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13 November 1959 Report DSP-TM-13-59

SATURN SYSTEM STUDY II

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13 November 1959 Report DSP-TM-13-59

SATURN SYSTEM STUDY II

DEVELOPMENT OPERATIONS DIVISION
ARMY BALLISTIC MISSILE AGENCY
REDSTONE ARSENAL, ALABAMA

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SATURN SYSTEM STUDY II

CHAPTER I: (C) INTRODUCTION

Design studies of a 1.5 million pound thrust, three-stage space carrier vehicle were initiated at ABMA in April 1957 as an inhouse effort based on four E-1 engines in the first stage booster. Detailed preliminary designs and performance studies were carried out during the years 1957 and 1958. The study program was redirected in July 1958 based upon a recommendation made by D. A. Young and R. B. Canright of ARPA to replace the four E-1 engines in the booster by available JUPITER engines; and, thus, eliminating a $60 million development program of an advanced engine (NAA E-1). This also would allow an early booster development initiation. Further efforts by ARPA resulted in ARPA Order 14-59 authorizing ABMA to initiate the design and development of a first stage booster capable of producing 1.5 million pound thrust at sea level. The immediate goal was to demonstrate the feasibility of operating an eight engine cluster of this size.

The original order was amended on 21 November 1958 to include the fabrication and launching of four SATURN boosters. The first two flyable vehicles would be booster only, with dummy upper stages, and the remaining two would be flown with an unsophisticated second stage providing a nominal orbital capability.

On 18 December 1958 a SATURN System Study was initiated by an amendment to ARPA Order 14-59 with emphasis on the selection of upper stages for the 1.5 million pound thrust booster. The required report was completed and forwarded to ARPA on 13 March 1959 (Ref. 1). An evaluation committee, chaired by ARPA and consisting of DOD and NASA members, made a recommendation to proceed with a development plan based on a modified TITAN first stage as SATURN second stage and a modified CENTAUR as SATURN third stage. This directive was received at ABMA on 20 May 1959 with the request to submit a development and funding program.

On 13 February 1959, AOMC submitted to ARPA for approval a plan increasing the scope of the SATURN program. This plan outlined a 16 flight program resulting in an operational SATURN vehicle by 1963. The funding required for this program through FY 1961 totaled approximately $300 million. Supplement Number 2 of the development and funding plan was submitted to ARPA on 13 August 1959 (Ref. 2).

This plan was, as requested, limited to the first four flight vehicles and included funding breakdowns for FY 1960 only.

In view of the possible development of a TITAN C booster (four engine, 160-inch diameter) for such missions as DYNA-SOAR and a super ICBM, ABMA was requested to determine the compatibility and desirability of such a configuration for second stage application on SATURN.

Initial studies indicated that the 160-inch diameter, rather than the original 120-inch diameter, was more desirable for several reasons. Based on this, ARPA placed a stop order, 31 July 1959, on all second stage work applicable only to the 120-inch diameter.

ABMA was further requested to perform a study and present a program using SATURN as a carrier vehicle for DYNA-SOAR. This study was presented to ARPA during the latter part of August 1959.

Another evaluation was made during the month of September 1959 by an Ad Hoc Committee, chaired by Dr. York and Dr. Dryden, on the TITAN-C proposal versus the SATURN. This evaluation resulted in a decision reconfirming the SATURN program and dropping the TITAN-C proposal. It also resulted in a request for a new study on SATURN upper stages for a more optimum solution on a long term basis. This ARPA request was specified by a teletype, dated 24 September 1959.

The results of this new system study are summarized in this report. A verbal presentation to ARPA, NASA, DDR&E, USAF, and CMLC was given on 29 and 30 October 1959 in Washington, D. C.

Chapters I through IV of this report are a general summary of the study containing a description, cost, and schedule of the most promising initial SATURN configuration together with conclusions and recommendations. The technical details of the study are contained in Chapter V.

It should be mentioned, however, that although a decision on the upper stage configuration for the initial vehicles is still pending, progress on the SATURN booster and the necessary program facilities is continuing (Ref. 3). The captive dynamic firing of the eight engine static test booster is scheduled for March 1960.

The study covered by this report was performed by personnel from all the Development Operations Division Laboratories with assistance, primarily in the cost and schedule area, from The Martin Company, Denver, Colorado, and Convair Astronautics, San Diego, California.

The study was under the direction of and the report prepared by the Future Projects Design Branch of the Structures and Mechanics Laboratory.

CHAPTER II: (U) OBJECTIVES

A. OBJECTIVES OF THE PROGRAM

The objective of the SATURN program is to provide the United States with a reliable and economical all-purpose space carrier vehicle in the 1.5 million pound thrust class with an acceptable payload capability at the earliest possible date. It became apparent early in the SATURN program that the optimum solution for meeting the program objective required early decisions and more money than expected to be available in the first two or three years. This necessitated a compromise in the basic objectives. Rather than considering long range economy, the vehicle was limited to a configuration dictated by minimum expenditures during the early years of the system. This decision did not affect the reliability criteria, in fact, probably, resulted in a higher initial reliability. However, it also forced compromises in the payload capability and, possibly even more important, reduced the mission flexibility. The compromised SATURN configuration consists of a standard booster, a modified 120-inch diameter TITAN ICBM booster as a second stage, and a modified 120-inch diameter CENTAUR as a third stage. This configuration also provides a vehicle which will require considerable change to incorporate future growth.

B. OBJECTIVE OF THE REPORT

The objective of this report is to present the results of a study on the SATURN vehicle system. The purpose of the study is to design a optimum SATURN vehicle for initial development which will meet the program objectives and provide for any foreseeable growth potential without major changes.

The ground rules for the system study are presented in the following chapter and give the necessary latitude for a true optimization of the complete SATURN program.

Chapters I through IV of thereport areintended to give an overall view of the study and the most desirable vehicle configuration. It includes a summary, conclusions, and recommendations. Chapter V provides the necessary technical data to justify the conclusions and recommendations presented in Chapters I through IV.

CHAPTER III: (C) SUMMARY OF REPORT

A. STUDY GROUND RULES

ARPA specified certain assumptions for the new systems study which can be summarized as follows:

1. The diameter of the second and third stages can be larger than 120 inches.

2. The number of engines in the second and third stages is not restricted to two.

3. The vehicle should be designed for maximum reliability, mission flexibility, and economy.

4. The vehicle should be capable of carrying large payloads with wing areas of up to 1000 sq ft.

5. The R&D program of the early version should consist of ten flight vehicles, with an operational firing rate of six per year thereafter for a period of five years.

6. The vehicle configuration should lead smoothly into a follow-on development program requiring much greater mission capabilities with only minimum modifications in the basic vehicle.

7. One of the schedules studied for the R&D program should be based on a $70 million funding level for FY 1960, $122 million for FY 1961, and about $150 million thereafter as a minimum program. Operational program costs were not to be included in these funding limits.

8. An alternate schedule should be developed with R&D funding requirements not exceeding $250 million per year beginning FY 1961.

9. A follow-on R&D program for an improved version of the SATURN should be developed with an operational availability by 1967, based on an operational firing rate of 12 per year.

B. DESCRIPTION OF VEHICLES

The SATURN vehicle is a multipurpose multistage space vehicle based on a clustered 1.5 million pound thrust booster. The booster consists of eight H-1 engines with a sea level thrust of 188K pounds each and nine clustered tanks: five carrying liquid oxygen and four RP-1 fuel. Clustered tanks were selected on the basis of minimum cost for tooling and maximum transportability and flexibility when booster recovery and exchange of damaged tanks are points of consideration. The weight penalty of this approach when compared to a single tank is considered acceptable because a 10, 000 pound weight increase in the first stage of a three-stage SATURN vehicle results in a payload penalty of only 1.5 percent.

A detailed description of the booster design and operational characteristics can be found in the "SATURN Development and Funding Plan," (Ref. 2).

There is a large number of possibilities for the selection of upper stages. The ARPA ground rules for the initial SATURN System Study (Ref. 1) prescribed, as the cheapest solution, available stages from present programs (ICBM and CENTAUR hardware). As stated previously these ground rules have changed and now allow for the study of optimum performance upper stages. The best possible performance, excluding nuclear propulsion, can be obtained by an uprated booster in connection with high energy upper stages. These offer up to 90, 000 pounds orbital net payload capability and up to 34, 000 pounds net payload for escape missions. This combination would require the development of a new hydrogen-oxygen engine in the 150K pound thrust class. If this development were initiated in 1960, the flight testing could begin in 1964/1965. This date is not satisfactory for the early SATURN program; however, it would be desirable for a follow-on program. Several configurations using this engine have been studied and are discussed in Chapter V of this report. Since these configurations are not of immediate interest in the SATURN program, they are not summarized in this chapter.

The following upper-stage combinations were considered for the early SATURN program:

1. Minimum Solution SATURN B

Second Stage: A standard TITAN I stage, reinforced to withstand the loads occurring at booster cutoff. The reinforcements are limited to the present tooling capabilities. A high-altitude engine ignition system is incorporated with no increase in the engine expansion ratio.

Third Stage. A standard CENTAUR with two XLR-115 Pratt & Whitney engines. The rigidity and critical bending frequency are increased by structural reinforcement.

Advantages.
1. Low initial cost.
2. Early two-stage flights.

Disadvantages.
1. Very limited performance for all missions.
2. No two-stage orbital capability.
3. Dead end development, stages have no growth potential.
4. Poor mission flexibility.

2. Near Minimum Solution SATURN B (Fig. 1)

Second Stage. Same as 1 above, except for an expansion ratio increase from 1:8 to 1:16 and a possible increase in propellant capacity.

Third Stage. Same as 1 above, except for an increase in the propellant capacity, up to 75% of standard.

Advantages. Some performance increase at moderate cost increase.

Disadvantages.
1. The control problem becomes much more difficult. The very low first mode bending frequency requires a completely new control system. This in turn reduces the reliability.
2. No capability exists for winged payloads in excess of 250 sq ft and 10,000 pounds weight.

NOTE: The feasibility of this configuration is questionable.

3. Interim Solution SATURN B (Fig. 2)

Second Stage. A 160-inch TITAN I stage, with a larger expansion ratio of 1:16.

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FIG. 1

SATURN B

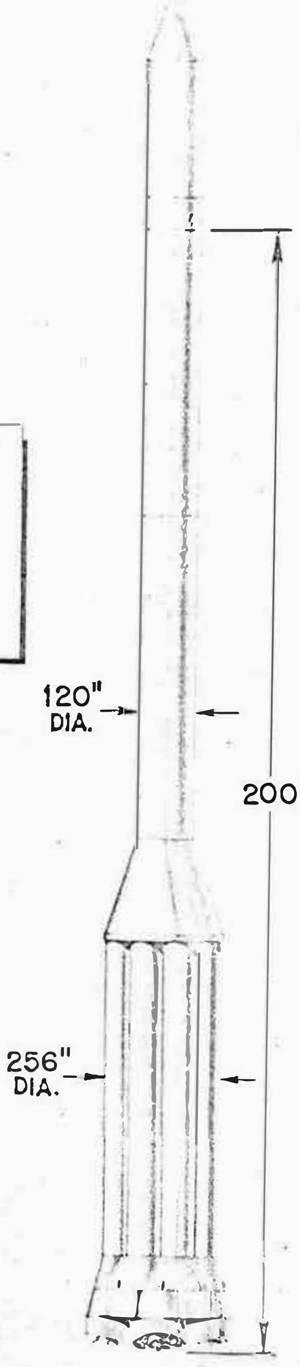
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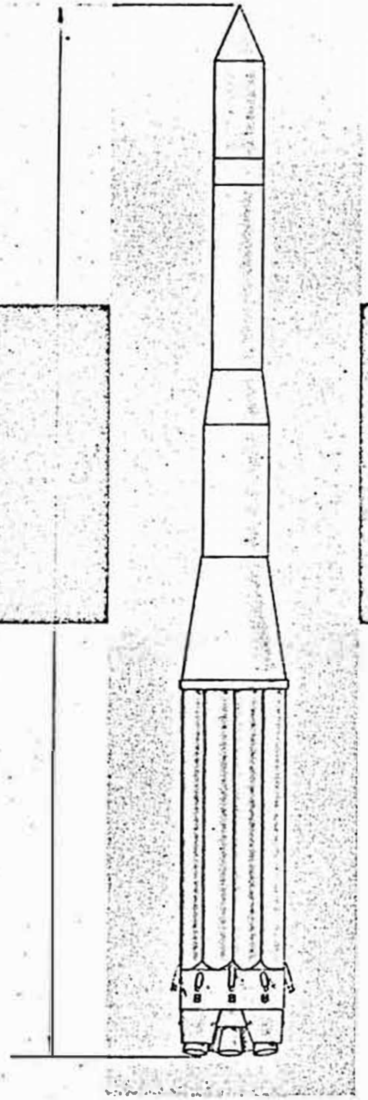
FIG. 2



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Third Stage. Same as 2 above.

Advantages. The vehicle would have fairly good performance capabilities, with fewer control problems.

Disadvantages. The second and third stages have no growth potential.

NOTE: This solution would be acceptable, but it is uneconomical from the overall program point-of-view. It does not offer growth potential unless completely new stages are developed.

4. Optimum Solution SATURN B-1 (Fig. 3)

Second Stage. A four engine cluster with 750 to 880K thrust and a 220-inch diameter.

Third Stage. An 80K four engine cluster using hydrogen and oxygen as propellants. It would have a 220-inch diameter.

Fourth Stage. A standard CENTAUR stage would be highly desirable for the 24-hour orbit and planetary missions, increasing the payload capability considerably. This stage is optional and is based on a 120-inch diameter.

Advantages.
1. Maximum performance.
2. Maximum mission flexibility.
3. Maximum growth potential.
4. Best economy on a $/lb payload basis.

Disadvantages. Requires either higher initial cost in FY 1960 to 1962 or a slow schedule and corresponding delay in operational availability.

NOTE: This program, if funded on an optimum basis, requires less overall program funding if the schedule is optimized as shown later.

Table 1 tabulates the data discussed in this section. Although all except the SATURN B (near minimum) appear feasible, only one is considered optimum from the technical and the overall program points-of-view. If the funding in the early years will not be available at the optimum rates, a slippage of the schedule will result, but the B-1

FIG. 3

TYPICAL B-1 CONFIGURATION

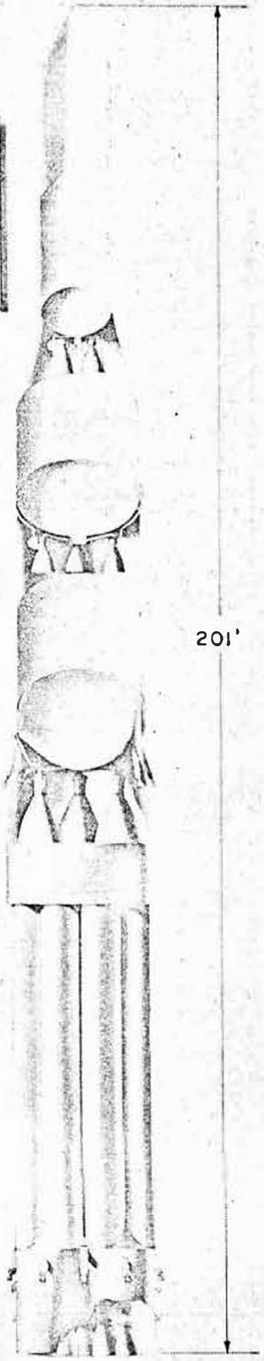

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Table 1
COMPARISON DATA FOR EARLY SATURN VEHICLES

Vehicle	Stage	Propulsion System (Thrust) lb	Specific Impulse, sec	Approximate Propellant Loading, lb	Diameter, in.	Vehicle Length, ft	Remarks
SATURN B (Minimum)	1	8 H-1 = 1.500K	257	742,000	257	195 to 210	Marginal Performance for all Missions
	2	2 × 180K = 360K	289	168,000	120		
	3	2 × 15K = 30K	412	26,000	120		
SATURN B (Near Minimum)	1	8 H-1 = 1.500K	257	742,000	257	230 to 240	Very Marginal from Control Viewpoint
	2	2 × 180K = 360K	299	168,000	120		
	3	2 × 15K = 30K	412	50,000	120		
SATURN B (Interim)	1	8 H-1 = 1.500K	257	697,000	257	195 to 210	Acceptable, but no Growth Potential
	2	2 × 180K = 360K	299	218,000	160		
	3	2 × 15K = 30K	412	47,000	120		
SATURN B-1	1	8 H-1 = 1.500K	257**	600,000	257	O 201	Optimum Configuration
	2	4 × 180 = 720(880)	299(312)	300,000	220		
	3	4 × 20 = 80K	420	75,000	220		
	4*	2 × 20K = 40K	420	25,000	120		

*Optional.

**Varies with mission.

program seems to be feasible and most attractive even with a shortage of funds early in the program.

C. VEHICLE PERFORMANCE AND CAPABILITIES

The various vehicle configurations under consideration in this study have been optimized for specific missions and stage propellant distribution. The trajectories were shaped for each mission to obtain maximum performance. The missions considered were the 307-nautical mile orbit, the escape mission, and the equatorial 24-hour orbit doglegging from the Atlantic Missile Range. The soft lunar landing capabilities, unless stated otherwise, are based on the escape capability, assuming a high energy propellant combination (420 seconds I_{sp}) for the landing maneuver.

The accuracy of the payload capabilities obtained is limited since the weights of the upper stages are estimated, not detailed design weights. The trajectory calculations as such are accurate.

The capabilities of the early possible SATURN configurations are listed in Table 2, showing the net and gross payloads. The net payload includes the actual payload delivered plus the payload container. This would include all payload attitude and position controls that may be required. The gross payload includes the net payload plus any shrouds required for payload protection, the instrument compartment and all instrumentation, and the guidance and control components required to bring the payload in the desired injection trajectory. It also includes any unused propellant reserves. In all calculations presented in this report, a propellant reserve corresponding to 3% of the velocity requirement of the vehicle is provided and is included in the gross payload listed in Table 2.

The payload capabilities for planetary missions, such as Mars and Venus satellites and landing vehicles, are given in Chapter V.

The difference in payload capabilities between the minimum and optimum configurations becomes apparent with the more demanding missions, like the 24-hour orbit and the lunar soft landing. In the low altitude missions, the payload is improved by about approximately 100%; whereas, the payload increases by about 300% in the more difficult missions.

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Table 2
PAYLOAD CAPABILITIES OF EARLY SATURN CONFIGURATIONS

Vehicle	Stages	Weight, lb	96-Minute (307-Nautical Mile) Orbit	Escape Mission	24-Hour Orbit Equatorial Dogleg AMR	Soft Lunar Landing	Remarks
SATURN B (Minimum)	3	Net	19,000	4,200	2,400	700*/ 1050	Very Poor Performance
		Gross	23,000	7,300	4,600	900*/ 1350	
SATURN B (Near Minimum)	3	Net	23,000**	8,400	5,000	2600	Very Marginal Bending Frequency
		Gross	27,000**	12,000	8,100	3100	
SATURN B (Interim)	3	Net	27,000	8,400	5,000	2650	No Growth Potential
		Gross	31,500	12,000	8,100	3100	
SATURN B-1	3	Net	35,000	10,250	5,200	3400	Orbital Refueling Capability
		Gross	40,000	14,000	8,800	3900	
SATURN B-1	4	Net	Not Feasible	11,900	7,800	4000	Only for High Speed Missions (Optional)
		Gross		15,500	10,200	4550	

*300 seconds specific impulse.

**Heavier structure to carry larger payload. Required only for low orbit mission.

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Another important advantage of the B-1 vehicle is the possibility of a four-stage configuration for high speed missions. The thrust limitation of the B versions in the second and third stage prevents higher propellant loadings or another stage. The thrust-to-weight ratio of these stages becomes so small that it will be difficult to fly the desired trajectory and the performance losses become excessive.

The B-1 vehicle has a third advantage of almost constant mass characteristics for all missions.

