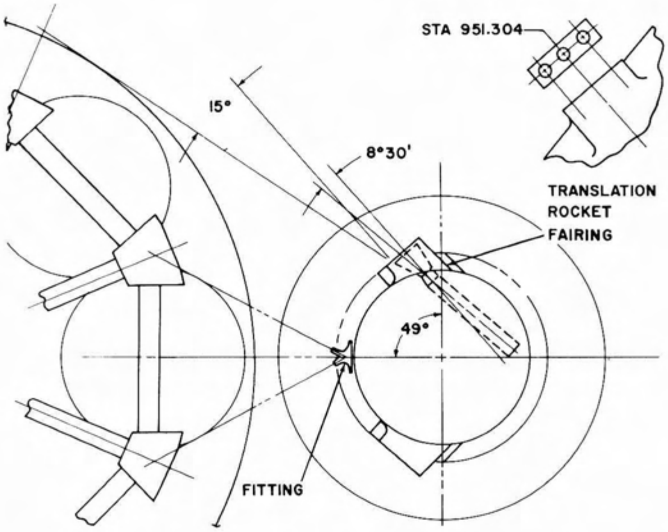


Fig. 25 - SRM forward translation motor

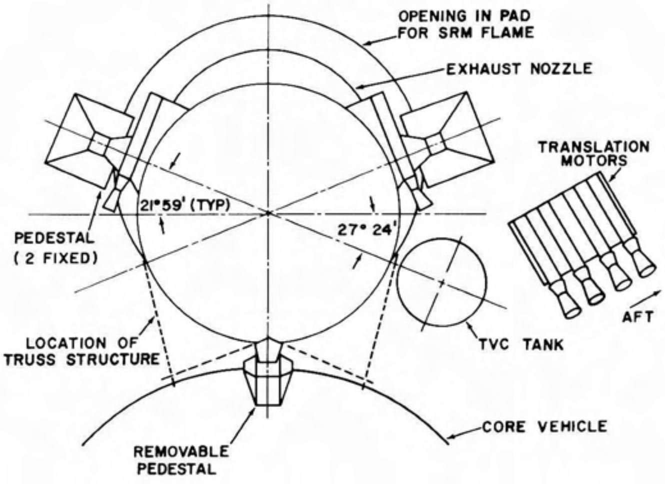
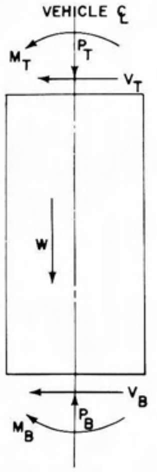


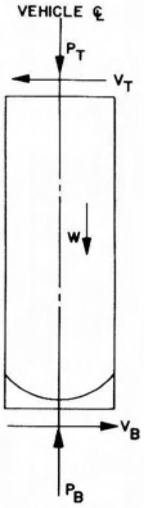
Fig. 26 - Launch configuration (view looking aft)



*	S-IB	S-IB/O
P_T	29.3	-9.7
M_T	8.1	-22.3
V_T	-9.29	18.0
P_B	278.7	335.0
M_B	-13.1	26.4
V_B	67.1	112.1

* P AND V VALUES IN LBS X 10^{-3}
M VALUES IN IN-LBS X 10^{-6}

Fig. 27 - 105 in. LOX tank loads



70" LOX TANK

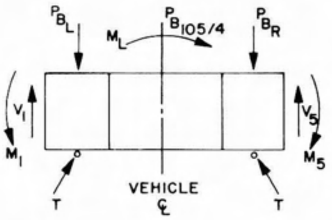
*	S-IB	S-IB/O
P_T	344.4	300.5
V_T	-5.3	-13.7
P_B	447.0	465.6
V_B	15.3	18.7

70" FUEL TANK

*	S-IB	S-IB/O
P_T	0	0
V_T	-3.5	-6.6
P_B	73.7	110.0
V_B	14.3	10.8

* P AND V VALUES IN LBS x 10^{-3}

Fig. 28 - 70 in. LOX and fuel tank loads

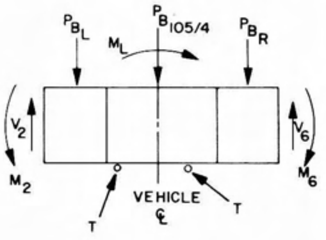


*	S-IB	S-IB/O
T	216.5	0
V_1 V_5	0	250.0
M_1 M_5	0	0
M_L**	13.1	16.6
$P_{B_{105/4}}$	69.7	83.8
P_{B_L}	126.7	125.0
P_{B_R}	238.8	290.3

* T, V, AND P VALUES IN LBS x 10^{-3}
M VALUES IN IN-LBS x 10^{-6}
** M_L = MOMENT AT STA 190.

Fig. 29 - Thrust structure (thrust support outrigger)

The skin and bulkhead thicknesses of the pressurized gion of all tanks will be increased slightly. The aft skirts of the 70 in. LOX tanks will be increased in skin thickness; the forward skirts will not be modified. Likewise, the forward and aft skirts of the 70 in. fuel tanks will not require modification (Fig. 32).



*	S-IB	S-IB/O
T	216.5	0
V_2 V_6	0	250.0
M_2 M_6	0	0
M_L	0	0
$P_{B_{105/4}}$	69.7	83.8
P_{B_L}	60.4	125.0
P_{B_R}	229.5	290.3

* T, V, AND P VALUES IN LBS. x 10^{-3}
M VALUES IN IN.-LBS. 10^{-6}

Fig. 30 - Thrust structure (fin support outrigger)

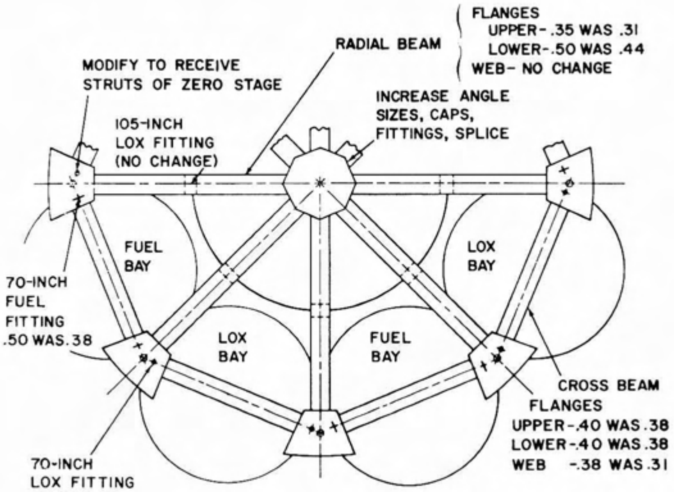


Fig. 31 - Spider beam

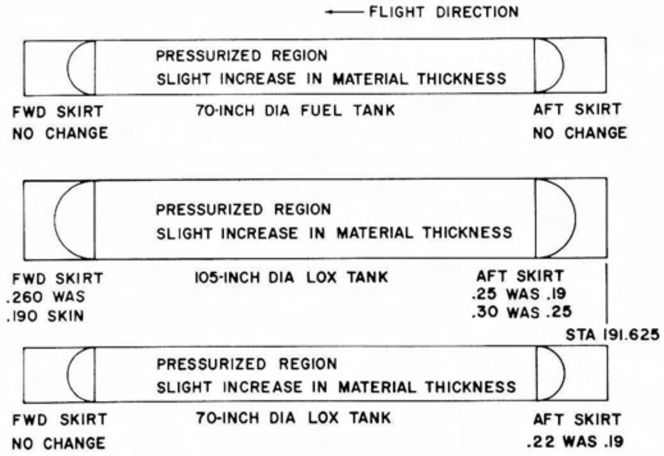


Fig. 32 - Fuel and LOX tanks

The forward and aft skirts of the 105 in. LOX tank will be increased in skin thickness. The No. 1 fuel tank sump and skirt will be redesigned to accept the repositioned fuel fill and drain port (Fig. 33).

The access door to the fuel tank bay will be relocated to the edge of the upper shroud panel (Figs. 33 and 34). Eight fittings will be added to receive the trusses of the SRM's. The fittings will be attached in the former fin positions (on each outrigger). An additional web will be near the outboard end of each outrigger to form a vertical box beam. This beam will distribute the side thrust loads from the SRM. Portions of the thrust structure will require a change in thickness (Fig. 35).

All fins and the outboard engine skirts will be removed.

Propulsion Design - The S-IB propulsion system was examined for the Zero Stage application. It was assumed that standard 205,000 lb thrust H-1 engines will be used with maximum use of existing hardware. The performance characteristics of the H-1 engine are shown in Table 3 and Fig. 36. Note that the I_{sp} used is higher than for present S-IB flights. The new value is being used by NASA and may be interpreted as anticipated improvement potential for the H-1 engines. The following sections discuss the changes required to the propulsion subsystems.

Oxidizer Pressurization System - The oxidizer pressurization system provides pressure to the LOX tanks to prevent them from buckling and insures that a sufficient net positive suction head (NPSH) is provided to the pumps. The tank top pressure requirement to satisfy pump NPSH is shown in Fig. 37. The required pressure for S-IB/Zero Stage ignition has been shown as the same as that required for standard S-IB ignition since it is possible that, because of the tolerance on thrust ignition, the acceleration at ignition could drop to 1.0 g instead of 1.2 g as indicated by analysis. The pressure required at S-IB stage burnout has been raised 2 psi over the standard S-IB burnout requirement. This will provide a margin of safety in the event temperature stratification in the LOX becomes more accentuated than for a standard S-IB flight. (This corresponds roughly to an increase of 1.25 F in the temperature of the upper layers of LOX in the tank.) A higher burnout LOX temperature is considered a possibility for the Saturn IB/Zero Stage vehicle since its

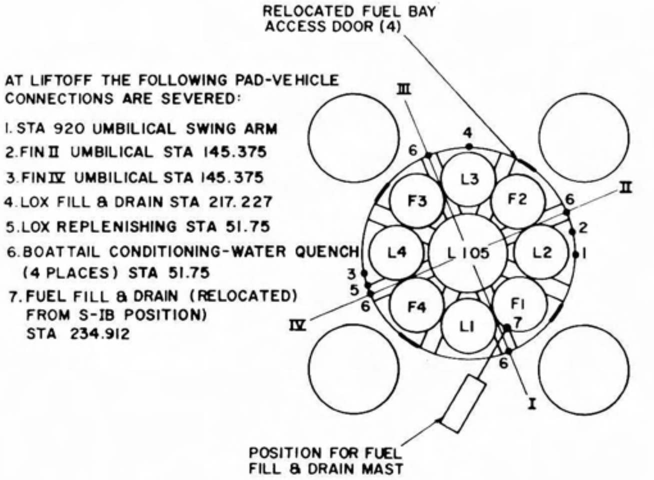


Fig. 33 - Umbilicals and access (view looking aft)