The B-1 configuration is also the only one capable of placing into orbit a single unit tank large enough to make orbital refueling missions feasible without orbital assembly. With orbital refueling, the individual payload capability can be increased by a factor of ten, and a manned lunar soft landing and return becomes a possibility for the SATURN system.

D. OUTSTANDING CAPABILITIES AND CONFIGURATION COMPARISON

The SATURN program in general offers the country the following outstanding capabilities in the area of space missions:

1. Earliest possibility for surpassing all presently known and planned payload capabilities.

2. Earliest carrier vehicle capable of landing a sizeable non-stationary payload on the lunar surface.

3. Earliest 24-hour orbit communication system capability.

4. Earliest non-marginal space vehicle for planetary satellites and landings.

5. Earliest capability for manned lunar circumnavigation and return.

6. Earliest capability for manned lunar landing and return (by orbital refueling).

7. Earliest capability for large orbital space station and large winged space vehicles.

8. Earliest deep space probe capability beyond Mars and Venus.

In reviewing the upper stage diameter question, the following comparisons illustrate the advantages of the B-1 configuration over the 120-inch diameter upper stages of the SATURN B:

	B (120-inch Diameter)	B-1 (220-inch Diameter)
1. Winged Payload Capability	250 sq ft wing area and 10,000 pounds weight	1000 sq ft wing area and 35,000 pounds weight
2. Low Orbit Capability		
a. Maximum Weight	23,000 pounds	35,000 pounds (three stage)
b. Nominal Payload Diameter	120 inches	220 inches
c. Nominal Payload Volume	920 feet^3	3100 feet^3
d. Testing of Nuclear Propulsion Systems	Very limited	Excellent
3. Twenty-Four Hour Orbit Payload Capability	5000 pounds	7800 pounds
4. Manned Lunar Circum-navigation	Marginal for one man 8400 pounds	Ample for two-men 11,900 pounds
5. Manned Lunar Landing and Return	Orbital Assembly and orbital refueling Approximately 20 vehicle flights Very high launch rate	Orbital refueling Approximately 11 vehicle flights Moderate launch rate

6. Direct Lunar Soft Landing Capability	2650 pounds Marginal for roving payload and maximum reliability	4000 pounds Adequate for roving payload and maximum reliability
7. Deep Space and Solar Probe Capability	In the order of 100 pounds Marginal	In the order of 1000 pounds Adequate
8. Booster Recovery	High (M = 6.3) cutoff and re-entry velocity results in lower recovery reliability	Low (M = 3.3) weight off and re-entry velocity results in higher recovery reliability
9. Upper Stage Growth Potential	For continued 120-inch diameter practically zero growth potential	Excellent
10. Engine Out Capability Upper Stage	Only if occurring near each stage burn out, if at all	Engine out from ignition of each stage possible
11. Cost/Pound in Orbit (Net Payload - 300 Miles)	652 $/lb	536 $/lb
12. Ability to Equal or Surpass Soviet Capability	Poor to questionable	Very good to excellent

E. SCHEDULE AND FUNDING PLANS

It was requested by ARPA that for this system study several different schedules and funding levels be investigated for the initial program. In addition, that one or more follow-on configurations should be shown from a funding and schedule standpoint. Due to the large number of possibilities in funding plans the follow-on vehicle configuration will not be given in this section. Trends in costs and schedules for follow-on programs will be discussed, and typical examples shown in detail in Chapter V.

In order to establish the proper perspective for the data presented in this section, the SATURN B (standard booster, 160-inch diameter second stage and elongated CENTAUR third stage) was chosen as a typical reference. The launch schedule used for this configuration, which has been presented on numerous occassions during the past 6 months, will be used as a datum and, for this report, referred to as the "Original Schedule."

Figure 4 gives the development plan and launch schedule for the B-1 vehicle based on the "Original Schedule". The first four vehicles, launched between the second quarter of CY 1961 and the fourth quarter of CY 1962, will be live first stages and dummy upper stages. Vehicles 1 through 4 will be flown with a reduced engine thrust increasing propulsion system reliability for the initial test. Vehicles 5 and 6 will have live first and second stages and will be launched the second and third quarter of CY 1963, respectively. The primary mission of these two flights, as well as the first four vehicle flights, will be development testing of the carrier vehicle and the booster recovery system. To obtain the maximum vehicle development data from the test of vehicles 5 and 6, it would be more advantageous to incorporate a dummy third stage and duplicate two stages of a three-stage trajectory. However, it would be possible to carry a minimum, up to 10,000 pounds, payload into a low orbit by leaving the dummy third stage off and not duplicating the three-stage trajectory. Vehicles 7 and 8 would be complete three-stage vehicles. They are scheduled for launch during the fourth quarter of CY 1963 and the first quarter of CY 1964, respectively. These two vehicles would have full orbital capability; however, the primary mission, as before, is vehicle development testing. Vehicles 9 and 10 are shown as complete four-stage vehicles, but the final configuration could be changed to a three-stage, same as 7 and 8, depending on the requirements of the program at that time. As mentioned earlier, the SATURN can be flown as either a three- or four-stage vehicle with only a bare minimum of change. The initiation dates for various upper stage developments are shown at the bottom of Fig. 4. This illustrates that the procurement date for the fourth stage is not until the third quarter of CY 1961. For the funding plans presented later in this section, the type of vehicle and mission for the first ten vehicles will be as presented above unless otherwise stated. As stated before, the launch rate for the operational SATURN, was set for this study at 6 flights per year. Since the missions are

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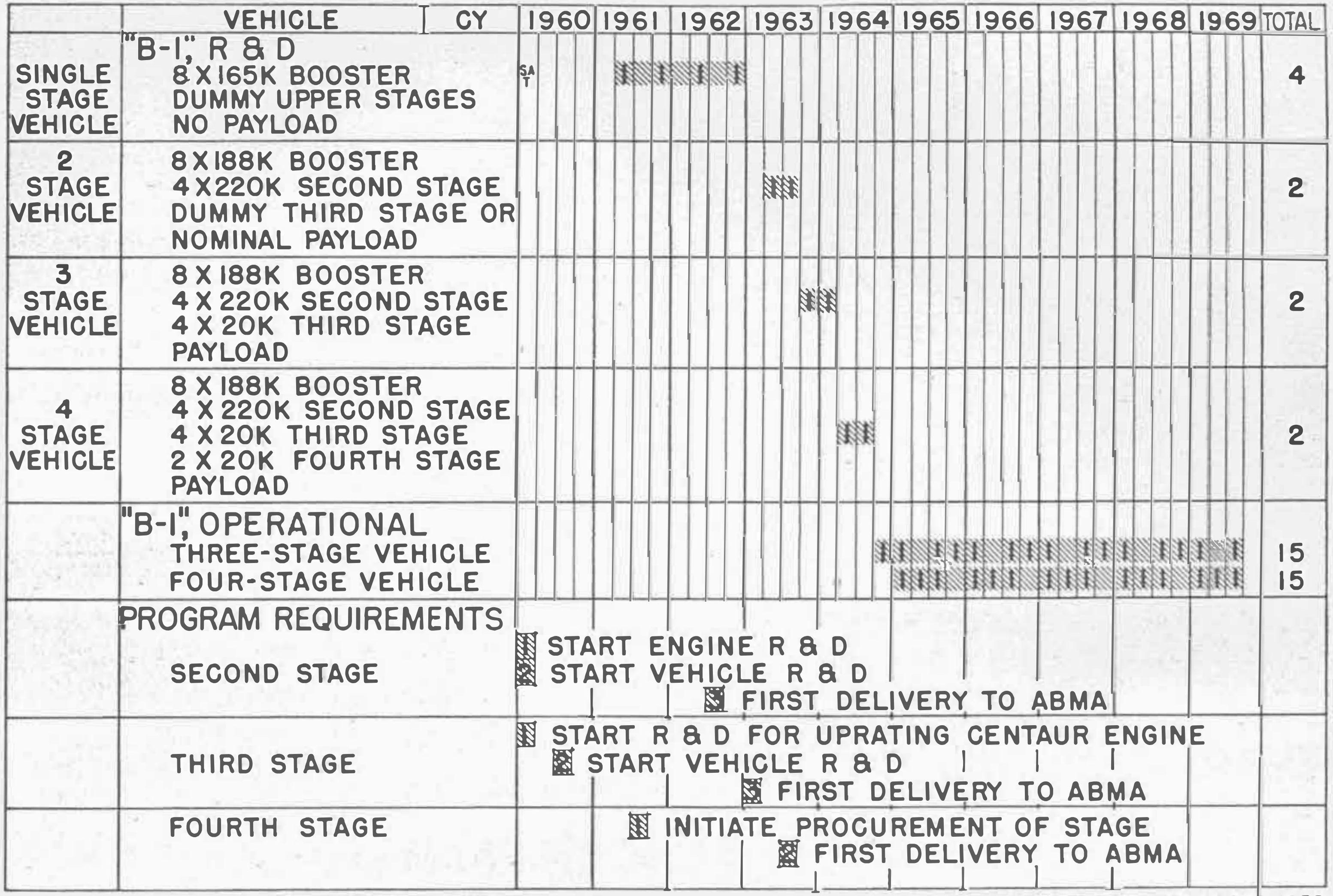
SATURN DEVELOPMENT PLAN & LAUNCHING SCHEDULE
(BASED ON ORIGINAL SCHEDULE)



	VEHICLE CY	1960	1961	1962	1963	1964	1965	1966	1967	1968	1969	TOTAL
SINGLE STAGE VEHICLE	"B-I", R & D 8 X 165K BOOSTER DUMMY UPPER STAGES NO PAYLOAD	SA T	▨	▨								4
2 STAGE VEHICLE	8 X 188K BOOSTER 4 X 220K SECOND STAGE DUMMY THIRD STAGE OR NOMINAL PAYLOAD				▨							2
3 STAGE VEHICLE	8 X 188K BOOSTER 4 X 220K SECOND STAGE 4 X 20K THIRD STAGE PAYLOAD				▨	▨						2
4 STAGE VEHICLE	8 X 188K BOOSTER 4 X 220K SECOND STAGE 4 X 20K THIRD STAGE 2 X 20K FOURTH STAGE PAYLOAD					▨						2
	"B-I", OPERATIONAL THREE-STAGE VEHICLE					▨	▨	▨	▨	▨	▨	15
	FOUR-STAGE VEHICLE						▨	▨	▨	▨	▨	15
	PROGRAM REQUIREMENTS											
	SECOND STAGE	▨ START ENGINE R & D ▨ START VEHICLE R & D		▨ FIRST DELIVERY TO ABMA								
	THIRD STAGE	▨ START R & D FOR UPRATING CENTAUR ENGINE ▨ START VEHICLE R & D			▨ FIRST DELIVERY TO ABMA							
	FOURTH STAGE		▨ INITIATE PROCUREMENT OF STAGE		▨ FIRST DELIVERY TO ABMA							

FIG. 4 GE 140-28-59, 17 OCT 1959
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not known, and assuming that the four-stage B-1 configuration will be used for high-speed missions, the operational flights were divided - 50% for the three-stage vehicle and 50% for the four-stage vehicle.

Before presenting cost data, the following items are clearly stated so that the proper interpetation can be made of the data:

1. First-stage development and production cost, excluding engines, was determined by ABMA.

2. Engine development and production cost was determined by the respective engine manufacturers.

3. Upper-stage development and production cost was determined by Martin-Denver and Convair Astronautics.

4. Ground support equipment development and production cost was derived by all stage and engine developers.

5. Propellant requirements were derived by engine and stage developers and include launch propellants.

6. Launch facilities were determined by ABMA. All other facility requirements were determined by the respective stage and engine developers.

7. Supporting research, transportation, mission and payload integration, and launch operation cost was established by ABMA.

8. The FY budgets cover the period October through September rather than July through June.

9. Cost data received from the various contractors (Rocketdyne, Aerojet, Pratt & Whitney, Martin, and Convair) were used as received with the exception that 10% was added for fee, inflation rate, and contingencies.

10. The funding limitation ground rule of $70 million in FY 1960, $122 million in FY 1961, and $150 million thereafter applies only to the R&D program. Funding requirements for the operational program can be added to the R&D cost giving a total SATURN requirement (R&D plus operational) in excess of these limits.

The schedule and funding plan for the SATURN B, based on the original schedule, is given in Fig. 5. The schedule shows the

SATURN "B" SCHEDULE AND FUNDING PLAN
(BASED ON ORIGINAL SCHEDULE)

CY	1961	1962	1963	1964	1965	1966	1967	1968	1969	TOTAL
OPERATIONAL FLIGHT SCHEDULE				1	2 1 2 1	2 1 2 1	2 1 2 1	2 1 2 1	2 1 2	30
R&D FLIGHT SCHEDULE	1 1	1 1	1 1 1	1 1 1						10

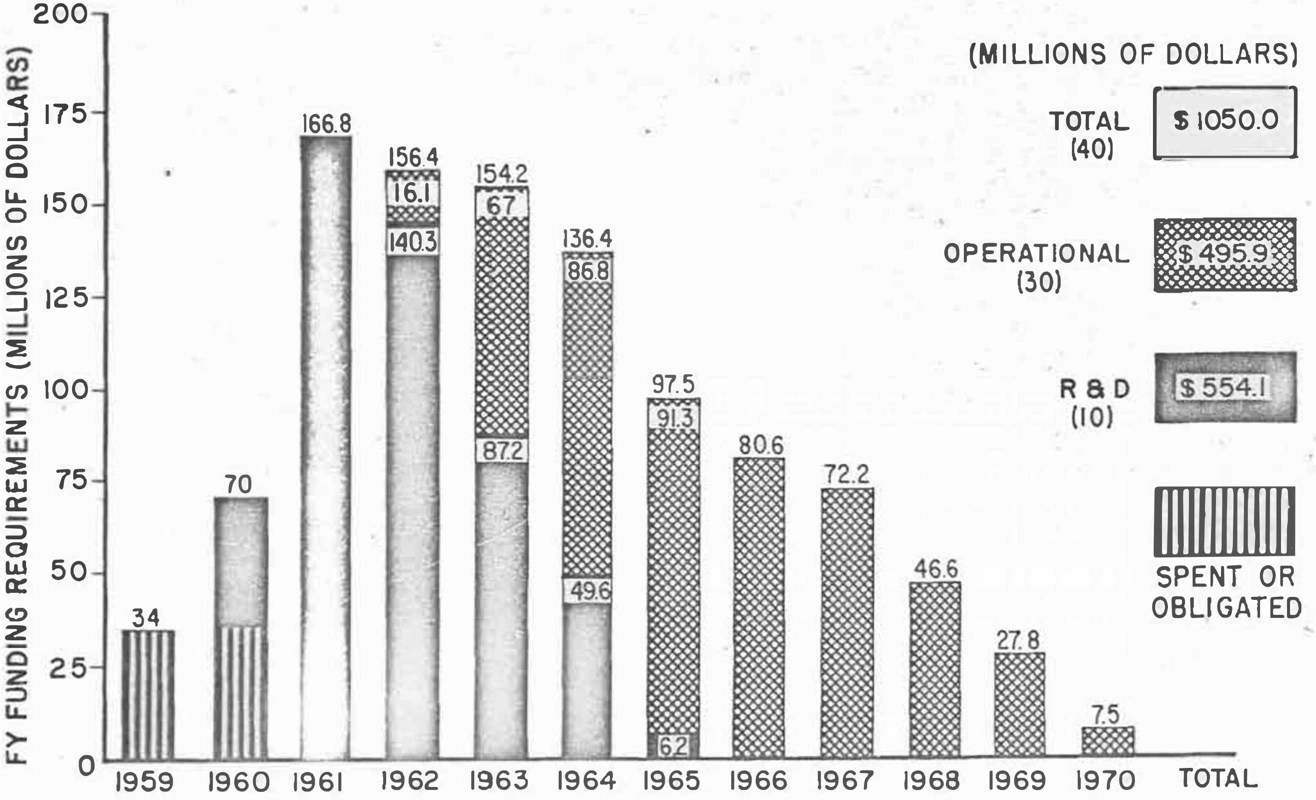

FIG. 5 GE 140-29-59 17 OCT 59
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completion of the ten-vehicle R&D program in the third quarter of 1964 at a R&D cost of $554.1 million. With a launch rate of six per year, starting the fourth quarter of 1964, and a total of thirty flights, the cost of the operational program will be $495.9 million. As can be seen, the FY 1961 requirement of $166.8 million exceeds one of the study ground rules (FY 1960 budget equal $122 million) which requires a schedule slippage. Such a slippage results in a 6-month delay in the operational availability date and a R&D program cost increase of $57 million.

Figure 6 shows the schedule and funding plan for the SATURN B-1 based on the original schedule. This schedule results in a total R&D cost of $599.5 million and an operational cost of $565.5 million. These cost are based on the assumption that one prime contractor performs the development and manufacturing of both the second and third stages. If two prime contractors, one for each of the upper stages, are used, the R&D cost increases to $645.4 million and the operational cost to $602.3 million. Based on the ground rule of $70 million in FY 1960, $122 million in FY 1961, and $150 million thereafter, it can be seen that these limits are exceeded in FY 1961, 1962, and 1963. If these ground rules are used and the schedule slips, the R&D requires an additional 9 months and $57.3 million for completion (Fig. 7). Assuming two prime contractors for the upper stages and the use of the ground rules, the R&D program would slip 12 months and cost $712.6 million. The funding plans for using two prime contractors are given in Chapter V.

One of the requirements of the study was to investigate the effect of $70 million in FY 1960, $122 million in FY 1961, and $250 million in FY 1962. The findings of such an investigation indicate that the program could be accelerated by 3 to 6 months over the program with a FY 1962 budget of $150 million; however, the total program cost would still be more than the cost of the original schedule. The possibility of an additional $100 million (from $150 million to $250 million in FY 1962) could be of much more benefit to the overall program if distributed between FY 1960 and FY 1961.