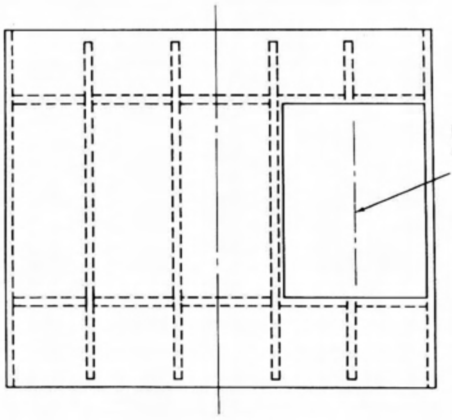


Fig. 34 - Upper shroud panel

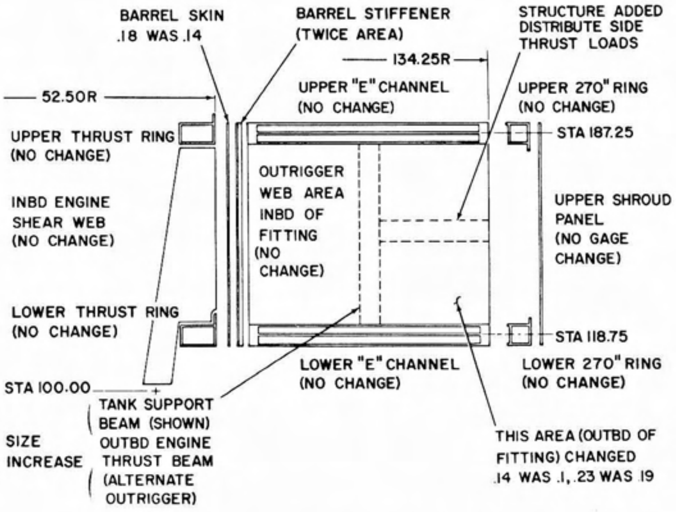


Fig. 35 - Major components of thrust structure

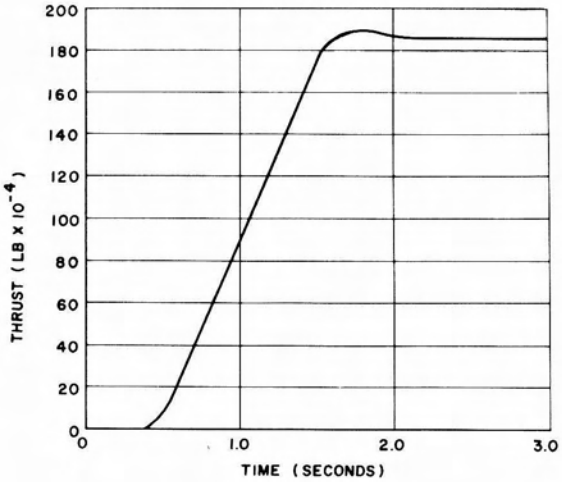


Fig. 36 - Saturn IB start transient

flight time is 107 sec longer than the standard S-IB trajectory. The warmer LOX will probably rise to the surface and result in greater temperature stratification.

The maximum and minimum LOX tank pressure profiles selected for the Saturn IB/Zero Stage application are shown in Fig. 38. These profiles were selected to satisfy pump requirements at ignition and burnout. It should be noted, however, that the selected profile will in itself dictate a change in the lower dome, and probably the upper dome. The maximum structural limit of the upper dome of a standard S-IB LOX tank at burnout (vacuum conditions) corresponds to a tank pressure of 58.8 psia, which falls below the relief band established for this application. The lower dome of a standard S-IB has a structural limit of 98 psi (free-standing), which would be exceeded shortly after S-IB ignition.

An alternative pressurization scheme is depicted in Fig. 39. The salient features of this method are an initial polytropic expansion until bulk boiling occurs, a period of bulk boiling to decrease stratification temperatures in the upper portion of the LOX, and LOX pressurization until burnout. It is anticipated that the implementation of this method could result in a weight saving of about 300-400 lb dead weight and an equivalent amount of additional usable propellant. However, the cost of testing this concept and changing the pressurization system hardware would probably outweigh the potential performance gains. (This concept could not be employed on a standard S-IB because of a structural requirement for a minimum of 45 psia.) If this alternative pressurization method were used, it would also be desirable to calibrate the engines at a lower suction pressure to avoid a performance degradation.

Fuel Tank Pressurization - The fuel tank pressurization system is tentatively assumed to be the same as used on the standard S-IB (helium blowdown system), since structural requirements alone establish the tank top pressure limits for the fuel tanks. However, if the tank structural requirements do change, the effect on the fuel pressurization system will be negligible (Fig. 40) in terms of total system weight, costs, or schedules.

Oxidizer Vent Systems - The oxidizer vent system planned for the Saturn IB/Zero Stage vehicle is the same as that

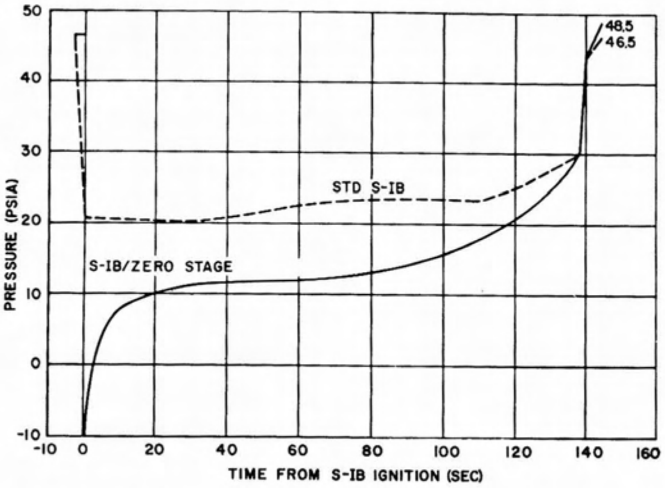


Fig. 37 - LOX tank pressure to satisfy pump NPSH

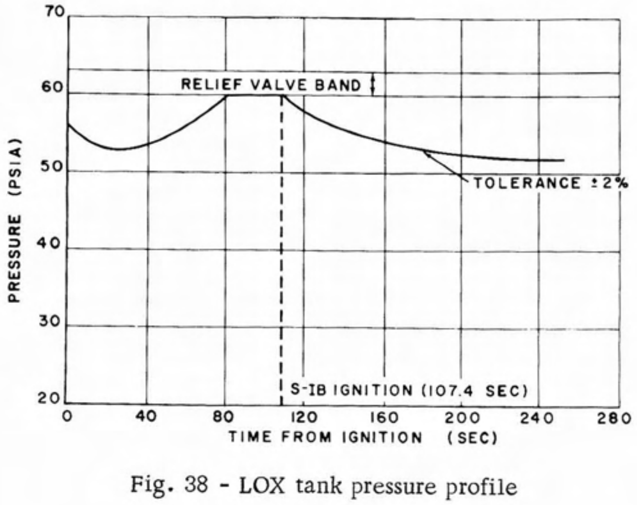


Fig. 38 - LOX tank pressure profile

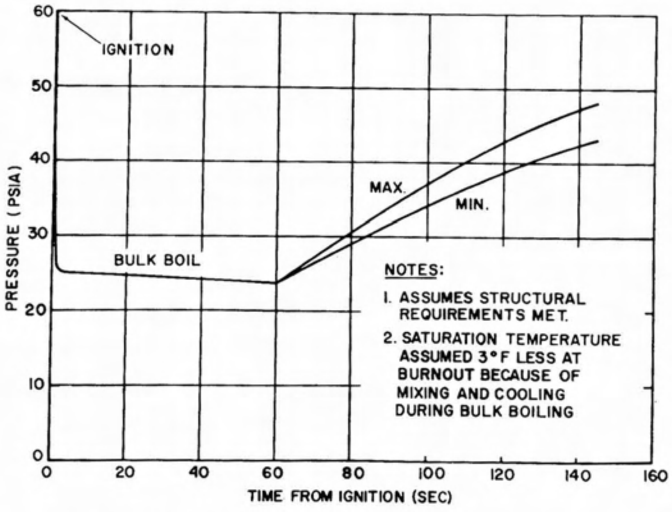


Fig. 39 - Alternate pressurization scheme

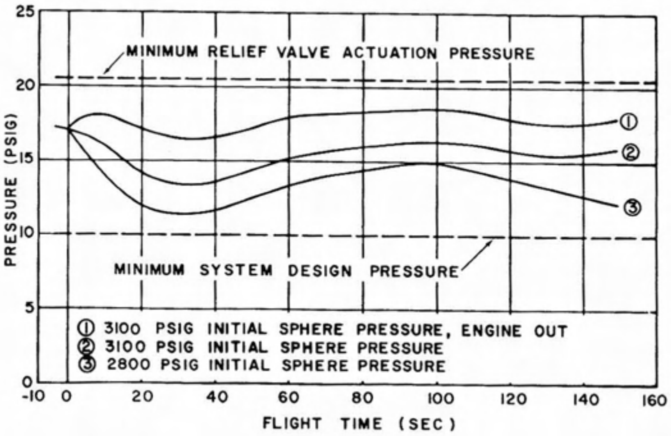


Fig. 40 - Performance of blowdown fuel tank pressurization system

sed on the standard S-IB except that the main vent valves will only operate in the mechanical mode (the pressure switch mode of operation will be deleted).

Fuel Vent Systems - No change in the standard S-IB fuel vent system is planned.

Auxiliary Pressurization Systems - An additional small supply of helium is required to hold the tank top pressure in the LOX tank at a minimum of 47 psi for 2 sec during H-1 engine ignition. The helium source is a facility item on a standard S-IB but will have to be an airborne item for this application. This subsystem will consist of a helium supply, an on-off type valve (probably a 1 in. solenoid valve), and associated plumbing. Operation of the system will require the opening of the shutoff valve at S-IB ignition, thus allowing the helium to expand into the main tanks until the supply is depleted. The system will be sized for a flow rate of 2.5 lb/sec (average) for 2 sec. The additional weight of this system is estimated at 100 lb. As this is a new system, a test program will be required to evaluate its performance. This system, however, should not be pacing or major expense item in the program.

To satisfy engine purge requirements before S-IB ignition, it will be necessary to add an additional nitrogen supply (55 lb of nitrogen) to the S-IB stage. The system will include four 15 in. 3000 psi nitrogen spheres, a 3/4 in. pressure regulator, a bypass bleed around the regulator, and associated plumbing and fittings. The system will be capable of supplying approximately five standard cubic feet per minute (scfm), via the bypass bleed, during all of Zero Stage burn and 325 scfm for 15 sec before S-IB ignition. Total system dry weight is estimated at 100 lb.

H-1 Engine Ignition and Operation - Altitude start of the H-1 engines is not expected to pose any difficulties. Vacuum tests indicated that there will be a pressure buildup to 7 psia in the combustion chamber, and should provide sufficient chamber pressure to successfully start at altitude.

One of the basic requirements for starting the H-1 engine is that the initial LOX temperature must be -275 F or lower. This is presently achieved on the standard S-IB by helium bubbling, which cools the propellant in the suction lines to about -285 F. Indications are that the rate temperature rise, after helium bubbling has been terminated, is slow enough to negate the requirement for an onboard helium supply for bubbling during Zero Stage burn. However, it may be necessary to pre-cool the helium to insure a sufficiently low LOX suction temperature at liftoff.

Because of the particular acceleration profile and tank top pressures used with the Saturn IB/Zero Stage vehicle, there will be a slight change in the propellant mixture ratio and thrust. This change will be reflected in the propellant loading requirements and performance; however, an evaluation of this effect was not warranted at this point in the study. In general, the sea level thrust may be expected to be slightly higher than for a standard S-IB.

Ground power (110 v) is presently used on the standard Saturn IB vehicle to power heaters on various engine components. This procedure will be used on the Saturn IB/Zero Stage vehicle.

The nozzle exits of the H-1 engines will be protected by cork plugs during Zero Stage burn. These plugs will be ejected at H-1 ignition.

H-1 Engine Locking During Zero Stage Burn - On the standard Saturn S-IB, the outboard H-1 engines are positioned by means of hydraulic actuators powered by pumps driven off the outboard H-1 engines. Since, with the Saturn IB/Zero Stage vehicle, the H-1 engines will not be ignited during Zero Stage burn, a holding device will be required to keep the engines from floating. One means of holding or locking the engines is to pressurize the hydraulic system prior to launch and depend on the accumulator to hold pressure. A new or redesigned accumulator will probably be required for this application. A second method for holding the H-1 engines is through the use of a mechanical locking device which would be released after H-1 engine ignition.

Electrical Design - Installation of the four SRM's will require integration of the electrical systems for each of the motors with the S-IB stage electrical system. A preliminary review of the S-IB electrical system indicated that it can be modified to provide the control functions and telemetry channel capacity necessary to satisfy SRM requirements.

Electrical Interface - An elementary block diagram for a single SRM interconnected with the S-IB stage is shown in Fig. 41. Two cross-over bundles from the S-IB stage to each SRM will be required (Fig. 42).