In conducting the schedule and funding investigations for this system study, it was desirable to determine the optimum developmen and funding plan. Figure 8 illustrates how the total R&D cost varies with the time required to complete the program (first ten flights). The accelerated schedule (Point A on the curve), is near the limit of schedule compression possible, due to minimum lead times requi primarily in the area of facilities. As additional time is allowed,

SATURN "B-1" SCHEDULE & FUNDING PLAN

(BASED ON ORIGINAL SCHEDULE-ONE CONTRACTOR FOR 2nd & 3rd STAGE)

CY	1961	1962	1963	1964	1965	1966	1967	1968	1969	TOTAL
OPERATIONAL FLIGHT SCHEDULE				1	2121	2121	2121	2121	212	30
R&D FLIGHT SCHEDULE	1 1	1 1	111	111						10

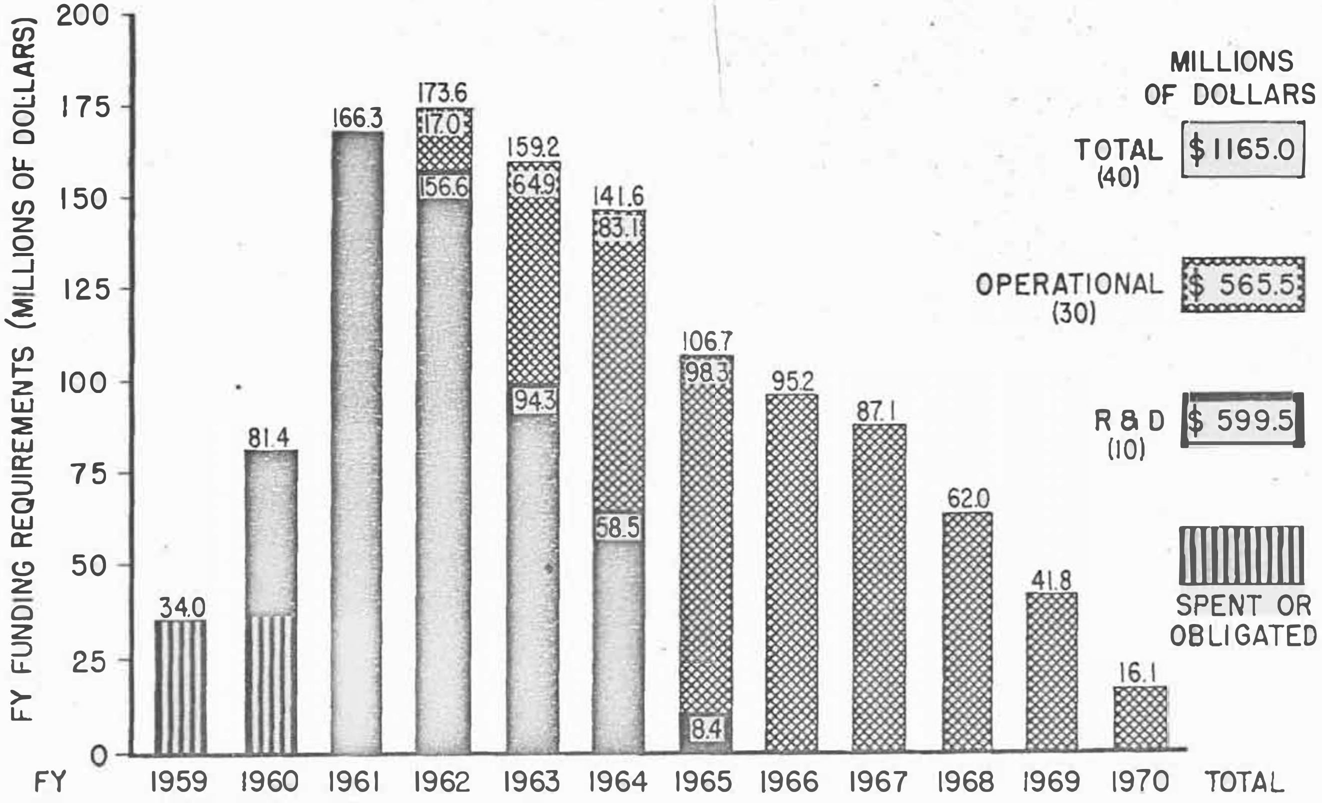

FIG. 6

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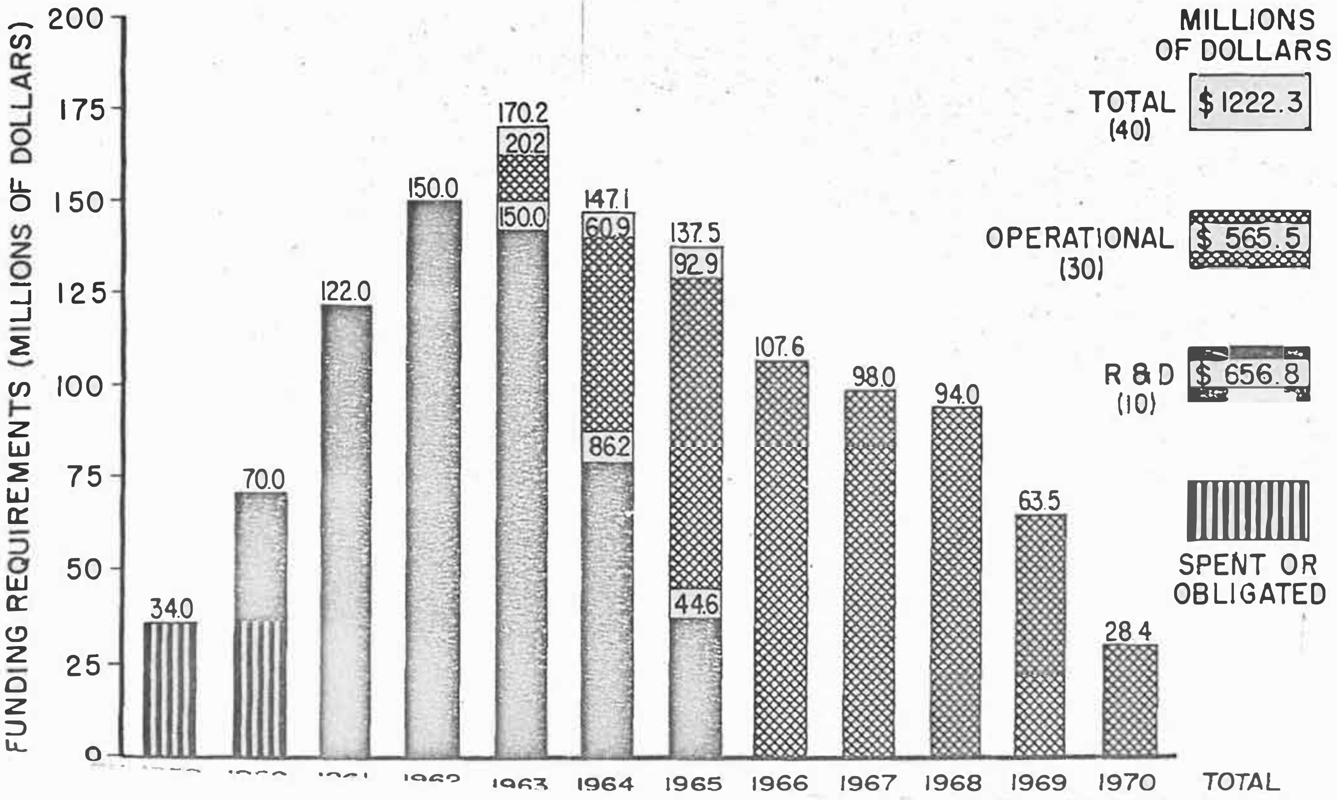
SATURN "B-I" SCHEDULE AND FUNDING PLAN

(BASED ON LIMITED FUNDED PLAN-ONE CONTRACTOR FOR 2nd AND 3rd STAGE)

CY	1961	1962	1963	1964	1965	1966	1967	1968	1969	1970	TOTAL
OPERATIONAL FLIGHT SCHEDULE					12	12 12	12 12	12 12	12 12	12	30
R&D FLIGHT SCHEDULE	I	I I	I	IIII	II						10



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COMPARISON OF SATURN R&D COST VS DEVELOPMENT TIME

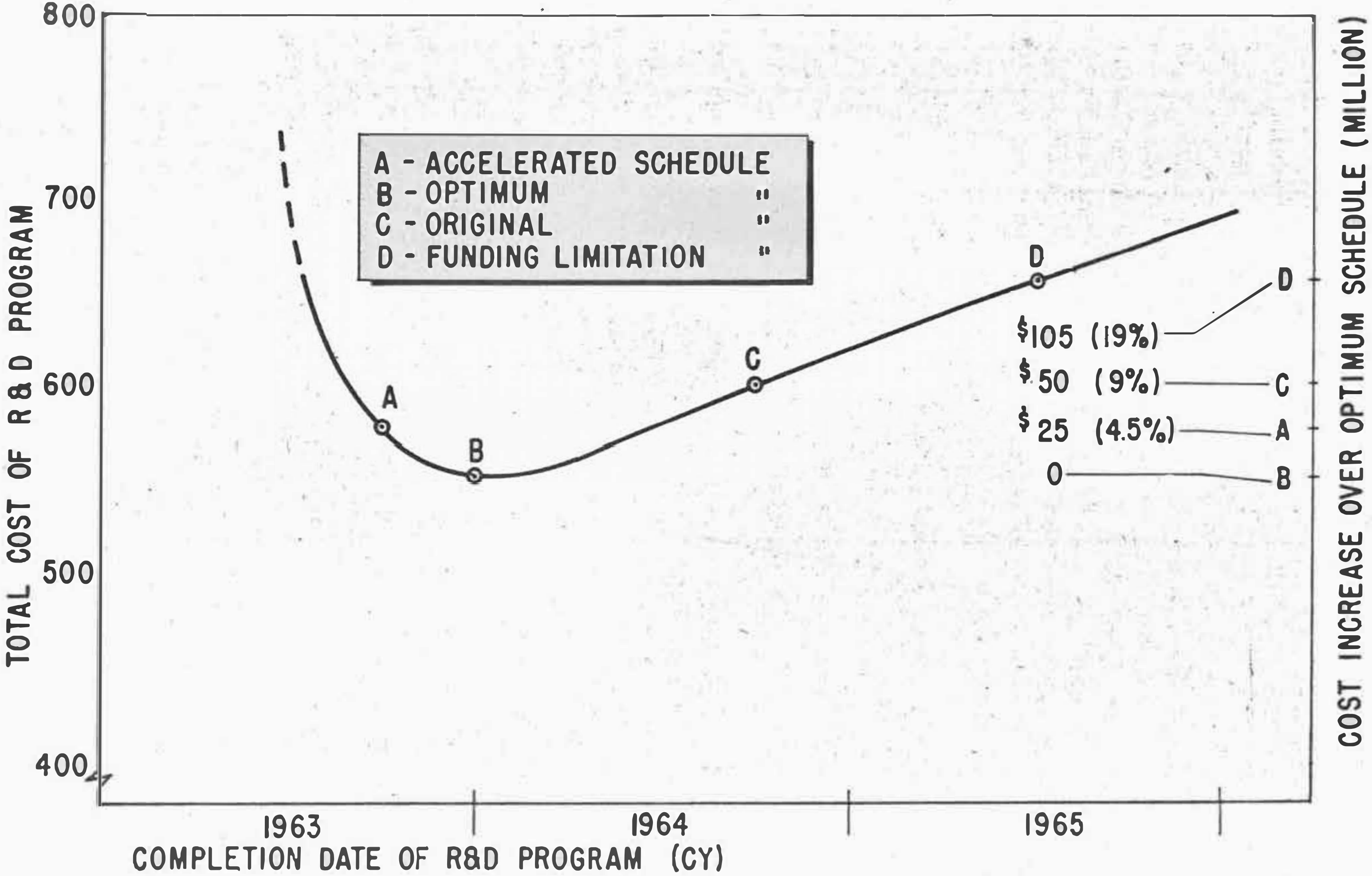

FIG. 8

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from the accelerated schedule, the total cost decreases to a minimum of $550 million and then increases as the program is stretched out. Also plotted is the optimum schedule, point B; original schedule, point C; and the funding limitation (FY 1960 $70 million, FY 1961 $122 million, and thereafter $150 million) schedule, point D.

A development and funding plan for the optimum SATURN B-1 schedule is presented in Fig. 9. In determining the cost for the optimum schedule, no limitations were placed on the availability of funds in FY 1960 and FY 1961. It was therefore possible, with adequate funding in the early years, to assume initiation of second and third stage development during February 1960. This results in a modification to the types of vehicle flown during the 10 vehicle R&D program as follows:

1 through 3	single stage
4 and 5	two-stage
6 through 8	three-stage
9 and 10	four-stage

This is considered to be a near optimum development sequence. Although the funding limits in FY 1960 and 1961 are exceeded, it provides an operational vehicle 9 months earlier than the original schedule and 18 months earlier than the funding limitation schedule for a net savings of $49.5 million and $106.8 million, respectively. In addition to the earlier operational availability and the monetary savings, the optimum schedule would make maximum use of available manpower.

A summary of the schedules and R&D cost for the SATURN B-1 is compared with the SATURN B original schedule in Fig. 10. As shown in this figure, the optimum SATURN B-1 program when compared to the original SATURN B program provides an operational vehicle 9 months earlier, which has larger payload capabilities and a greater mission flexibility for less total money. No cost for the operational program is given since in all cases the launch rate and total number of vehicles are identical for all B-1 schedules and the total operational cost is also equal regardless of the starting date.

The distribution of the total R&D cost is shown in Fig. 11. The data presented in this figure is based on the B-1 vehicle with one upper-stage prime contractor and the original schedule. Although the total cost of the program may vary as much as 20%, as indicated

SATURN B-1 SCHEDULE & FUNDING PLAN

(BASED ON OPTIMUM FUNDED SCHEDULE-ONE CONTRACTOR FOR 2nd & 3rd STAGE)

FY	1961	1962	1963	1964	1965	1966	1967	1968	TOTAL
OPERATIONAL FLIGHT SCHEDULE				1212	1212	1212	1212	1212	30
R&D FLIGHT SCHEDULE	1 1	1 1 1	1 1 1 2						10

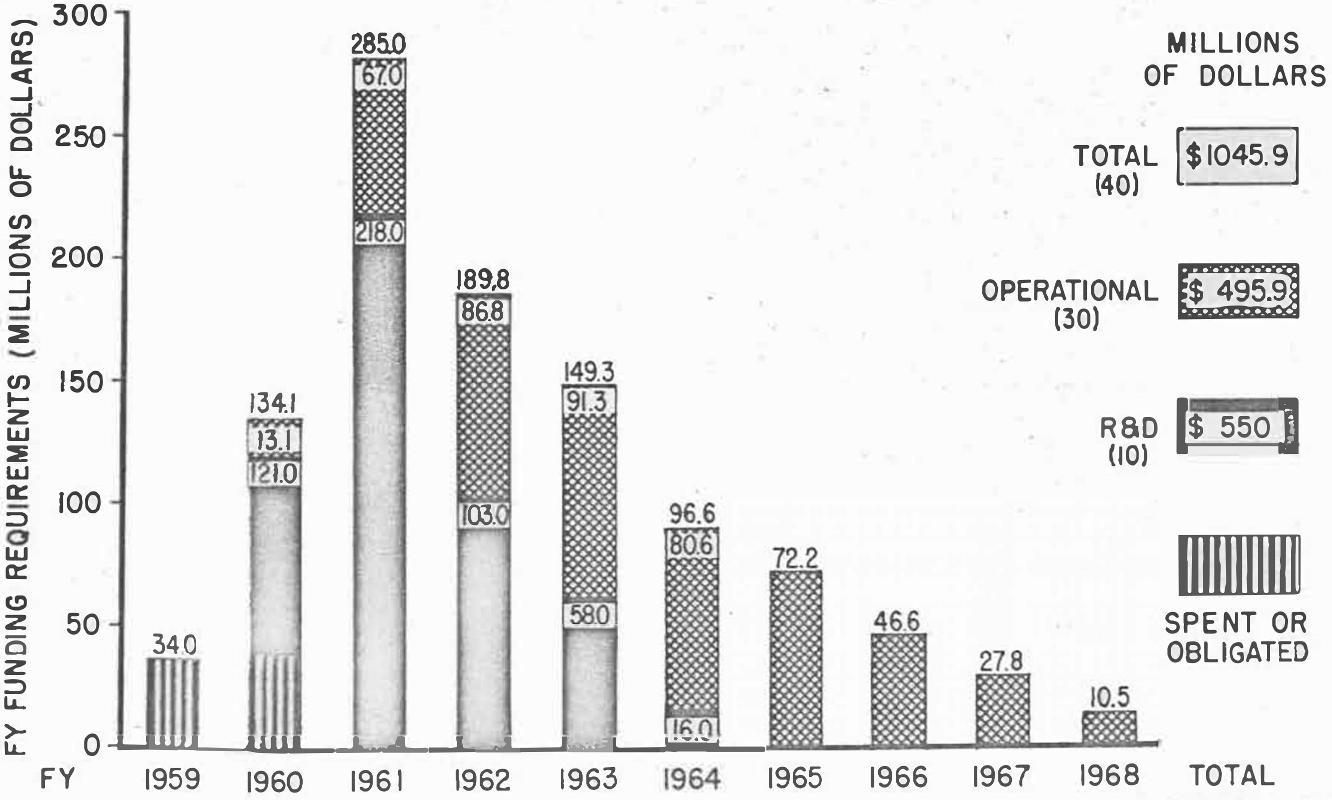

FIG. 9

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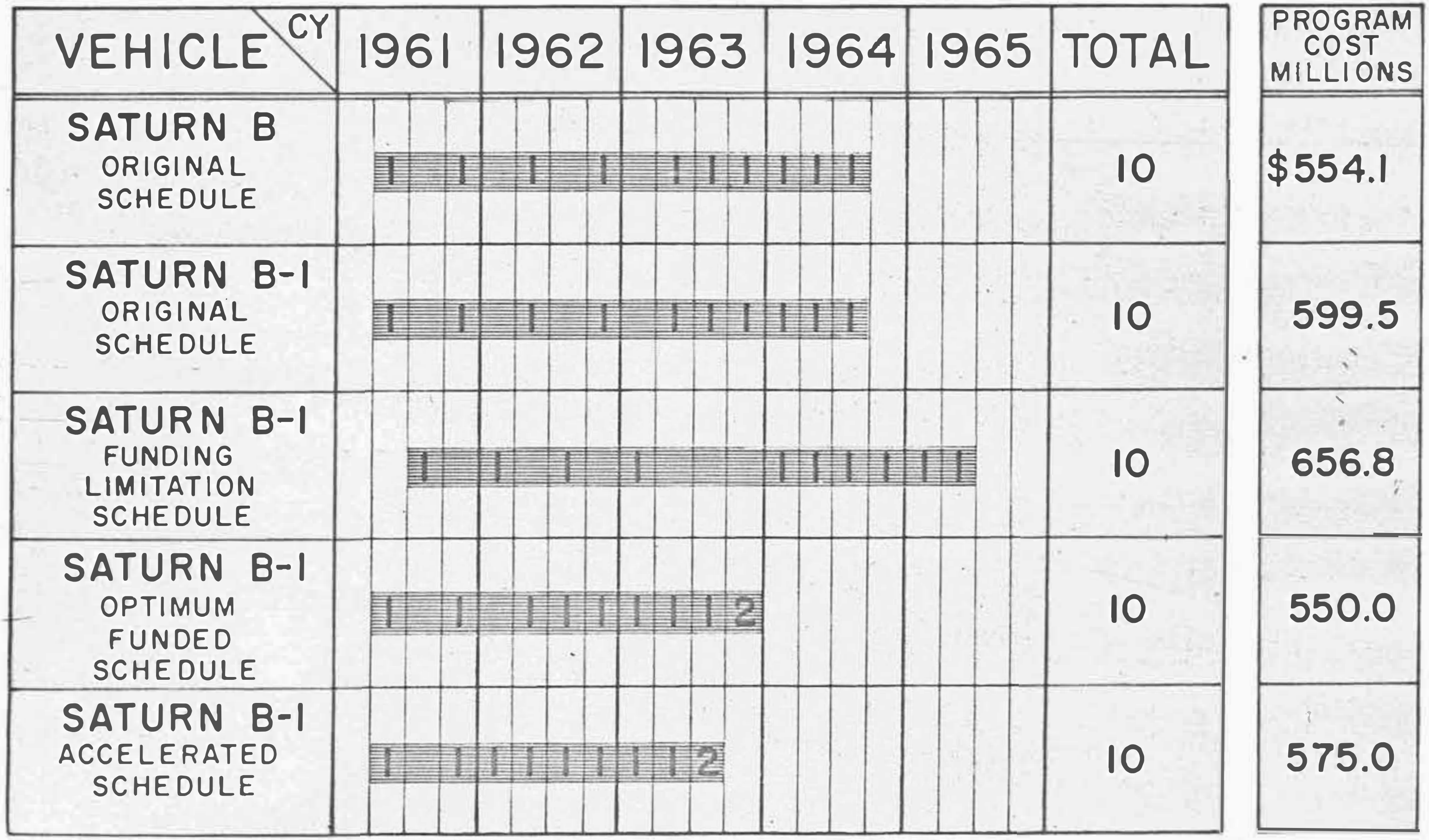
COMPARISON OF SATURN SCHEDULES AND FUNDING PLANS



VEHICLE / CY	1961	1962	1963	1964	1965	TOTAL	PROGRAM COST MILLIONS
SATURN B ORIGINAL SCHEDULE						10	$554.1
SATURN B-1 ORIGINAL SCHEDULE						10	599.5
SATURN B-1 FUNDING LIMITATION SCHEDULE						10	656.8
SATURN B-1 OPTIMUM FUNDED SCHEDULE						10	550.0
SATURN B-1 ACCELERATED SCHEDULE						10	575.0

FIG. 10 REV. "A" GE 140-38-59 17 OCT 59

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TYPICAL SATURN B-1 FUNDING DISTRIBUTION