Stage Networks - The S-IB stage switch selector will be modified to accommodate the control functions of the SRM's. Additional measurement signal distribution networks, measuring racks, telemetry equipment, and associated cabling will be required to provide for Zero Stage instrumentation requirements. Additional networks for H-1 engine altitude start functions, such as LOX tank pressurization, engine purging, thermal conditioning, and ejection of closures and insulators will be required. Preliminary studies indicate sufficient onboard power exists for altitude start of the H-1

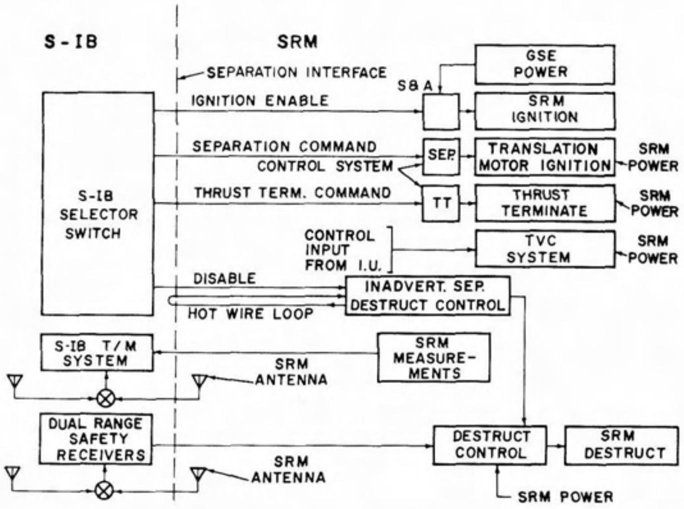


Fig. 41 - S-IB stage/SRM electrical interface

engine. New networks for interface with the Zero Stage will also be required.

H-1 Altitude Ignition - The basic H-1 engine will be modified to provide for altitude ignition and gimbal locking during Zero Stage burn. An additional electrical control system will be required for inflight operation. These signals will be furnished by the S-IB switch selector.

Telemetry System - The same basic telemetry system used for the R & D Saturn IB vehicle will be used on the Saturn IB/Zero Stage R & D vehicles. The expandable basic design will permit the incorporation of additional telemetry subassemblies to accommodate SRM measurement requirements. Additional modules will be added to the S-IB stage to accommodate approximately 150 pulse code modulator (PCM) encoder channels, 4 single side-band (SSB) channels, and 12 wide-band channels for each SRM. Additional antennas will be required due to the radiation interference caused by the SRM's. A preliminary review indicates that these alternate SRM antennas will be located in the SRM nose cones, and that the TM output signal will be switched to the S-IB antennas at separation.

Measurements - Provisions will be made aboard the S-IB stage to accept the performance required for SRM monitoring. These measurements will be in addition to those specified by the basic Instrumentation Program and Components list for the operational SA-205. SRM instrumentation signal conditioning will be accomplished within each SRM, and the signals will be relayed through the staging disconnects to the S-IB measurement systems.

Flight Control System - A three-stage control system Instrument Unit (SA-IU-501 type) will be required for the Saturn IB/Zero Stage vehicle. This unit will be programmed to provide control signals for the SRM functions through the S-IB stage selector switch. Both the Zero Stage and S-IB stage will be flown with "open-loop" guidance. Deviations in trajectory will be calculated, stored, and later corrected during the S-IVB stage burn. Additional control filters will be designed to meet the controllability requirements and present suitable signal transfer characteristics to the SRM TVC system.

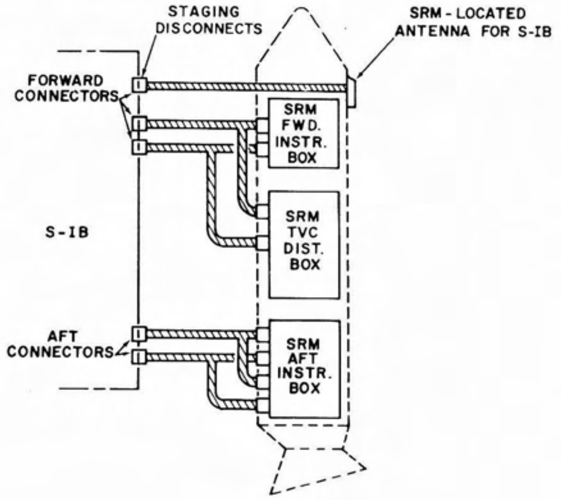


Fig. 42 - S-IB stage/SRM electrical cabling

120 IN. SOLID ROCKET MOTORS - Most of the design changes required to adapt the SRM's to the S-IB stage will be associated with the attach hardware. A minimum of solid rocket motor structural and systems modifications will be required.

Aft Skirt - Redesign of the SRM aft skirt will include minor modification of the skirt structure, removal of existing and the addition of new retrorockets and attachment hardware, and a redesign of the aft truss attach linkages (refer to Figs. 22, 23, and 26).

Nose Cone - Redesign of the SRM nose cone will include removal of the existing retrorocket housing and provision of new retrorockets and hardware, and the addition of fittings to receive struts from the S-IB spider beam (refer to Figs. 24 and 25).

Systems - A variety of system modifications, primarily electrical in nature, will be made to the following SRM systems: ignition, separation, destruct, power TVC, and EDS.

Solid Rocket Motor Ignition System - The SRM ignition system used on the Titan III-C vehicle was found acceptable for use in the Saturn IB/Zero Stage application. This system, consisting of a safe and arm (S & A) device and igniter assembly containing two redundant squibs, will satisfactorily ignite the five-segment SRM's. The SRM ignition system uses 28 v d-c squibs, whereas the Saturn IB ordnance system uses a 2300 v d-c Exploding Bridgewire (EBW) squibs. This former system has been qualified for a man-rated vehicle on the Titan III-C program and minor requalification will be necessary for Saturn IB/Zero Stage application.

The SRM igniter assembly consists of an intermediate initiation motor whose exhaust products are directed into the main propellant port. The ignition fire command will originate from the ground support equipment.

Solid Rocket Motor Separation System - Following a normal separation command, the separating devices at the forward and aft ends of each SRM will breach the attachment struts. Dual frangible nuts, detonated by an EBW system on the S-IB stage, will break the aft ball joint staging bolts and the forward outrigger staging bolts. A prior destruct-disable command from the S-IB selector switch will prevent the activation of the SRM inadvertent separation destruct system. Ignition of the 12 translation motors on each SRM (eight aft and four forward) will provide sufficient force to drive the SRM clear of the vehicle. The translation motors will be ignited by the SRM separation control system which is activated by a command from the S-IB selector switch.

Each SRM will contain an inadvertent separation control system which will be activated in the event of a structural break from the core vehicle. Should the SRM be inadvertently separated from the S-IB stage, a "hot wire" connection to the S-IB will initiate thrust termination and SRM destruct.