TOTAL R & D PROGRAM

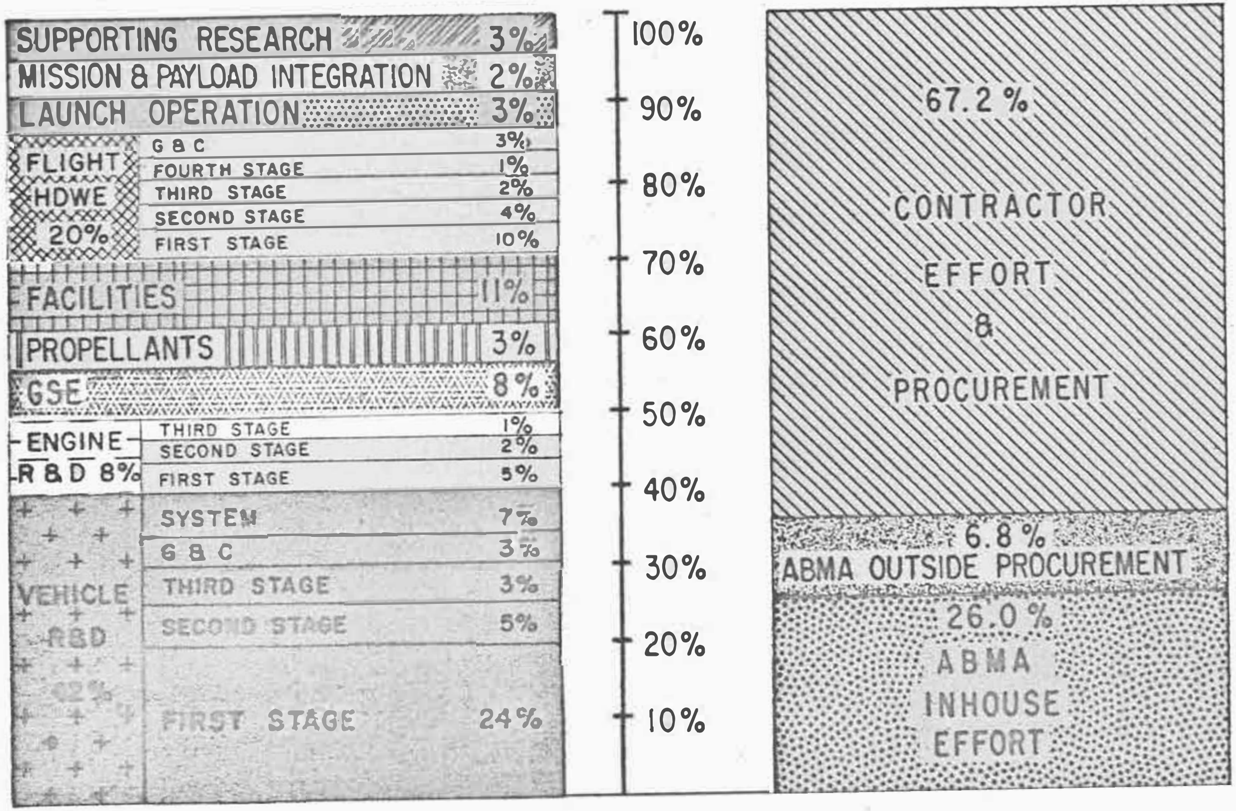

FIG.11 GE 140-35-59 17 OCT 59
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earlier, the percentage of each of the constituents shown in the figure remains relatively stable. The 6.8% for ABMA outside procurement, shown on the right of the figure, is for such items as supporting materials and equipment, and does not include subassemblies, such as engines for the booster.

A similar distribution for the operational cost is given in Fig. 12. This figure is based on a 30 vehicle operational program at a launch rate of 6 per year and assumes the boosters will be recovered and their components refurbished. It has been assumed that all first stages will be assembled at ABMA; therefore, the facilities required to equip a contractor for this work are not included. The facility cost for the operational program is for those facilities required over and above the R&D facilities which will be fully utilized. The 21% for engineering support and product improvement provides for a continuing effort for all engine and stage manufacturers during the complete operational period.

The manpower required to support any of the four program schedules (accelerated, optimum, original, and funding limitation) for the B-1 configuration is available. The number of ABMA personnel required to accomplish the initial development program is plotted versus time on Fig. 13. These curves do not include manpower for either the B-1 operational program or a follow-on development discussed in Chapter V. If these phases of the overall SATURN program are included, each of the manpower curves would tend to level off rather than decline as shown. Both Convair-Astronautics and Martin-Denver has assured ABMA that the necessary personnel to support the program could and would be made available as required.

In evaluating the cost of the SATURN B and SATURN B-1, one criteria for comparison purposes is cost to transport a payload into orbit. Table 3 gives a summary of transportation cost of the SATURN configurations as compared to the ATLAS with two upper stage configurations.

TYPICAL SATURN B-1 FUNDING DISTRIBUTION

TOTAL OPERATIONAL PROGRAM

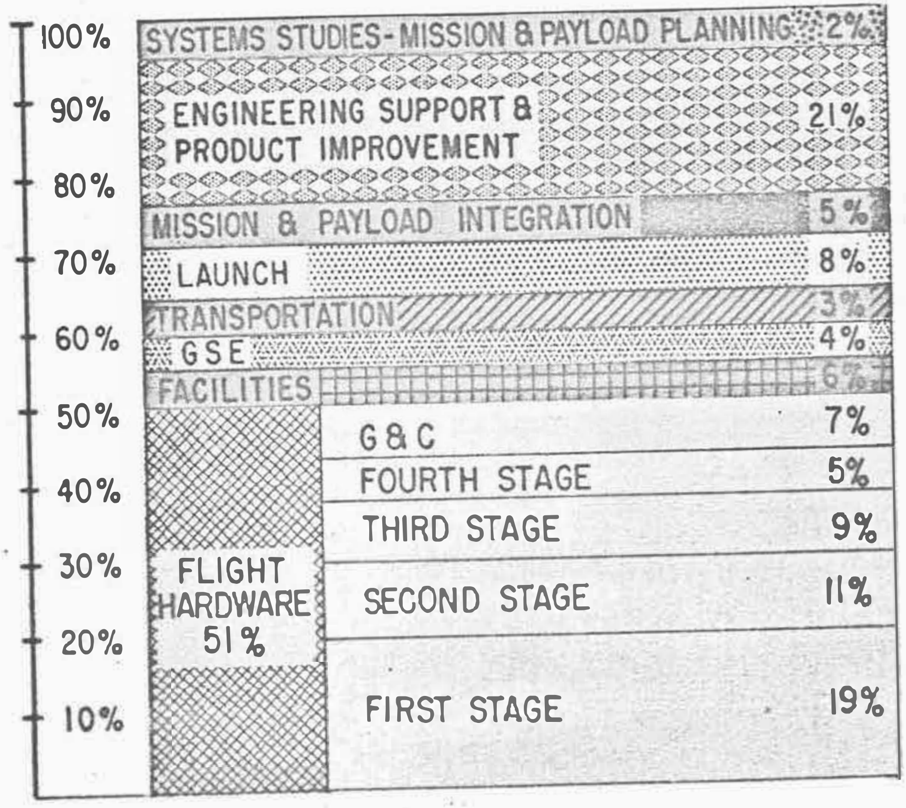


FIG. 12 GE 140-36-59 REV A 17 OCT 59

ABMA DIRECT MANPOWER REQUIRED FOR SATURN DEVELOPMENT PHASE
(BASED ON VARIOUS DEVELOPMENT PLANS)

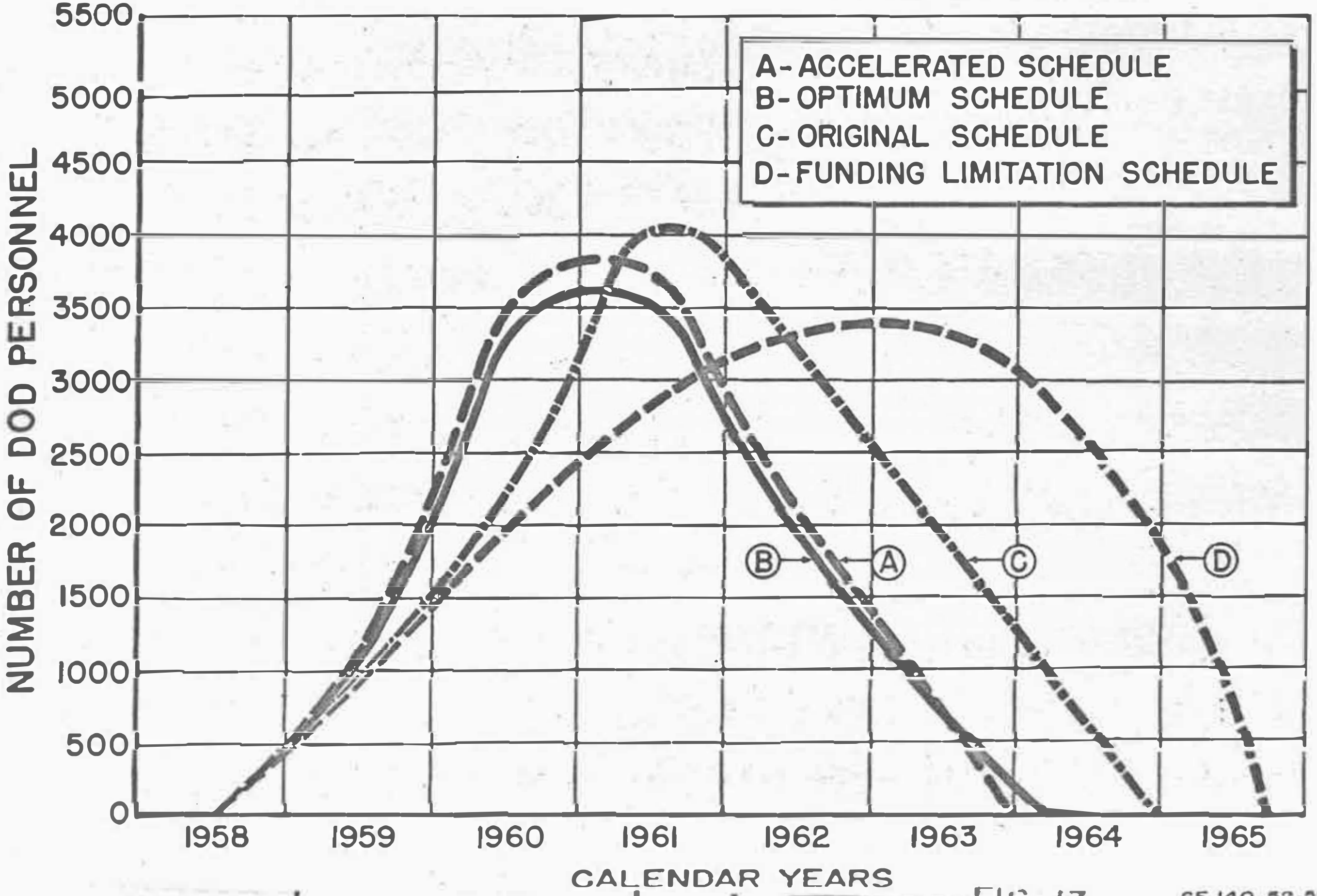


FIG 13

GE 140-52-59

Table 3
TRANSPORTATION COST SUMMARY

(Operational)

ATLAS-VEGA	7.0 million per flight
ATLAS-CENTAUR	7.5 million per flight
SATURN B	15.0 million per flight
SATURN B-1	18.8 million per flight

(Based on Gross Payload)

$/lb	96-Minute (307-Nautical Mile) Orbit	Escape	24-Hour Orbit	Lunar Soft Landing
VEGA	1,400	5,840		
CENTAUR	750	3,000	6,250	12,500
SATURN B (120-inch upper stage)	556	1,250	1,855	4,850
SATURN B-1	469	1,210	1,845	4,140

CHAPTER IV: (C) CONCLUSIONS AND RECOMMENDATIONS

A. TECHNICAL CONCLUSIONS

1. From this study, the following conclusions for the early SATURN program are drawn:

a. The SATURN B-1 is the most desirable of the vehicles studied for the initial program. It offers higher payload capability, greater mission flexibility, and greater growth potential than any of the three B configurations considered, at only a 10 percent increase in cost on the same schedule.

b. The SATURN B-1 offers the following mission flexibility advantages over the B configurations:

(1) Ample payload capability for a manned lunar circumnavigation flight.

(2) Capable of manned lunar landing and return (via orbital refueling).

(3) Capable of carrying large winged payloads (up to 35,000 pounds and 1000 sq ft wing area).

c. The second-stage engine should be selected on the basis of stage contractor familiarity since working arrangements and procedures between stage and engine contractors are already established.

d. The F-1 engine is not compatible with the reliability and schedule requirements for the early SATURN program; however, the SATURN vehicle could provide a flight test-bed for the F-1 when it is available.

2. For the B-1 follow-on program, discussed in Chapter V, it is concluded that:

a. The SATURN B-3 and C vehicles offer the most promise for the B-1 follow-on program. Either could be operational by about 1967; however, no firm conclusion can be made at this time.

b. Additional study is required on the follow-on SATURN configuration before establishing a recommended configuration and insuring its compatibility with the overall national program. This study should be completed in FY 1961.

B. SCHEDULE AND FUNDING CONCLUSIONS

Based on this study, the following conclusions on funding and schedules are made:

1. The funding limitation schedule (Based on $70 million in FY 1960 and $122 million in FY 1961) is inadequate for early SATURN availability and makes only partial use of the ABMA manpower and facility capabilities.

2. The funding limitation, original, and accelerated schedules are not compatible with a minimum cost program. The R&D program cost would increase approximately $110, $50, and $25 million, respectively, over the $550 million required for the optimum schedule.

3. Decisions on the SATURN vehicle configuration and stage manufacturer(s) must be made immediately if the early flight schedule is to be maintained. The present schedule has already slipped due to the lack of a firm vehicle configuration.

4. The development and manufacturing costs of the SATURN B-1 upper stages can be minimized by utilizing one contractor for both upper stages.

C. PROGRAM RECOMMENDATIONS

Based on this study and the preceding conclusions, it is recommended that:

1. The SATURN B-1 configuration (Fig. 14) be approved immediately, regardless of existing funding limitations.

2. The ten vehicle R&D program be approved and funded.

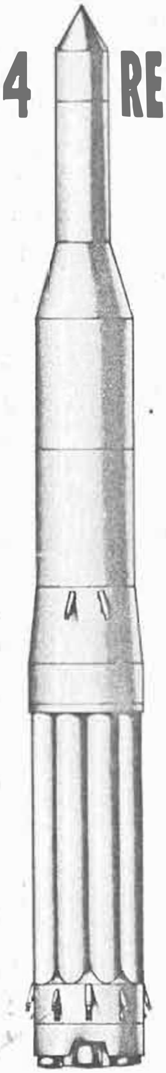
3. Every effort be made to increase FY 1960 and FY 1961 funds since these years dictate the R&D program schedule (Fig. 9).

4. Payloads for the R&D vehicles be defined by July 1960.

5. The types of missions for the first ten operational vehicles be determined by December 1960.

6. The development of the 150K hydrogen/oxygen engine be initiated during FY 1961 and supported.

FIG. 14 RECOMMENDED SATURN CONFIGURATION & PROG



B-1 VEHICLE

FIRST STAGE, STANDARD, 1.5 MILLION LB. THRUST
SECOND STAGE, 220" DIA., 4 ENGINES (LOX/RP)
THIRD STAGE, 220" DIA., 4 ENGINES (LOX/H_2)
FOURTH STAGE, 120" DIA., 2 ENGINES (STD. CENTAUR)

TOTAL 10 VEHICLE R&D PROGRAM COST, 550 MILL
(WITH OPTIMUM FUNDING)

OPERATIONAL DATE OF SATURN, 4 TH QTR. 196
(COMPLETION OF R&D PROGRAM)

NET PAYLOAD CAPABILITIES

300 MILE ORBIT	35,000 LB
ESCAPE	11,900 LB
24 HOUR SATELLITE	7,800 LB
LUNAR SOFT LDG.	3,900 LB

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7. System studies be initiated during FY 1961 for the B-1 operational program and the improved SATURN follow-on program.

8. A detailed design study and the development of long-lead time components for engineering satellites to be flown on the early SATURN R&D vehicles be initiated during FY 1961.

CHAPTER V: (C) TECHNICAL DISCUSSION

This chapter is a summary of technical information compiled during this study. The material presented was selected to provide a basis for the comparisons and conclusions given in the preceding chapters.

A. BASIC PROGRAM CONSIDERATIONS

The primary parameters of the SATURN space vehicle development program are mission flexibility, mission reliability, and economy.

1. Maximum Mission Flexibility

The present national booster program emphasizes that a minimum of basic carrier vehicles should be able to perform all space flight missions, thus obtaining the highest economy and probability of success possible during the next decade. Mission flexibility in the SATURN vehicle is obtained by designing a rugged vehicle capable of carrying all anticipated payloads including large winged payloads, and by selecting a tank capacity for the individual stages which allows optimum propellant loadings for all planned missions. An acceptable schedule flexibility is achieved by a relatively large margin of performance which would make slight changes in the velocity requirements possible. These changes become important in a slippage of the firing schedule and are of specific interest for planetary missions. The optional use of a two-, three-, or four-stage configuration allows for large changes in the overall impulse or total velocity capabilities. A large diameter third stage permits the transportation of large volume payloads for any desired mission. Another important factor which adds considerably to mission flexibility is the requirement for minimum changes to the basic vehicle for various missions. All of the above criteria for mission flexibility have been observed during the layout of the basic SATURN vehicle system.

2. Maximum Mission Reliability

The following design characteristics will greatly aid in reaching maximum mission reliability with a minimum number of flights:

a. Cluster Approach. The choice of clustering available and well-proven rocket engines with reliabilities above 0.95 (connected with engine out capability) provides a maximum of operational safety and a high probability of mission accomplishment. The cluster approach also helps to eliminate potential engine trouble spots early in the program because several engines are tested in each flight. This along with booster recovery promises higher reliability in the development phase.

b. Large Margin for Error. The large size of the vehicle allows for a comfortable margin of performance tolerances. Propellant outages in all stages compensate for mixture ratio shift, for technical tolerances, and for changes in temperature. Approximately 3% of the overall velocity capability of the vehicle is recommended in the first stage for performance margin and dispersions. This margin can be even higher for critical lunar or planetary missions. All performance calculations have been based on these assumptions.

c. Use of Proven Components. Engine subassemblies and engine and tank accessories, as well as guidance and control components from present programs, will be used wherever possible. The selected materials and their characteristics will be known to close tolerances since a large amount of ground and flight testing has been carried out during the present and past programs. Unknowns will be avoided and no "chances" will be taken.

d. Conservative Design Approach. Comfortable safety factors can be used throughout the design of the vehicle and its components. If higher reliability can be obtained, weight is of secondary importance. The small number of vehicles to be flown per year does not allow marginal approaches. This conservative approach will result in relatively heavy vehicles; however, the weight can be reduced in the course of the development by further design and test efforts, if time and funds are available. This conservative design approach provides additional growth potential with respect to performance by allowing refinement of the design whenever it is required. The requirement for manned flights dictates this conservative design approach.

e. Booster Recovery. Recovery of boosters will provide for postflight inspection and possible reuse of components and subassemblies in ground and/or flight testing. This will result in an economical development of component and system reliability at the earliest possible time. Booster recovery is considered mandatory due to the limited number of flights planned for the research and development phase of the SATURN program.

f. System Reliability Considerations. The reliability of past systems may give indicative information for the SATURN in this area. Figures 15 and 16 show the booster and system reliability of certain ballistic missiles. In Fig. 15, which shows the booster reliability versus number of launchings, the following two assumptions are made: if the booster fires for 95% of the calculated burning time, the flight is considered a full success; if it fires for at least 50% of the predicted burning time, it is considered a partial success. The achieved reliabilities for the REDSTONE, JUPITER, ATLAS, THOR, and VANGUARD (first stage) are shown. It is illustrated that a fairly large number of flight tests are required before obtaining a reasonable reliability. Plotting mission reliabilities versus number of flights results in even lower figures as can be seen from Fig. 16. This diagram shows only mission reliabilities of orbital carriers, such as JUNO I, JUNO II, VANGUARD, THOR-ABLE, and THOR-AGENA. In this case only those flights that satisfied the mission requirements were considered successful. This figure shows that a mission reliability of 50% after 10 flights must be considered satisfactory.

The reliability trend for the SATURN, a fairly sophisticated multistage space vehicle, will not be too different from the trends of past vehicles. The larger vehicle must necessarily have a larger number of components, resulting in a potentially less reliable vehicle. This trend can be compensated only by using proven components and a new approach to the reliability problem. The extremely high cost for one flight test and the very low firing rate in the beginning of the program necessitates a new look at the reliability problem, which is discussed further under paragraph A. 3. d.

g. Simplicity. Simplicity is not easy to obtain in a vehicle that should have good mission flexibility and performance with maximum reliability; however, a very careful study must be made to eliminate unnecessary components. If the 120-inch diameter in the upper stages can be avoided, the relatively complex control system of a very slender vehicle is not needed. Great progress has been made in the propulsion area and the H-1 engine is simpler, by an order of magnitude, than the early versions of this series. The SLR-115 engine to be employed in the third stage of SATURN is, by virtue of its hydrogen/oxygen propellants and a very simple cycle, potentially a very reliable engine. Further efforts should be made to simplify the overall system. Unfortunately, this would require a large number of engineering manhours, which can not be achieved in a short period of time.

BOOSTER RELIABILITY VS NUMBER OF LAUNCHES

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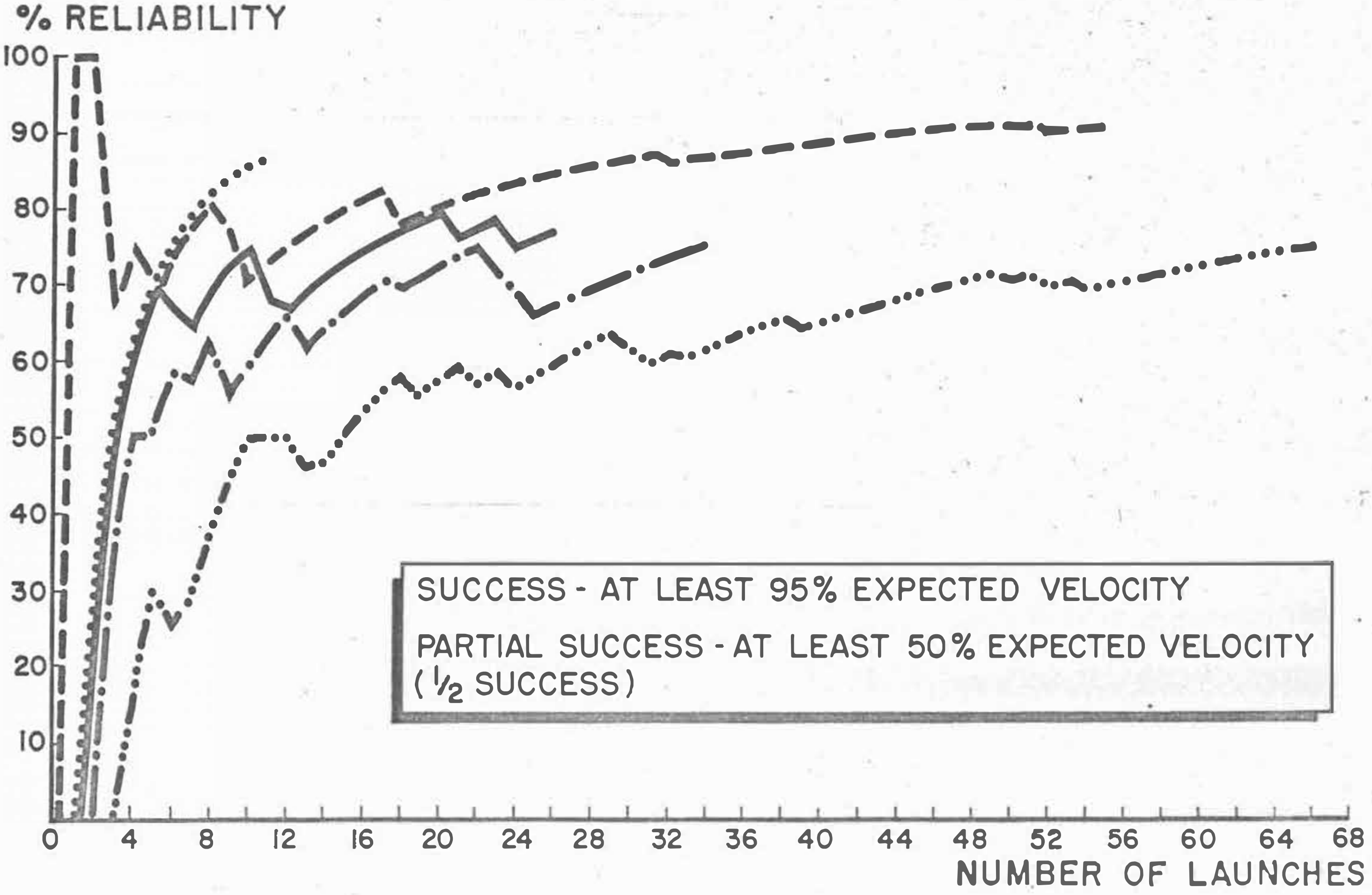

FIG. 15 GE140-12-59
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MULTI-STAGE VEHICLE BOOSTER-MISSION RELIABILITY COMPARISION

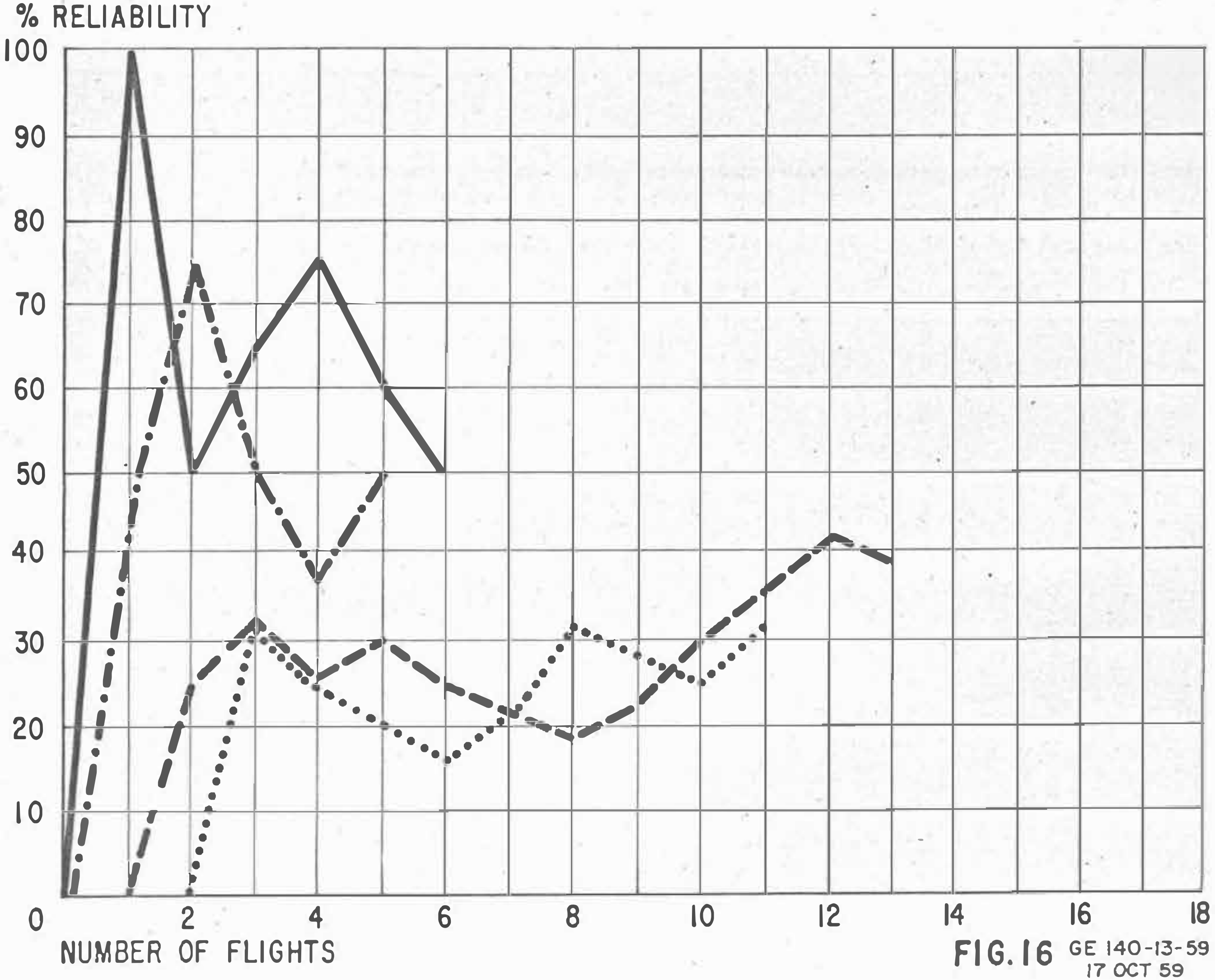

FIG. 16 GE 140-13-59
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h. Crew Safety. The potential use of the SATURN for manned space flight does not increase the reliability of the vehicle per se, but it is a very strong reason, if not the strongest reason, why reliability cannot be compromized for increased performance or reduced cost. An extraordinary effort must be made to obtain a manned rating for the SATURN, even though the basic design features and requirements for such a manned rating have been incorporated in the early SATURN vehicles.

3. Overall Program Economy

It becomes more and more apparent that the funding limitations are and always will be the limitations of this country's space flight activities. Therefore, it is of the utmost importance to design a vehicle which promises the most economical operation. With an initial cost of SATURN in excess of $20 million per flight and the operational cost close to $15 million per flight, the SATURN vehicle is the most expensive space vehicle ever built in this country. On the other hand, it promises to become the most economical means for space transportation in the foreseeable future. The cost for orbital transportation will be reduced from the initial value of $1 million per pound in orbit for the VANGUARD and the present $20,000 per pound for the THOR-AGENA to approximately $500 per pound or less for SATURN. The following factors are considered to influence the cost of the overall program:

a. Minimum Research and Development Program. While the military missile programs allowed 30 to 50 research and development flights for relatively simple single-stage vehicles, such a generous program does not seem to be feasible for the SATURN vehicle. The philosophy of a minimum of R&D flights has been adopted recently for the development of space carrier vehicles. In the case of the SATURN, this will consist of only 10 flights. The payloads of these ten vehicles will be of a relatively simple nature and reasonable cost and will be flown on a non-interference basis only. If there is a conflict of interest, the vehicle development requirements should have the higher priority. The principle might be adopted that a payload should be carried even if the chances for successful completion of a mission are only 30%; however, the cost of the payload should be considerable smaller than the vehicle cost. After the R&D program, the payload and mission requirement should have overriding priority.

b. Booster Recovery. As mentioned earlier, booster recovery not only helps to develop reliability, but also offers a great potential savings for a relatively small investment. The booster represents approximately 50% of the total launch cost and is therefore the most attractive area for potential savings. Figure 17 illustrates the expected expenditures for the recovery of the first 40 SATURN flights, showing the initial investment in the development of the recovery system, the expected furbishing cost (Ref. 4), and the potential savings. It is clear that booster recovery is a desirable feature from the economical point of view, promising savings of 20 to 30% in the overall program. This can amount to several $100 million in a program having reasonably high firing rates.

c. Clustered Propulsion System. The use of presently available engines or those under development is the fastest method and offers the most economical approach for obtaining high thrust levels. The development cost of the propulsion system is small when compared to a single engine booster of comparative thrust. Also the potential higher reliability, specifically in connection with engine-out capability, promises the highest return, in terms of successfully accomplished missions, for the invested dollar.

d. Optimum Reliability Program. If the funds available for reliability and qualification testing are small, only a few vehicles will accomplish their assigned missions. With increasing effort in this area; e.g., manpower and hardware, the success to failure rate will increase to the point where no further amount of money will increase the reliability. This basic relationship is illustrated in Fig. 18. In this figure various amounts of money (10%, 20%, etc.) have been added to a nominal $500 million development program with a total number of ten R&D vehicles plus 30 operational vehicles for the sole purpose of increasing the reliability. For the purpose of this illustration, it is assumed that a 20% increase in funds would result in better than 33 successful flights as compared to 25 for the nominal program. If the difference between cost and saving (one flight = $20 million) is plotted versus the additional reliability money, the curve in the right upper corner is obtained. Under these assumptions this diagram suggests that the optimum reliability will be obtained for a 25% increase in funds over the nominal program, that 20% is a reasonable objective, and that 10% is the absolute minimum that should be considered. There can be no question that this relationship will exist in all missile programs and in the low firing rate space vehicle programs. Based on the best possible estimates, the above orders of magnitude are considered realistic for the SATURN program.

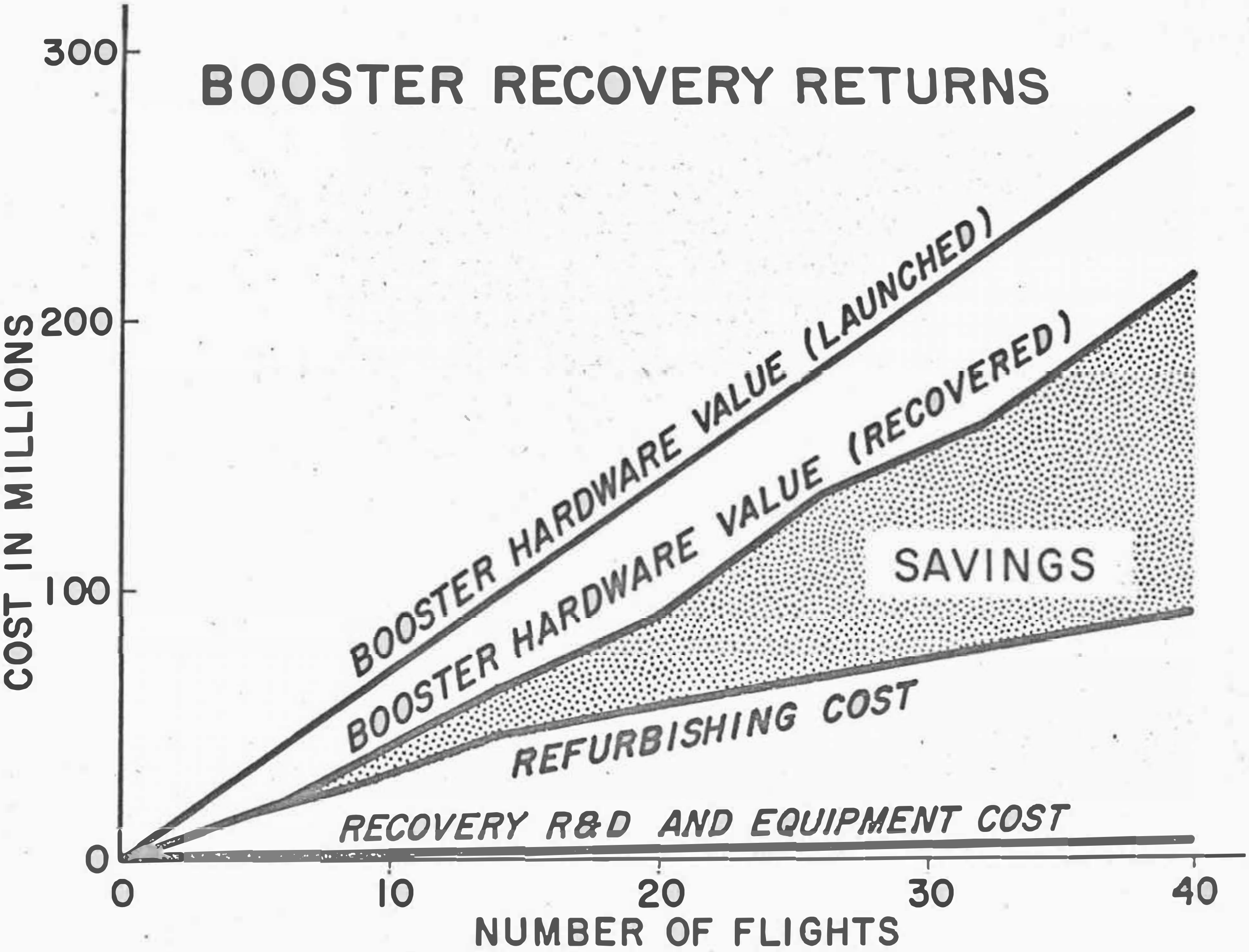

FIG. 17

GE 140-16-59
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FUNDAMENTAL RELIABILITY RELATIONSHIP

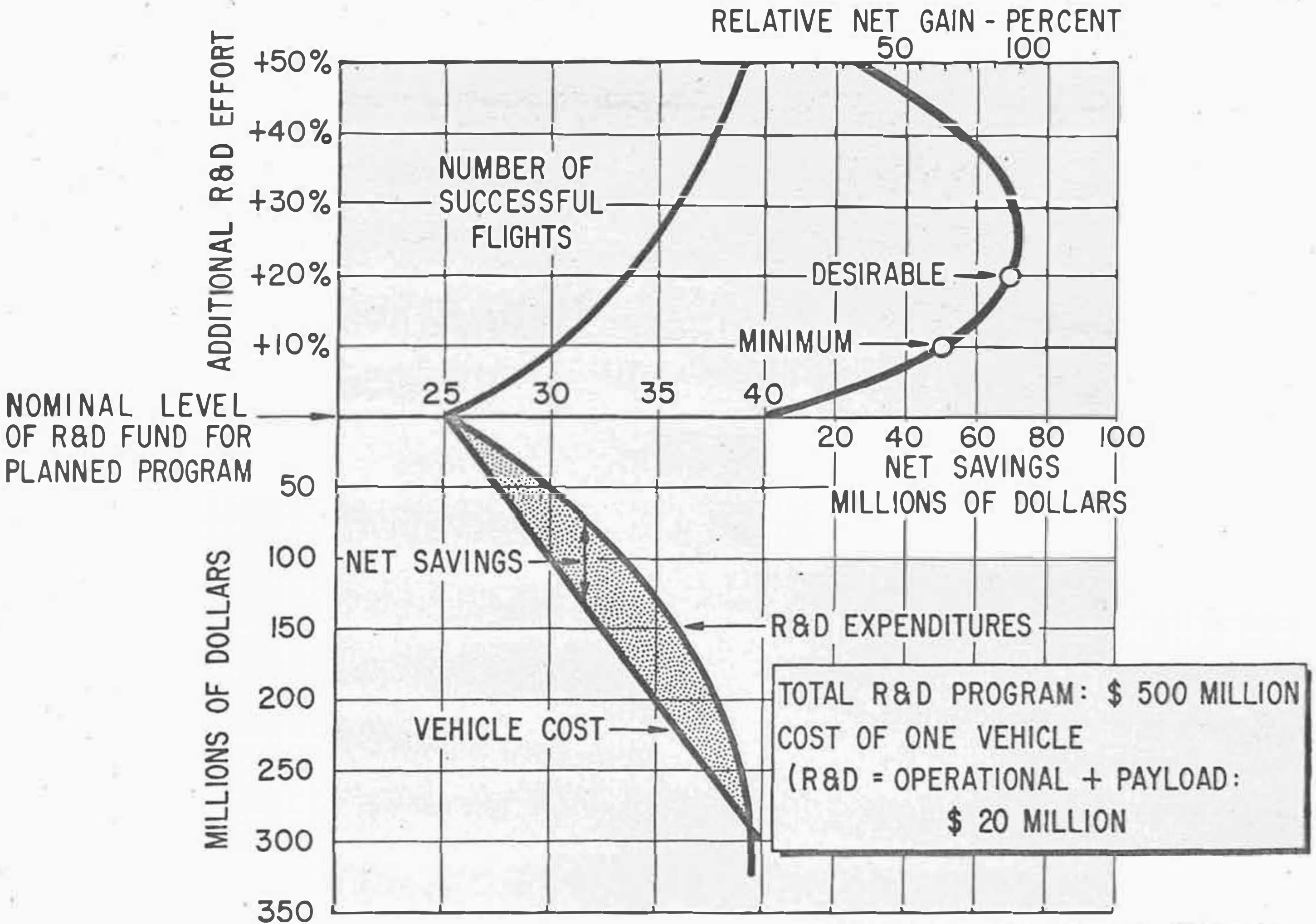

FIG. 18 GE 140-15-59 17 OCT 59

e. Standard Propellant Capacity. Each space vehicle configuration has for each mission an optimum propellant distribution for the individual stages. The SATURN vehicle with a requirement for mission flexibility and versatility will be designed for a standard propellant capacity, but different propellant loadings can be used for the individual missions. This design approach has the advantage of the vehicle having standard dimensions and not requiring changes in the tank lengths from flight to flight. This results in considerable cost savings during manufacturing and operation. The design is feasible because the relatively large payload allows for the small penalties of non-optimum loadings. Each stage for SATURN is designed to carry the optimum propellant loading for any foreseeable mission.

f. Growth Potential With Minimum Changes. Another factor which heavily influences the overall cost of the national program is the inherent capability of the booster to absorb new developments at minimum cost. Only in this way can a vehicle have a long lifetime, which in turn results in high reliability and the best possible economy. The SATURN design is capable of including all presently anticipated new components and improvements in the state of the art. It can not only make use of high energy stages as they become available,but, also can be adapted for thrust level increases. It can even flight-test new propulsion systems, such as nuclear upper stages. Thus, the SATURN is designed for maximum growth potential assuring a lifetime of 10 to 20 years in its basic concept.