Solid Rocket Motor Destruct System - The standard SRM destruct system will not be changed for the Saturn IB/Zero ge application and will operate independently from, but on the command of, the Saturn IB range safety system. On command, an electrical signal from the existing dual-range safety receivers on board the S-IB stage will initiate the destruct command on the four SRM's. Unlike the liquid S-IB stage, the SRM's contain no engine shutoff provisions. Successful flight termination will be accomplished by severing two ports on the head, rendering the SRM's nonpropulsive (Fig. 43).

Solid Rocket Motor Power Requirements - Each SRM contains dual 28 v d-c batteries that provide local power for all its electrical requirements. The only components in the SRM's requiring power from the S-IB stage are the S & A devices of the thrust termination, destruct, and ignition systems.

The electrical requirements transferred at each SRM are:

1. Safe and arm (3 amp at 24-31 v d-c).
2. Safe and arm monitor (200 ma).
3. Firing current to each squib and detonator (6 amp at 24-31 v d-c).

Solid Rocket Motor TVC System - Thrust vectoring the SRM's will be accomplished by the individual TVC systems attached to the SRM exhaust nozzles (Fig. 44). Each SRM contains provisions for 24 electrically controlled valves uniformly spaced around the periphery of the nozzle. For the Saturn IB/Zero Stage application, preliminary valve assignment to meet TVC requirements involve five valves located in each of four quadrants, with four dummy positions (Fig. 45).

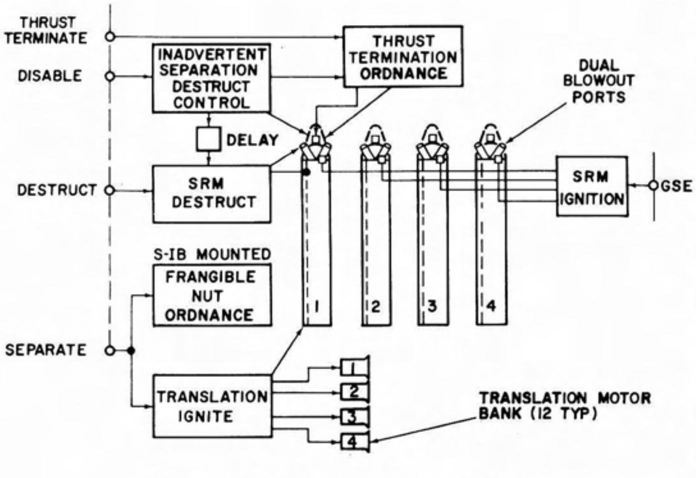


Fig. 43 - SRM ordnance system

Nitrogen tetroxide fluid is fed from the manifold to the injector valves being modulated from zero position to full open by signals from the TVC system.

Emergency Detection System - The Saturn IB vehicle emergency detection system (EDS) will be modified to accept the additional requirements of the four SRM's. The vehicle EDS will contain additional monitors to sense the operational status of each SRM. These sensors will monitor the following functions:

1. SRM separation indication.
2. SRM translation motor.
3. Inadvertent separation indication.
4. SRM power supply system.
5. Flight instrumentation.
6. Case head pressure.
7. Aft closure compartment temperature.

S-IVB STAGE - Loading in the S-IVB stage of the Saturn IB/Zero Stage vehicle will be slightly higher than in the Saturn IB application, but not as high as S-IVB loads in the Saturn V application. Therefore, modification of the S-IVB stage structure will not be required. Likewise, since all S-IVB systems will operate as they do in the Saturn IB, no systems modifications will be required.

INSTRUMENT UNIT - The Instrument Unit (IU) used for the Saturn IB/Zero Stage vehicle will be a standard design IU as used on vehicles SA-206 and subsequent. The IU will perform the following vehicle functions: navigation, guidance, and control (Fig. 46). The following system modifications will be required:

1. Guidance Computer - Either the Saturn IB or Saturn V type guidance computer will be modified for this configuration, incorporating the proper filters and gains.

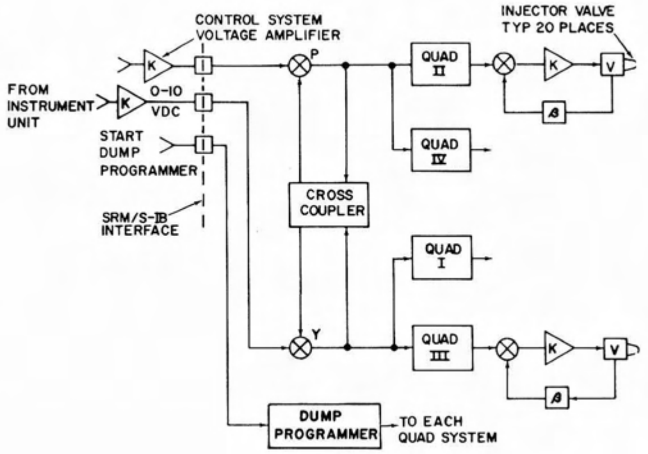


Fig. 44 - TVC electrical system (simplified)

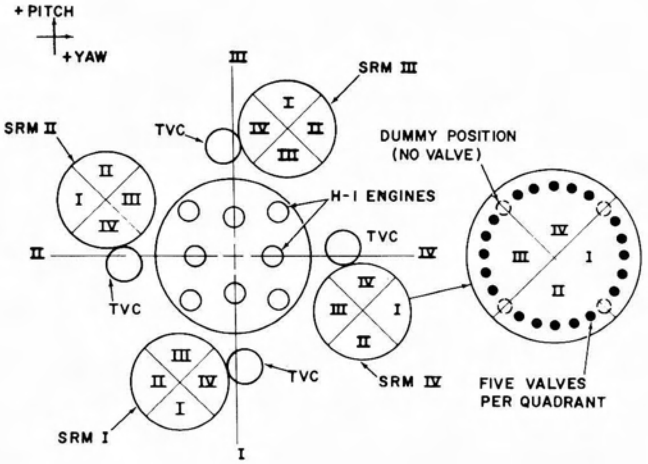


Fig. 45 - SRM quadrants for thrust vector control

2. Switch Selector - The switch selector will not be modified for the Saturn IB/Zero Stage application. Stage code 12 will be used to direct control signals to the solid motor switch selector (SMSS) located in the S-IB instrumentation compartment.