These are but a few of the factors which make SATURN the most economical and versatile space transportation system of the national booster program as it is seen today.

B. DESCRIPTION OF VEHICLES UNDER CONSIDERATION

The objective of this study is to select a configuration which satisfies the early requirements and the desire for growth potential at a minimum cost for a follow-on program with operational availability around 1967. Several configurations using different boosters and upper stages look quite attractive under the specified ground rules. In addition to the minimum solutions described in Chapter 3, several configurations for the initial and follow-on program were considered. The initial program configurations were limited to configurations using engines now available or under development, while the follow-on program would allow the development of new engines, increasing the flight performance and other characteristics.

By no means should it be assumed that all the configurations shown in this chapter are recommended for development, nor are they the only possible configurations.

Figure 19 illustrates the available building blocks that represent the individual stages on which the SATURN vehicle system development program can be based. This figure gives the thrust levels, propellants, propellant loadings, specific impulses, and diameters of the individual stages for the configurations studied in this report. It indicates clearly that the original B configuration with a 120- or 160-inch diameter second stage is a dead-end development for growth potential. However, if the larger diameter of 220 inches in the second and third stage is chosen, higher thrust levels become possible, offering more growth potential. The B-1 and B-4 configurations are based on a 1.5 million pound thrust booster; whereas, the B-2, B-3; and C configurations would use a 2 million pound thrust booster. If the B-1 is selected for the early program, it will be relatively easy to obtain growth potential by incorporating a new third stage based on a double barrel 2 × 150K hydrogen/oxygen engine. The development of this engine is to be initiated by NASA next year. The present 4 × 20K third stage would then become the fourth stage, and if the booster could be uprated at the same time, considerable performance increases would be obtained. Preliminary investigations show that either the B-3 or the C version seem to be most desirable for a follow-on program to a B-1 configuration.

With these building blocks, the configurations shown in Fig. 20 are possible, but require further study. The building block series consists of the B version with a 160-inch diameter second stage, the B-1 configuration as an alternate and more optimum configuration for the early development program, and several other configurations for the follow-on program. The B-1 version consists of the standard clustered 1.5 million pound thrust booster now being developed, a 220-inch diameter second stage with 4 × 180 to 220K Rocketdyne or Aerojet engines and a 220-inch third stage with 4 × 20K XLR-115 high-energy propellant engines as the propulsion system. A fourth stage can be added for high speed missions, if desired. The standard CENTAUR stage is near optimum for this purpose and offers additional performance increases. For low altitude orbits, only the three-stage version should be considered. The B-2 configuration is nearly identical to the B-1 with the exception of a 2 million pound thrust first stage and added tankage length in the upper stages. The propulsion systems in the upper stages are the same as in the B-1. The B-3 version is similar to the B-2 with one change. The B-1

SATURN CONFIGURATIONS

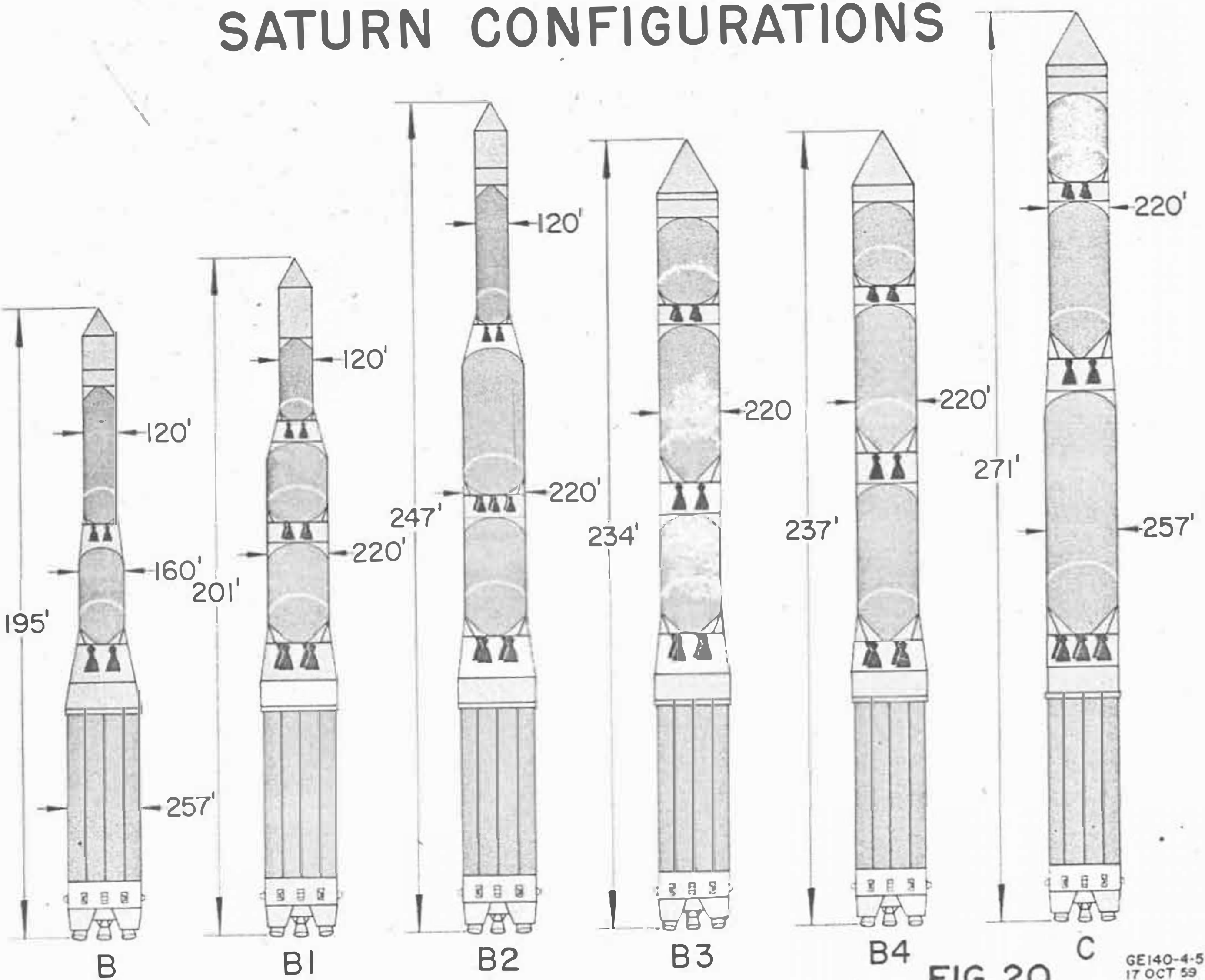
195'
120'
160'
257'
201'
120'
220'
247'
120'
220'
234'
220
237'
220'
271'
220'
257'
B
B1
B2
B3
B4
C

FIG. 20

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third stage becomes the fourth stage and a new third stage with the same diameter (220-inch) and two new 150K high-energy propellant engines would be used. The B-4 is an alternate solution using the 1.5 million pound thrust level booster and a second-stage propulsion system of a high-energy cluster (4 × 150K), providing high-energy propellants in all the upper stages. The most important technical data of these configurations have been summarized in Table 4 for comparison purposes.

C. SATURN BOOSTER DESCRIPTION

1. Design

The SATURN booster design has been described in detail in previous publications (Ref. 2). Figure 21 is an overall view of the booster and Fig. 22 a cross section through the basic structure. The loads are carried through the central lox container which is rigidly connected at the lower and upper end to the four outer lox containers. The four outer fuel containers have a sliding bearing to take care of the contraction during loading and flight. Both container systems are interconnected to provide equalization of the individual container liquid levels in case of engine failure. The basic structure is designed for a 2 million pound thrust level. The container capacity at the present time is 750,000 pounds of propellants. This capacity would be reduced to 650,000 pounds for the B-1 configuration, which would be beneficial from the bending frequency point-of-view. The eight H-1 engines are attached to a spider type thrust frame, four fixed in the center and four gimbaled in an outer ring. The engines are canted by three and seven degrees for the inner (fixed) and outer (control), respectively, to minimize the disturbing moments in case of engine failure at critical stagnation pressure. The booster has a simple recovery system consisting of one stabilization parachute, three main parachutes, and eight brake rockets for reducing the impact velocity to near zero at the moment of impact. The weight of the recovery system is approximately 10% of the cutoff weight of the booster stage, and reduces the payload capability by only about 1.5%.

2. Uprating Possibilities

Preliminary studies have shown that the most effective way of increasing performance is to increase the amount of high-energy propellants in the upper stages. This can be accomplished

TABLE 4

DATA COMPARISON OF SATURN VEHICLES

Configuration	F	I_{sp}	Propellant	Diameter	Length	Propellant Loading
B I	1,500K	257	lox/RP	257"	195'	697 K
II	360 (2x180)	299	lox/RP	160		218
III	30 (2x15)	412	lox/LH	120		47
B_1 I	1,500	257	lox/RP	257	201'	600
II	880 (4x220)	312	lox/RP	220		300
III	80 (4x20)	420	lox/LH	220		75
IV	40 (2x20)	420	lox/LH	120		25
B_2 I	2,000	257	lox/RP	257	247'	746
II	880 (4x220)	312	lox/RP	220		435
III	120 (6x20)	420	lox/LH	220		140
IV	40 (2x20)	420	lox/LH	120		26
B_3 I	2,000	257	lox/RP	257	234'	650
II	880 (4x220)	312	lox/RP	220		435
III	300 (2x150)	420	lox/LH	220		190
IV	80 (4x20)	420	lox/LH	220		80
B_4 I	1,500	257	lox/RP	257	237'	550
II	600 (4x150)	420	lox/LH	220		190
III	300 (2x150)	420	lox/LH	220		170
IV	80 (4x20)	420	lox/LH	220		75
C I	2,000	257	lox/RP	257	271'	650
II	900 (6x150)	420	lox/LH	257		430
III	300 (2x150)	420	lox/LH	220		190
IV	80 (4x20)	420	lox/LH	220		80

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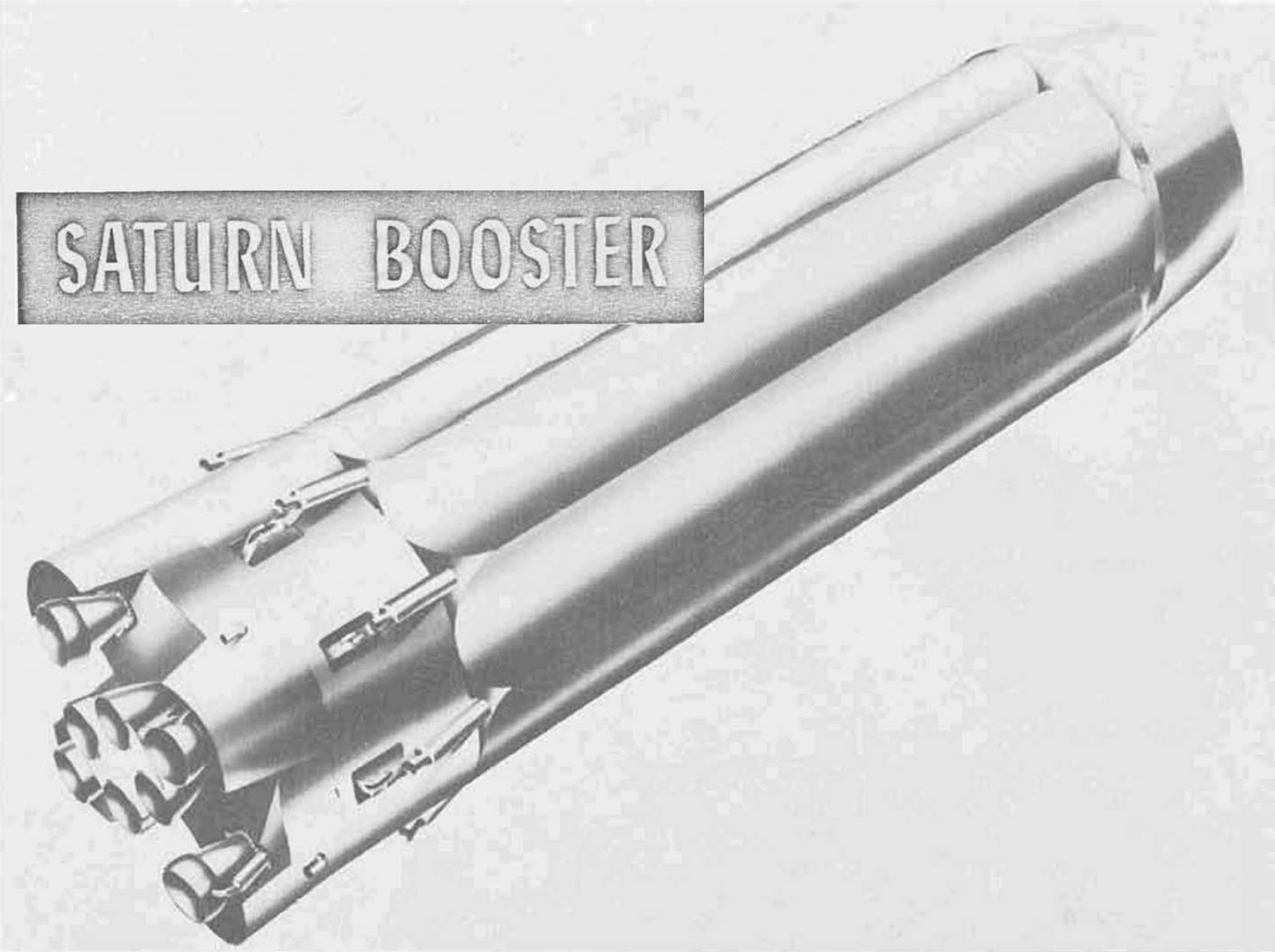
SATURN BOOSTER

FIG. 21 GE 81-59 2 JULY 59

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SATURN BOOSTER STRUCTURE

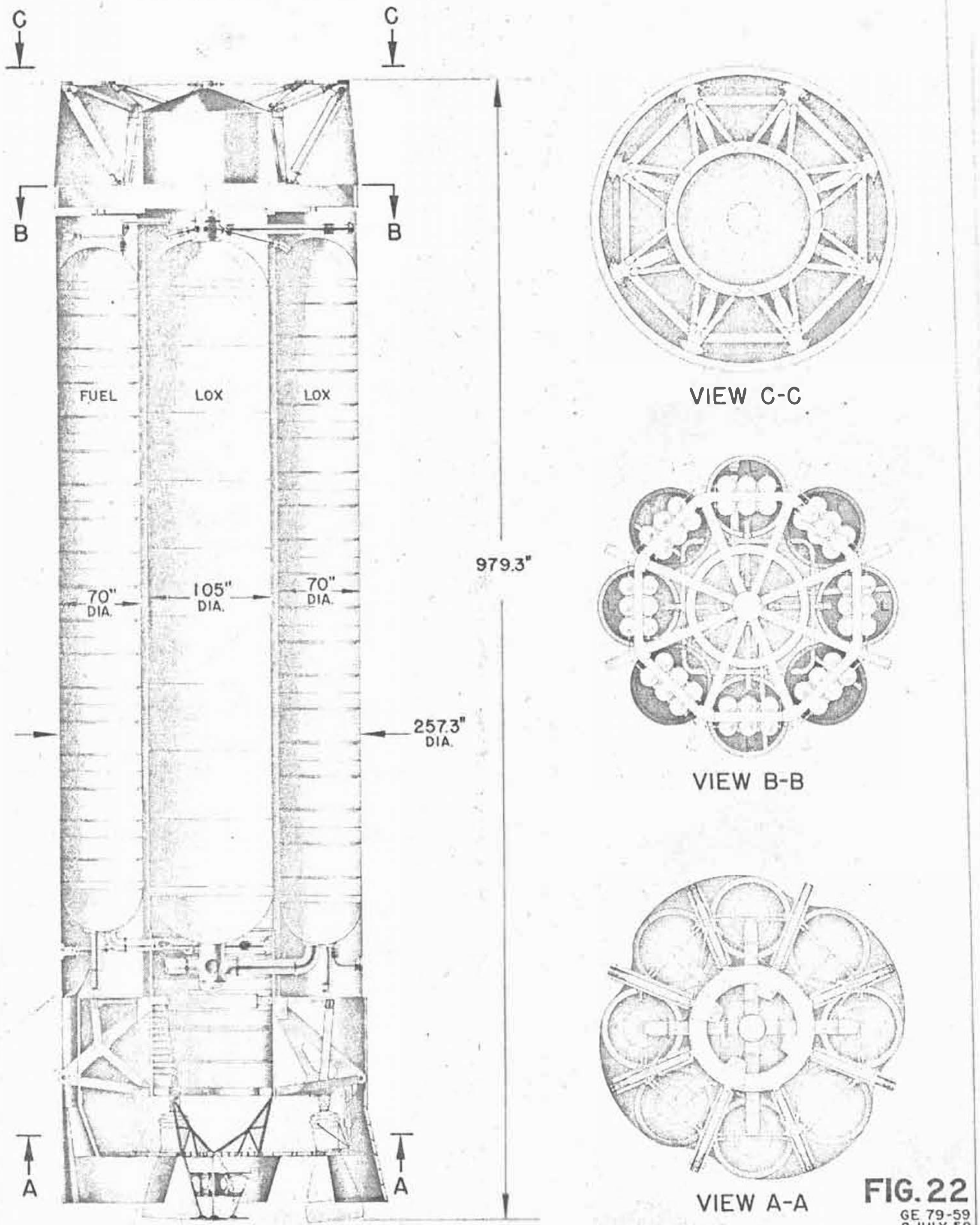
C
C
B
B
FUEL
LOX
LOX
70" DIA.
105" DIA.
70" DIA.
979.3"
257.3" DIA.
A
A
VIEW C-C
VIEW B-B
VIEW A-A

FIG. 22

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primarily by providing larger thrust in the booster. There are several methods of reaching this goal in a follow-on program:

a. Increasing the number of H-1 engines from eight to ten and operate these at 200K, which seems to be a practical limit for this basic engine. This would not necessitate a new engine development, but makes an extensive redesign of the booster mandatory. However, advantage would be taken of the high reliability of the H-1 engine.

b. Replacing the H-1 engine by the H-2 engine which is, component-wise, in the advanced design stage. The Mark XIV turbopump would permit a thrust of 250K, with an additional growth potential to 300K. This engine is designed for simplicity, but would require a development effort in the order of $30 million. The H-2 would start out with a reduced reliability when compared to the H-1; however, it should be superior in the long run. Extensive booster redesign is desirable but not mandatory.

c. Incorporating the F-1 engine, when available, in place of the four inner engines. The four outer engines,providing the major control forces,would remain the same. If this combination is considered for operational flights rather than for testing only, a new single tank booster design becomes desirable or almost mandatory. The rate of progress and funding of the F-1 engine is a decisive factor if this combination is to offer the same overall reliability as the eight engine cluster. The engine-out capability is greatly reduced, because failure of the least reliable engine (F-1) before 90 percent of the flight time will result in mission failure.

Further detailed design studies are required to determine the best way for achieving a 2 million pound thrust level at a minimum cost without reduction of reliability.

3. Typical Weight Breakdowns

Firm conclusions concerning weight differences in the dry weights and cutoff weights for the individual booster approaches cannot be made because only limited preliminary design effort has been expended; however, some preliminary trends can be pointed out. Table 5 is a summary of weight estimates expected for the SATURN booster. The Block I booster is the early design which will be carried out under the time and funding restrictions necessitating shortcuts in various areas. It will be used only for the single-stage flights where weight is a minor consideration, since a very heavy dummy will be flown as upper stages. Emphasis is on maximum reliability. With further design and test effort, the structural weights can be considerably reduced. The Block II booster represents the typical booster expected to be flown during the remaining flights of the R&D program. The Block III booster is representative of the operational vehicle based on the present design philosophy. Single tank configurations are most promising for later designs if weight reduction is considered important. It should be kept in mind that these later booster designs also assume some progress in the state of the art of large booster manufacturing and improved materials, as well as a reduction in the engine net positive suction head requirements. It should also be noted that a 10, 000 pound reduction of the booster cutoff weight results in only a 1.5% increase of the payload capability. It might be considered more desirable to use the required manpower for such a booster redesign to improve the upper stages , resulting in higher performance benefits.

4. Cluster Reliability

Many investigations have been and are being carried out concerning the reliability of clustered-engine propulsion systems. To achieve a SATURN high thrust level booster, there is no choice but to cluster available engines since the single-barrel F-1 engine will not be ready for operational use prior to 1965 or 1966. However, even if both systems were available today, there is still a question of which approach is the most desirable. From the performance point-of-view, the single engine might provide a small performance increase, because it has a higher chamber pressure. From the reliability point-of-view, the answer is hazy. This is illustrated in Fig. 23 which shows the reliability of an eight-engine cluster versus the reliability of the single engine. For the SATURN which is designed to have an engine-out capability right after take-off and a second engine-out after passing the maximum dynamic pressure area, an equivalent of almost $1^1/_2$ spare engines exists. The individual engine reliability of the H-1 engine is presently 94% and is expected to increase to about 96% when flight testing begins. With $1^1/_2$ spares this indicates a propulsion system reliability of better than 98%; thus, if the F-1 engine would be available at the same time, it would need a reliability of 98% to be competitive with the cluster approach. However, by the time the F-1 engine reaches this reliability the cluster should have approached the 99% level. Another consideration which favors the cluster approach is crew safety. A single-engine booster loosing the engine in the maximum dynamic pressure area will rapidly go out of control, leaving little time for the crew to escape. The time margin for a clustered

CLUSTER VERSUS ENGINE SUB-ASSEMBLY RELIABILITY

8 ENGINE CLUSTER

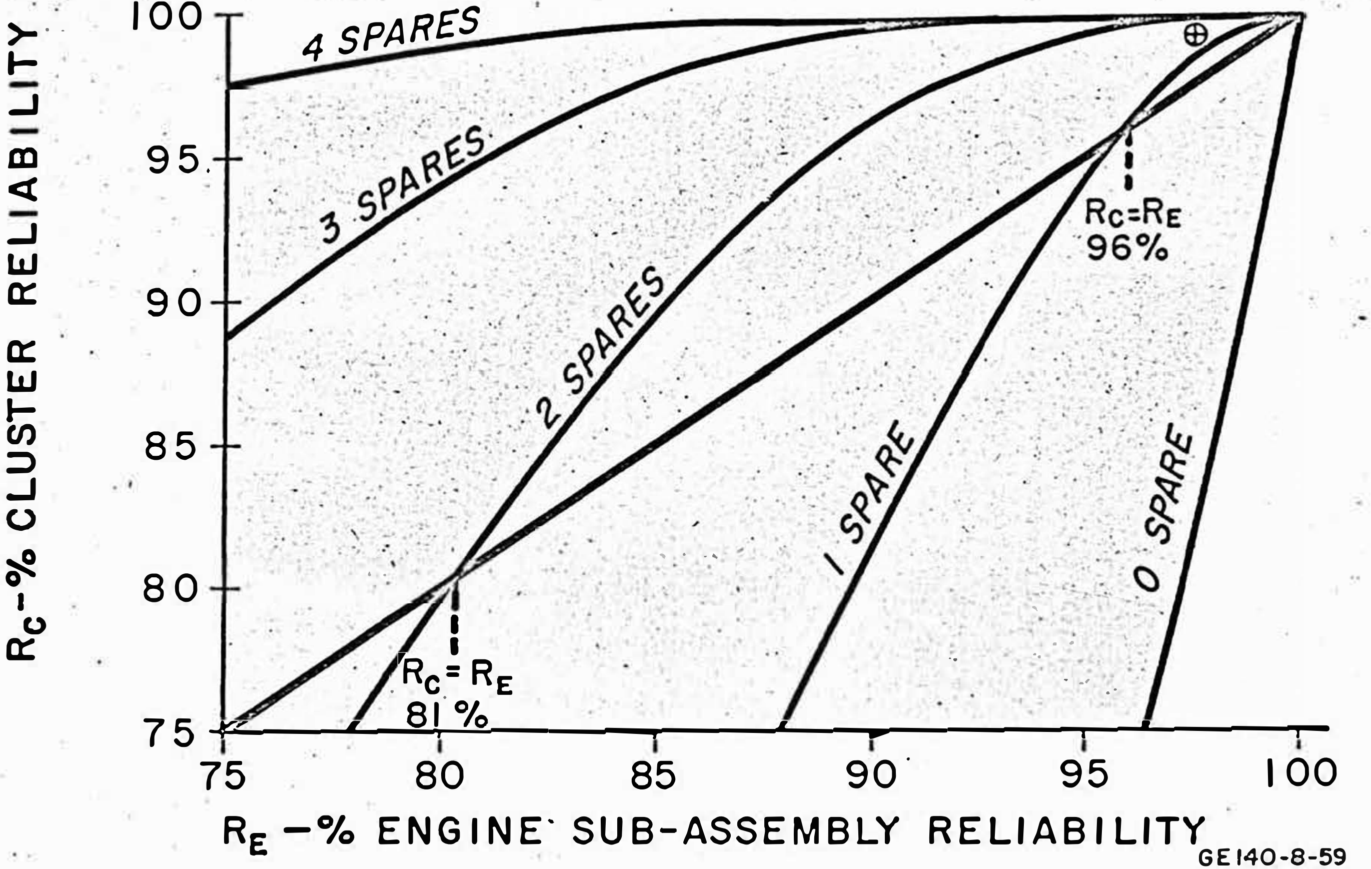

FIG. 23

GE140-8-59
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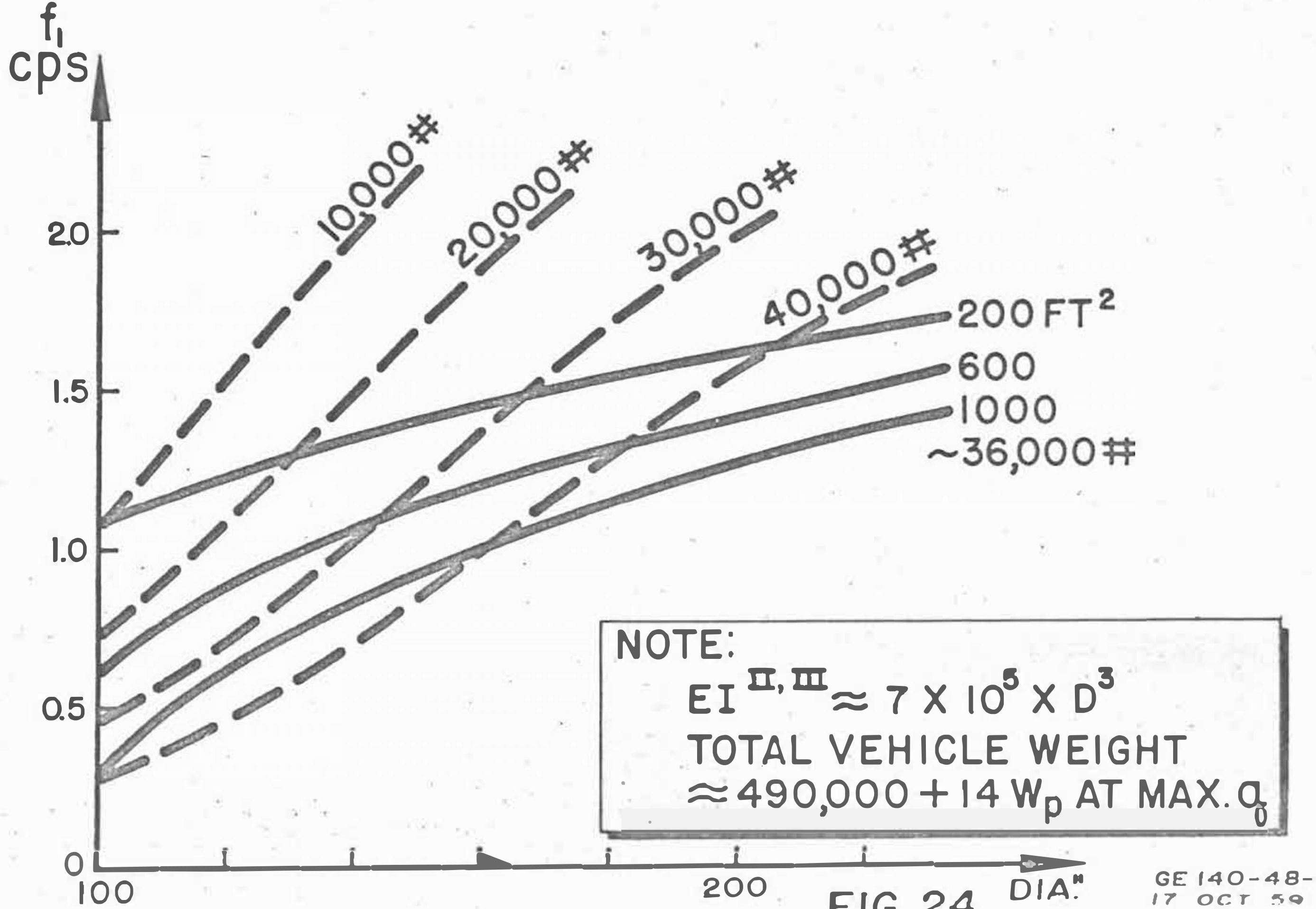
REV. A

FIRST FREE-FREE BENDING FREQUENCY VS DIA. OF UPPER STAGES & PAYLOAD WT. AT MAX. q



NOTE:
$EI^{II,III} \approx 7 \times 10^5 \times D^3$
TOTAL VEHICLE WEIGHT
$\approx 490{,}000 + 14\, W_p$ AT MAX. q

FIG. 24

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1000 ft^2 winged payload carrying approximately 12,000 pounds of fuel with a total weight of approximately 36,000 pounds. It is to be noted that the family of curves for winged payloads shows a much slower increase of frequencies with increasing diameters. This is due to the assumption that the fuselage of a winged payload should not be penalized beyond the strength requirement of its own cantilever bending moment as part of the overall vehicle. This results in:

$$EI_p \approx 0.2 \times 10^{11} + 0.9 \times 10^7 \cdot (A_{wing})^{1.5} \quad (\text{lb-in}^2)$$

For unwinged payloads, based on a packaging specific gravity of 0.10:

$$EI_p \approx 4 \times 10^7 \cdot W_{payload} \quad (\text{lb-in.}^2)$$

which is a conservative assumption.

The ratio of unwinged to winged EI_p is equal to approximately 5 on an average.

The EI_p of the second and third stage is equal to:

$$EI_{p_{II,\,III}} = \eta(n \cdot W_{prop} \cdot r^2 + \pi \cdot p_2 \cdot r^4)/\sigma_{allowable} \quad (\text{lb-in.}^2)$$

based on internal pressure plus dynamic head, p_2,, a given propellant weight, W_{prop}, and a longitudinal load factor, n, at full stage. A reduction factor, η, takes care of variations of wall thickness. The booster EI_p is held constant (40×10^{11} lb-in²). This is based on the average effective stiffness of the present tank cluster, considering that the lox center tank is reinforced by the four outer lox tanks which act as tension and compression ties only.

The stage diameter required for minimum structural weight of all stages based on internal pressure, dynamic head, volume, and thrust structure is found from: $D''_{opt} = 115\,(n\,W_{pr}^2/p_i\rho_b)^{0.2}$ in which p_i = internal pressure (psia), W_{pr} = total propellant weight of stage $\times\ 10^{-5}$ (pounds), $\rho_b = \left(1 + \frac{1}{m}\right) / \left(\frac{1}{\rho_L} + \frac{1}{m\rho_F}\right)$ lb/in.³, the bulk density of oxidizer and fuel with $m = W_{lox}/W_{fuel}$. For SATURN B-1, the optimum diameters are D_I = 240 inches, D_{II} = 225 inches, D_{III} = 200 inches. For SATURN B the optimum diameters are D_I = 240 inches, D_{II} = 180 inches, D_{III} = 150 inches.

Rigidity and bending strength requirements do not appreciably change these diameters. The weight minima are flat. A 10% change of D increases the weight by only 1.5%.

Table 6 summarizes the bending frequency estimates for the winged and unwinged payloads versus stage diameters. It is not possible at this date to make a definitive statement concerning the minimum required bending frequency. However, based on aeroelastic control-feedback stability analyses performed for previous SATURN configurations, it is felt that a frequency of 1.5 cps is uncomfortably low, if it is assumed that a rigid body control frequency of about 0.3 cps undamped (0.2 damped) requires quite sophisticated shaping networks.

Reducing the rigid body control frequency will generally have the effect of reducing the severity of the bending-control interaction, but, this may be at the risk of getting more involved with the

Table 6

SUMMARY OF ESTIMATED BENDING FREQUENCIES

B-1 Vehicle With a 36,000 Pound Unwinged Payload	
Diameter of Second and Third Stage	f_1 (cps) at q maximum
120 inches	0.6
160 inches	1.2
192 inches	1.7
220 inches	2.0
Same Vehicle Carrying a 36,000 Pound Winged Payload With 1000 ft^2 Wing Area	
Diameter of Second and Third Stage	f_1 (cps) at q maximum
120 inches	0.6
160 inches	1.0
192 inches	1.2
220 inches	1.4

sloshing phenomena, and has the further disadvantage of increasing the total required response of the control system to wind inputs with high gradients (wind shear). For winged payloads, it is definitely desirable to have a 220-inch diameter for the upper stages. For unwinged payloads, a 160- to 192-inch diameter appears acceptable.

The weight penalty imposed on the upper stages of the SATURN by winged payloads, versus diameter is shown in Fig. 25. The weights of upper stages carrrying an unwinged payload of the same weight were taken as the reference case for developing this diagram. Two families of curves indicate the weight increase due to strength requirements (M_b) and stiffness requirements (EI) for a first free-free bending frequency of 1.3 cps. Two winged payloads, 1000 ft^2 and 600 ft^2, are shown. The weight increase required to carry 250 ft^2 winged payloads is negligible even with a diameter of only 120 inches for both upper stages. To meet the strength requirement and to have a fair basis for weight comparisons, all calculations were based on a stringer or ribbed-shell design with the internal pressure optimized thus balancing the hoop strength and bending strength. The optimum pressure increases quite rapidly with decreasing diameter and a few values are indicated on the "M_b" curves.

Should the engine require a pressure greater than P_{opt}, the reference weight increases, resulting in an "apparent" reduction of ΔW. The "M_b" curve for $p = 40$ psi independent of D is also shown.

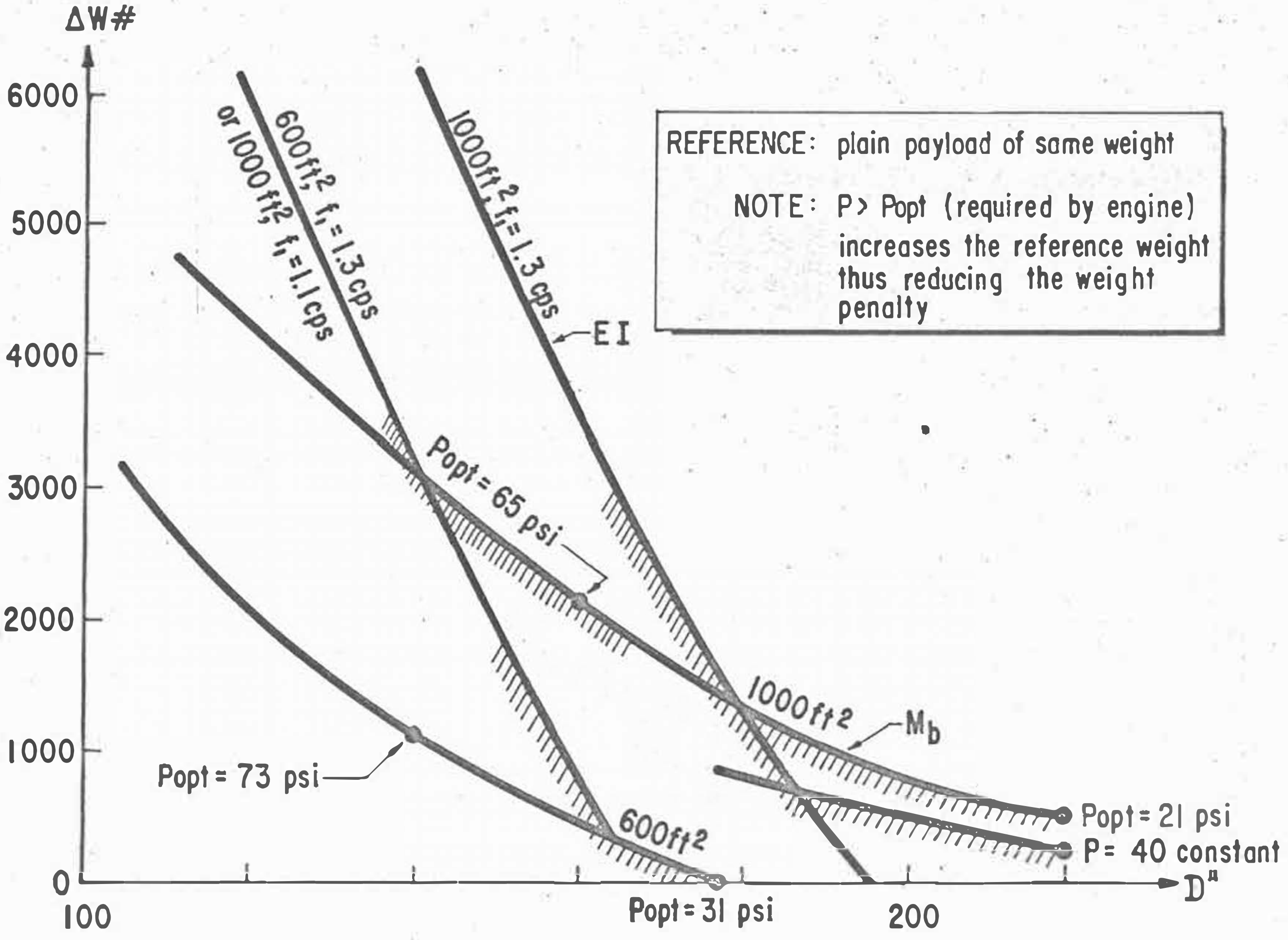
It is interesting to note that the "M_b" curves drop more slowly with increasing diameters than the EI-curves, so that with $D > 180$ inches the strength requirement alone is important.