3. Launch Vehicle Digital Computer (LVDC) - The LVDC will be used for H-1 ignition and SRM staging.

The program for the LVDC will be loaded into the memory cells. A data flow diagram for this program is shown in Fig. 47. At liftoff, the LVDC timer will activate the Zero Stage Attitude Ignition program. The computer will then monitor longitudinal accelerations from the stabilized platform and upon a negative value of da/dt, the H-1 ignition sequence will be performed.

Upon completion of H-1 ignition, the thrusts will be monitored by the EDS system. If no malfunction in H-1 engine thrust buildup is indicated, SRM staging will then be performed.

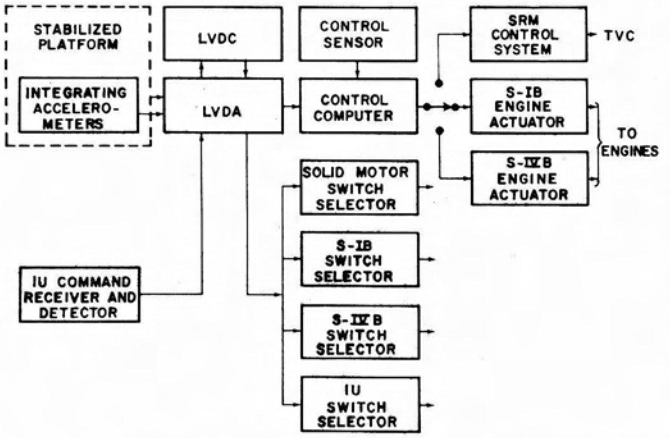


Fig. 46 - Saturn IB/Zero Stage navigation, guidance, and control system

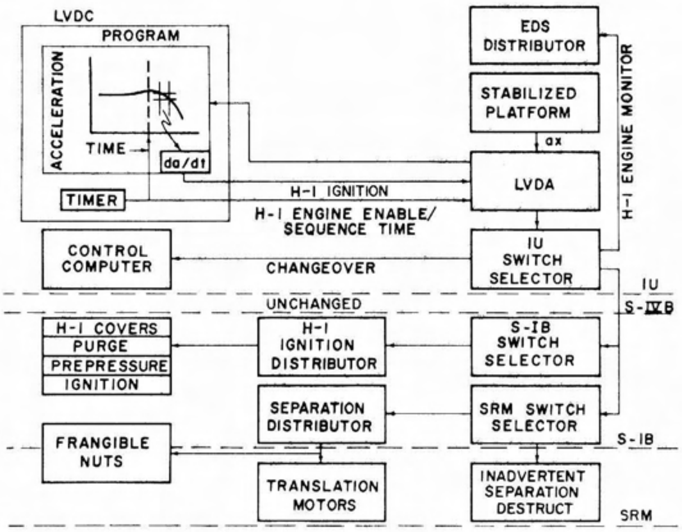


Fig. 47 - H-1 engine ignition and SRM staging system

CENTAUR - The Centaur application for the Saturn Zero Stage will be the same as that for the Saturn IB. In gration of the Centaur stage with the Saturn IB is covered in Chrysler Corp. Space Division Technical Report TR-AE-65-5, "Saturn IB/Centaur Integration."

MODIFIED SERVICE MODULE - Use of the Apollo Service Module (SM) as an upper stage or "kick" stage for the Saturn IB/Zero Stage vehicle has been investigated in detail with the following results. The Service Module can be used essentially "as is"; its performance can be optimized, however, by making certain modifications. SM propellant capacity can be increased by using space now allocated for experiments for propellant storage. Likewise, the fuel cells now installed on the Service Module to support crewmen in the Command Module can be deleted, making available even further propellant storage capacity. Even though a small part of this space will be required for communication and G & C equipment, the additional propellant storage space made available will greatly extend the operational life of the Apollo Service Module.

LAUNCH OPERATION AND PAD BUILDUP PROCEDURE

Fig. 48 summarizes the launch operation for the Saturn IB/Zero Stage vehicle, based on an analysis of the use of Kennedy Space Center launch complex 37B by CCSD.

The SRM components for a particular Saturn IB/Zero Stage vehicle will be delivered to the Titan III-C facility at Cape Kennedy by railway car. There they will be inspected and subassembled in accordance with current Titan III-C SRM procedures. The SRM components will then be transported to the launch complex 37B by semitrailers when required for SRM buildup.

The Saturn IB core vehicle will be erected and checked out in accordance with existing operating procedures, to the extent possible considering the minor modifications which will be made to the vehicle. The S-IB stage will be erected on four of the existing support and hold-down arms; the re-

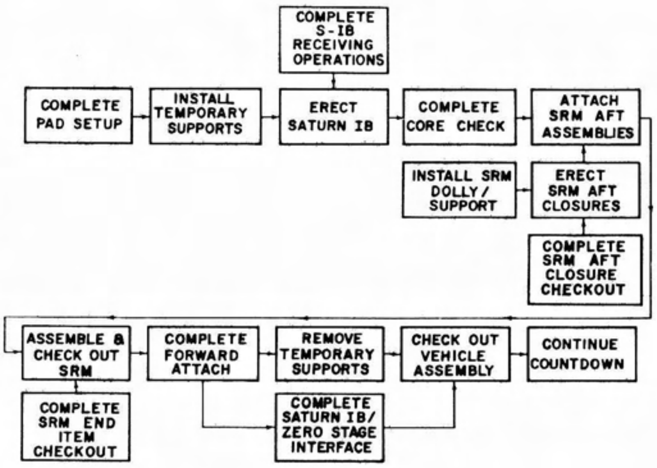


Fig. 48 - Saturn IB/Zero Stage launch operations

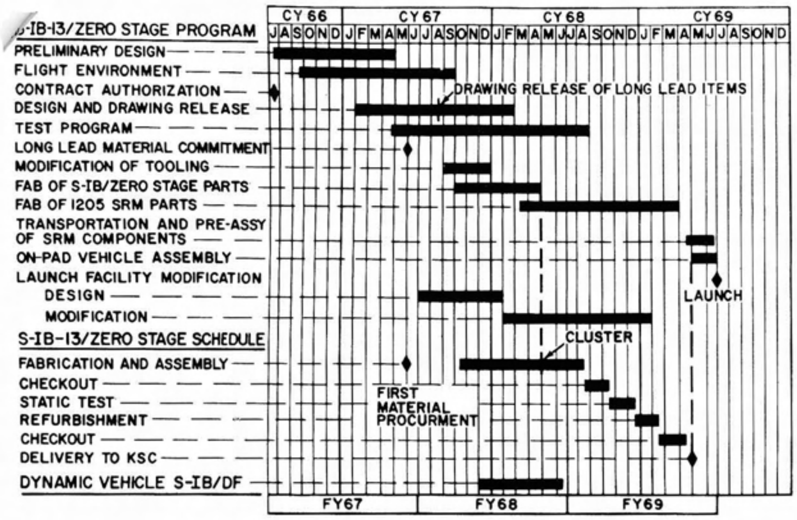


Fig. 49 - Development plan

maining core vehicle stages and payload will then be erected and checked out through electrical mating of the spacecraft with the core vehicle. Each of the four SRM aft closures will then be mounted on three supports (two permanent and one temporary), using the service structure crane and temporary scaffolding. The jacks built into the SRM supports will be lowered to allow sufficient clearance to position the SRM on core ball joints underneath the mating socket on the vehicle. The SRM supports will then be elevated to align the SRM aft closure assembly and provide a tight fit at the SRM core ball joint interface. Stacking of the SRM's will then begin.

Following completion of SRM erection, the scaffolding will be removed and the service structure will be positioned around the vehicle to accomplish checkout of the SRM's. When SRM checkout is completed, mechanical and electrical mating of the SRM's and Saturn IB will be accomplished. The forward tie rods of all four SRM's will be attached to the core vehicle. The ball joint bolts will then be tightened to a snug fit and the temporary SRM supports removed. The core vehicle hold-down arms will be retracted and the compression supports adjusted so that the core vehicle weight is supported by the four SRM's through the attachment structure. The ball joint bolts will then be tightened to the required torque level.

DEVELOPMENT PLAN

The development plan for the S-IB/Zero Stage shown in Fig. 49 reflects the major events required to incorporate the stage into the current manufacturing schedule with an effectivity of S-IB-19. The pacing items in this schedule are the design and fabrication of the propellant tanks and the thrust structure.

Based on a contract authorization for preliminary design in July 1966 and a go-ahead for engineering production design in February 1967, delivery of the S-IB/Zero Stage 219 would occur in April 1969, with the launch of Saturn IB/Zero Stage 219 scheduled for July 1969. A boilerplate stage could be available four to six months earlier.

CONCLUSION

The addition of four 120 in. solid rocket motors to the Saturn IB vehicle as a "Zero Stage" offers an attractive and economical method of upgrading vehicle performance. The Saturn IB payload capability of 40,000 lb in low earth orbit can be increased to over 78,000 lb by implementing this concept. The addition of the Zero Stage will also provide the Saturn IB with an escape payload capability of 11,500 lb, or 22,000 lb with a Centaur stage. Similarly, the short lead time necessary to redesign the S-IB stage would permit active, deep space exploration in the early 1970's. Boilerplate research and development flights would further reduce redesign lead time.

The wide range of missions made available by the Saturn IB/Zero Stage suggests that this highly versatile vehicle may well become a standard launch vehicle for missions extending from the basic 40,000 lb of payload in low earth orbit, through synchronous orbit, lunar, and planetary missions, and finally to high-energy solar probe missions.

ACKNOWLEDGMENT

The authors of this paper would like to thank O. J. Thorsen of the Rocketdyne Division of North American Aviation, Inc., and J. H. Dodd and R. M. Kluzek of the United Technology Center Division of the United Aircraft Corp.

660452

Reusability — The Next Major Launch Vehicle Development

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NATIONAL SPACE program launch vehicle planners are faced with the problem of identifying the stable of launch vehicles required to assure adequate launch system capability for a continuing and vigorous space program. This development must be accomplished within certain economic constraints. The gamut of alternatives to be considered range from the utilization of existing expendable vehicles to development of completely new launch systems. Included in the alternatives are many possibilities for increasing launch capabilities, such as adding zero or parallel stage rocket motors similar to the Titan III-C, increasing thrust and propellant similar to S-IB, modifying existing boosters for reusability, and developing new expandable for reusable vehicles.

The current stable of launch vehicles has a launch to 100 mile orbit capability of from 300 lb for Blue Scout to in excess of 250,000 lb for Saturn V. If a major launch vehicle development is to be implemented, it must satisfy some new operational or payload requirement, or must increase launch system efficiency in terms of reliability or overall program economy. Reusability as compared to the other alternatives must show a significant advantage in one or more of these areas if it is to be considered for development.

The purpose of this paper is to identify gross launch vehicle requirements to support the space program for the next two decades; explore launch vehicle economic and operational development alternatives; and indicate the place reusability has in a logical development program.

LAUNCH VEHICLE REQUIREMENTS

The current national goal in space is to land a man on the moon and return him safely to earth in this decade. A large number of additional programs such as those indicated in Table 1 have been proposed. If the United States were to undertake to accomplish these programs, it would require a space budget of from 8-10 billion dollars per year -- well above the current experience of around 5.25 billion dollars per year. Even an annual increase equal to the gross national product growth, or about 5% per year, would not approach the funding required to implement the proposed missions. Fig. 1 indicates a space program that can be accomplished with the latter level of funding and the payload in terms of material required to accomplish the program.

Variations of this program (substitution of alternate missions) within the budget limits result in essentially the same payload requirements. From this figure, it is apparent that

ABSTRACT

This paper addresses itself to the question of what direction launch vehicle development activity should proceed after Saturn V/Apollo.

Development problems are identified, and a logical launch system development program is suggested. The space programs for the next two decades are identified; launch vehicle economic and operational alternatives are explored; and the place reusability may have in future launch vehicle development is examined in particular.

Five basic vehicle alternatives are considered: existing expendable boosters, existing expendable boosters uprated, new expendable boosters, expendable boosters modified for recovery and reuse, and new reusable vehicles.