If a diameter of 120 inches is selected for the upper stages, a 600 ft^2 winged payload would necessitate reinforcing the second stage with approximately 4000 pounds of material and the third stage with approximately 1900 pounds, which deducts from the usable payload. This large weight addition means a complete redesign of the ATLAS or TITAN airframes if either are used as a second stage. This is not necessarily the most economical way to obtain the needed rigidity.

b. Rigid Body Control Considerations. The requirements for control action and the angles of attack experienced are usually critical during the period of maximum dynamic pressure during the flight of the first stage.

WEIGHT PENALTY IMPOSED ON UPPER STAGE BY WINGED PAYLOADS

DUE TO: (1) STRENGTH REQUIREMENTS (2) STIFFNESS REQUIREMENT



REFERENCE: plain payload of same weight

NOTE: P > Popt (required by engine) increases the reference weight thus reducing the weight penalty

FIG. 25 GE140-49-59
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The assumption was made that a wind of 64 m/sec (which may be roughly associated with a 5% probability) would be encountered at that level with a maximum gradient over height of $0.02 \ \frac{\text{m/sec}}{\text{m}}$. Furthermore, it was assumed that the control system would be adapted to the principle of minimum drift which requires a ratio of the gain factors for acceleration to those for attitude input such as to produce vanishing lateral forces at equilibrium regardless of the wind. The unwinged B-1 vehicle of a 220-inch diameter would experience the following angles of attack and peak engine deflection angles as a function of the undamped control frequency for the specified wind gradient and at the time of maximum stagnation pressure points:

frequency (cps)	0.05	0.1	0.2	0.3
β maximum°	3.0°	2.5°	2.3°	2.2°
α maximum°	6.9°	5.8°	5.4°	5.2°

Stability parameters at the critical time are as follows:

$\frac{CP}{D} = 6.0$

$C_{z\alpha} = 2.28$/Radian (Normal force coefficient slope)

$\frac{CG}{D} = 3.5$

$$C_1/C_2 = -0.45 \quad \frac{\text{Aerodynamic restoring coefficient}}{\text{Control force coefficient}}$$

It is interesting to note that β maximum does not increase much with decreasing control frequency down to about 0.1 cps, permitting an eventual decrease in bending frequency. These values refer to nominal conditions excluding the wind disturbance. In addition to this investigation, a random deviation study was made assuming the following errors in the unfavorable direction:

$$\Delta \frac{CP}{D} = \Delta \frac{CG}{D} = 0.1$$

$$\Delta C_z = 5\%$$

Moment of inertia increment $\Delta \theta = 10\%$

The geometric sum of these random deviations from nominal values resulted in a maximum required swivel angle of 0.6° at 0.1 cps.

The corresponding values of β maximum for 192-inch diameter upper stages with identical propellant loadings are about 83% of those for the 220-inch version; i.e., $\beta_{max} = 2.6°$ versus $(2.5 + 0.6)° = 3.1°$ at 0.1 cps control frequency.

Both values are well within the 7° available, if all control engines are working. In case of one engine out, the required control angles increase, especially if the engine failure occurs early and leads to a rather large deviation from the standard trajectory. The torque resulting from engine failure critically depends upon the distance between the center of gravity and thrust line (cant angle). If the engine cant angle is selected so as to point through the missile center of gravity at q_{max}, the increase in control deflection through engine failure can be kept small. Thus, it appears that, although rigid body control favors a lower diameter, the difference is not large enough to be an important consideration, particularly for the unwinged payload.

The situation for winged payloads was also studied to some extent. A triangular wing of 1000 ft² planform shifts the CP forward by 1.5 D; the lift slope is increased by more than 100%.

We then have:

$\frac{CP}{D} = 7.5$ $\quad$ $C_{z_\alpha} = 5.4/\text{Radian}$

$\frac{CG}{D} = 3.5$ $\quad$ $C_1/C_2 = -1.5$

If the same control principle is applied, this configuration would then experience the following values for the same wind conditions as before (standard condition)

control frequency (cps)	0.1	0.15	0.2
β max °	6.3	5.4	5.0
α max °	4.3	3.6	3.5

Considering the need for additional allowances for nonstandard conditions, particularly the engine-out situation, these values appear too high for safe control. In the engine-out condition, the required control deflections rise rapidly unless a fairly high control frequency is selected. Improvements of this situation can be affected by the following different means, either separately or jointly:

(1) Modifying the gain factors from the minimum drift principle so as to make the vehicle head into the wind like a weather-cocking vehicle. This change is effective only for high control frequencies. At 0.2 cps control frequency β_{max} could be reduced from 5° to 2°, whereas for 0.1 cps the reduction would be insignificant. This results from the sluggish response of the controlled vehicle in the face of rising wind.

(2) Increasing the maximum engine deflection from 7° to 10°. While this is not provided for in the present missile, it would not be too difficult to provide and could be considered as a moderate modification.

(3) Placing a pair of 500 ft fins at the rear of the booster in the same circumferential location as the wings, would reduce the ratio C_1/C_2 from -1.5 to -0.45, which is the value pertaining to the unwinged vehicle, for a fin size of 500 ft^2. Doubling the fins to 1000 ft^2 would produce a neutrally stable configuration. A rough estimate places the weight of such fins around 3000 pounds, including mounting rings and attachments.

It can be expected that the introduction of any two of these items will probably produce a satisfactory solution; (1) and (2) probably being the simplest changes. The combined use of all three improvements appears to give a wide margin of safety.

c. Aeroelastic Control-Feedback Stability. As mentioned earlier, it is very desirable to have a spread between the control frequency and lowest bending frequency of at least 1 to 5 if not more. The closer these frequencies are together, the higher the sensitivity of the system will be to deviations from standard conditions, rapidly leading to instable modes caused by interaction of control, bending, and sloshing. Factors such as bending mode shape, dislocation of accelerometers during bending, bending frequencies, transfer characteristics of control sensors as well as servomotors, mass distribution, compliance of servolinkage, damping of feed motors, etc. enter into the problem. The complexity of shaping networks to meet stability requirements increases as the difference between those frequencies decrease, and for a spread smaller than 1 to 5, the need for variable shaping network may be introduced. Detailed calculations for the previous SATURN configuration indicate this trend, but due to their complexity and the time required, extrapolations to the new configurations have not been made to date.

In view of the estimated bending frequencies quoted previously and the requirements for rigid body control frequencies, it appears that an upper stage diameter of 220 inches is required to carry large winged payloads (approximately 1000 sq ft wing area). Although it is quite marginal, a diameter of about 160 inches might be acceptable from a pure control and stability viewpoint for unwinged payloads only. Such a configuration is not very attractive; however, for several other reasons mentioned in this chapter. Even for the unwinged version, the 220-inch diameter appears to be the most desirable choice in view of its beneficial effect upon bending frequencies,outweighing the slight disadvantage upon rigid body stability.

d. Transportation and Handling. Several methods of transportation are available for moving stages with diameters larger than 120 to 160 inches. Water transportation is the only method which place practically no limitations on diameter. Presently available airships could handle a 220-inch second and third stage with only minor modifications. Several studies have been conducted on road clearances to determine the ground transportability of the larger diameters. The results indicate that a 220-inch diameter can be transported by road over limited distances if power lines and other nonpermanent obstacles are cleared and the route is carefully chosen. Further details are given in paragraph E. 4. Rail transportation of stages with diameters in excess of 120 inches is considered feasible.

e. Cost Considerations. Increasing the upper stage diameter above 120 inches will require certain modifications in tooling and test facilities, raising the cost of the program. It should be pointed out, however, that some additional cost will be incurred in modifying the existing 120-inch tooling to meet the requirements of the SATURN design. The overall R&D program cost for the 220-inch diameter upper stage configuration will be approximately 10% to 20% higher than a 120-inch configuration, depending on whether one or two prime contractors are chosen for the development and production. This increased cost is appreciably offset by the additional payload capability and mission flexibility of the B-1 vehicle. One criteria for determining the economy of a space transportation vehicle is the cost to deliver a pound of payload into orbit. For the 120-inch diameter configuration the cost per pound is $556 and for the 220-inch diameter configuration, $469.

f. Orbital Refueling. Orbital refueling of complete stages transported into orbit is the simplest way of increasing the payload capability of vehicles going to the lunar and planetary surfaces by an order of magnitude. This also minimizes or delays the need for the development of a much larger booster vehicle which might approach the upper limits of feasibility to produce the same capability. Orbital refueling based on standard SATURN B-1 vehicles as tankers offers the opportunity for an early manned lunar landing and return. The orbital refueling technique does require, however, large diameters in the upper stages to provide the basic tankage for the orbital-launched vehicle. If the diameter is small, the slenderness ratio will exceed tolerable limits. Diameters in excess of 220 inches are highly desirable, while those much less than 200 inches are not acceptable if orbital refueling missions are considered.

g. Overall Vehicle Length. The presently planned launching service tower has a hook height of 245 feet above ground and can accomodate a vehicle of about 215 feet, using the present design of a 30 foot-high launch platform. Thus a vehicle using a 120- to 160-inch diameter for the upper stages and designed for maximum performance could not be accomodated without changes in the service tower now under construction. An additional length of 20 feet can later be incorporated for an additional cost of about $200,000.

In general a short and reasonably stubby vehicle is easier to handle and will tend to reduce the cost of ground support equipment.

h. Mission Flexibility. Larger diameters are less sensitive to changes in the payload weight or wing surface as pointed out earlier; therefore, relatively late changes in payload and mission assignments appear feasible, and lead times could be reduced. Large tank capacities somewhat above minimum requirements are feasible for large diameters but not for small ones due to control limitations. The large diameter therefore makes it possible to design a standard SATURN configuration and standard maximum propellant capacities allowing optimum propellant loadings for individual missions and slight variations from flight to flight. This flexibility always allows maximum performance for each mission at no additional cost.

i. Growth Potential. Large stage diameters make it feasible to lengthen the tanks considerably if more thrust in the booster or higher thrust H_2/O_2 engines are available. This would allow the change to high-energy propellants in all stages without major tank modifications. Large payload increases can be expected by such a change.

j. Summary. While it is not possible at this time to specify an exact minimum diameter requirement for the SATURN upper stages, it can be safely stated that the selection of 220 inches for the second and third stages will result in a vehicle which can be developed with a maximum of reliability, mission flexibility, and growth potential. As long as winged payload configurations are considered, diameters from about 120 to 160 inches appear marginal at best for low wing areas. For large wing areas (1000 sq ft) the 220-inch is almost mandatory. The large diameter is also more desirable for the unwinged-version. The cost will probably not be substantially higher than for the other diameters above 120 inches, and may become even lower in view of the increased reliability.

A 160-inch diameter in the second and third stage might be an acceptable interim solution if winged payloads larger than 500 sq ft are ruled out and if growth potential is not important. This must be considered definitely as a poorer solution.

2. Propulsion Systems

a. Second Stage. The second-stage propulsion system is based on a 180 to 220K altitude-thrust engine which is available now, either from Rocketdyne (H-1) or Aerojet (XLR-87). Both engines require modifications, including the incorporation of a high-altitude ignition system and, as a "nice to have" change, an extension of the nozzle for a high altitude expansion ratio. It can be shown that a performance advantage can be obtained with expansion ratios of up to 1:25; however, practical reasons might force the use of 1:20 or even 1:16 ratios.

The question of which engine (Rocketdyne or Aerojet) is superior has been studied in great detail. Such parameters as schedule, engineering support, facilities, experience in altitude testing, ground- and flight-test experience, performance, ground support equipment, packaging, hardware similarity, maturity and reliability, growth potential, logistics, cost,and other miscellaneous data have been evaluated and compared with great care. It was concluded that the engines are comparable in the above areas and the major point of consideration should be the familiarity of the selected stage contractor with their respective engine. The experience factor counts heavily for a finished integrated product. Thus the recommendation is that the stage contractor selected for the SATURN program should use the engine he is most familiar with,making use of established working relationships. This means that the engine would be the modified H-1 engine if Convair is selected as a second-stage contractor and the YLR-87 if Martin-Denver is selected.

The follow-on program envisions either keeping the present engines, or replacing them later with a 150K hydrogen/oxygen engine which may become available by 1965, depending on the selected configuration.

b. Third Stage. The third stage uses either two or four Pratt & Whitney XRL-115 hydrogen/oxygen engines which are rated at 15K each at the present time. The four-engine configuration is more advantageous because it promises considerable performance gains for most missions. The engine can be uprated easily to 20K since it was designed with the proper performance margins. A qualification test program at the new rating and possibly at a manned-flight rating would have to be initiated and would cost in the order of $10 to $15 million, depending on the degree of reliability required.

The follow-on program would require the introduction of the 150K hydrogen/oxygen engine in a double-barrel configuration for third stage application. This is most advantageous because the low thrust level in the third stage is the key to further performance increase. It would permit a noticeable rise in the percentage of high-energy propellants in the SATURN.

c. Fourth Stage. The fourth stage is optional for high speed missions. The standard CENTAUR stage with two Pratt & Whitney XRL-115 is well suited, with minor changes, for such an application.

The follow-on program envisions replacing the fourth stage (2 X 15 or 20K) with the B-1 third stage (4 X 20K) which has the 220-inch diameter. This requires no changes in the propulsion system.

d. Propellant Utilization. Preliminary studies have shown that a PU system in the booster stage will definitely not pay off because the eight engines will have mixture ratio tolerances to both sides, compensating each other. Also a small difference in residuals does not appreciably influence the payload capability. In case of a four-engine configuration, the second stage does not need a PU system. For a three-stage configuration, it is desirable to keep any PU provisions which are already incorporated in the engine, but the development of new components or systems does not seem to be justified.

The third and the fourth stages using the CENTAUR engine will have PU systems incorporated. The mixture ratio control is accomplished by controlling the booster pumps in the tanks, which determine the pump inlet pressures. PU control, if available, will pay off from the performance point of view for the third and fourth stages. Summarizing, PU control in the first two stages of SATURN is not required, but it will increase the performance if used in the third and fourth stages.

E. VEHICLE SYSTEM

1. Instrumentation

A total of 2000 pounds was used in all calculations as the full instrument compartment weight, which includes guidance and control components, measuring instrumentation, air supply, temperature control, and all the structural components that make up the total instrument compartment. Table 7 is a typical weight breakdown of the equipment expected to be in the instrument compartment. The weight of the primary power supply and the air supply will vary with the mission because of differences in operational times. Estimated power requirements also have been included in the table. The total weight of this equipment adds up to approximately 800 pounds for a low altitude mission and about 1100 pounds for the 24-hour mission. That would leave approximately 1200 pounds, or 900 pounds respectively, for the basic structure of the compartment, brackets, and any other equipment which might be required. This is considered to be a conservative assumption.

2. Payload Compartment

For all initial SATURN vehicles, the standard payload compartment is a 120-inch cylinder with a 60 degree nose cone. The weight of the payload compartment itself should be approximately 5% of the actual payload, and according to definition, it is included in the listed net payload weights. In the follow-on program the payload compartments are not necessarily limited to the 120-inch diameter and would allow the accomodation of large volume payloads. This is even possible for the three-stage version of the B-1, whose payload container could be as large as 220 inches. Mean specific gravities of typical payloads should be in the area of 0.25 to 0.35 to keep the total length of the payload compartment within reasonable tolerances.

Table 7

INSTRUMENTATION, GUIDANCE AND CONTROL EQUIPMENT IN THE INSTRUMENT COMPARTMENT OF PAYLOAD STAGE

Unit	Number of Operations	Weight, lb	Power Consumption, (DC), watts	Power Consumption, (AC), watts	Volume, cu in.
ST-90 Stabilized Platform Servo Amplifier Base and Tilt Frame Assembly	1	195		240	6000
Control Computer	1	35		100	
Guidance Computer	1	40		150	
Electronic Timer	1	3		3	180
Actuators, Hydraulic	4	20			160
Reaction Nozzles	8	8			40
Flight Sequence	1	2	11		60
Distributor, Control	1	25			1200
Distributor, Power	1	25			600
Distributor Measuring	1	25			1200
Primary Power Source, Based on 6 Hours Time (Maximum) Batteries					
300-Mile Orbit	1	50			
Soft Lunar Landing	1	75			
24-Hour Orbit	1	200			7000
Measuring Sensors	50	50	25		1800
Measuring Voltage Supply	1	4	1		90
Cable		125			3000
Radio Command Receivers	2	6	2		400
C Band Radar Beacon	1	20	50		480
Telemetry System	1	8	15		150
UDOP Transponder	1	30	120		840
Static Inverter	1	20	Input 940 at maximum load	Output 750	600
Relays and Heaters		50	250		600
Air Supply		50 to 200			43

3. Stage Separation

Stage separation was studied in detail. The problem is greatly influenced by the dynamic pressure at the moment of separation, and the shape and weight of the payload. The performance optimization, with respect to propellant loadings, indicates that the separation would occur at about 50% of q_{max}, which is too high for large winged payloads.

Two basic separation systems have been studied specifically for the most critical payload, which is a vehicle with a wing area of about 1000 sq ft. One system studied was the "firing in the hole" concept; the other one, retro rockets on the booster and ullage rockets on the second stage. There are many advantages and disadvantages to be weighed against each other before a final recommendation can be made. From the weight standpoint, the separation with retro and ullage rockets seems to be more favorable.

4. Transportation

Various studies, by different groups, have been made concerning the transportation of large containers. A summary of these studies show that rail transportation, as well as conventional aircraft transportation, of containers of more than 120 inches in diameter is not feasible during the early development of SATURN. Blimps might be used for air transportation of large diameter containers, but this requires a substantial development effort and would be relatively expensive for low firing rates. The most attractive mode of transportation is by water and requires road transportation between the manufacturing site and the nearest dock allowing waterway accessibility. This does not seem to be a major problem for any company located close to the coastal area; however, it becomes marginal for manufacturing sites in the interior. The Martin Company has made a study (Ref. 5) which concludes that a vehicle section in the range of 220 inches in diameter is the largest that can be feasibly transported over the routes studied. To accomplish this, however, an extremely thorough and detailed plan must be worked out in advance, with regard to police escorts, power and telephone line elevation, trimming of low tree limbs and overhanging foliage, state of highway repair, and snow conditions since the weather could be a limiting factor during the winter months. It is too early to state at this time how expensive such changes on the selected route would be; however, it should be in the order of $500,000 to $1,000,000, which still might be the cheapest way to solve this problem. The following additional statements can be made with respect to the individual routes:

a. Denver to Redstone Arsenal. The problem of transporting a large item such as the upper stages of the B-1 configuration has been preliminarily investigated. Longitudinal dimensions up to approximately 50 feet do not impose limitations since there are existing trailers of this length. A preliminary report was published by the Martin Company of Denver, Colorado, in which they investigated routes from Denver to either St. Joseph, Kansas City, Hannibal, or St. Louis. From the results presented in this report, it appears that the St. Joseph route is optimum, as there are less obstacles and easier by-pass methods on this particular route.

It is felt that the earliest contact that can be made with water transportation is the most desirable approach. Basically, once the missile component and transporter has reached a water port, then the only limitations are the dock dimensions, which do not present a problem with components of this size. Once the missile component has reached the Missouri River at St. Joseph, it would be barge-loaded for transportation by the Missouri, Mississippi, Ohio, and Tennessee Rivers to the Redstone Arsenal docks. Transportation from Redstone Arsenal to the Atlantic Missile Range (AMR) will be accomplished by water in an identical manner to the first stage; i.e., Tennessee, Ohio, and Mississippi Rivers and the Gulf of Mexico and Intercostal Waterway.

b. San Diego to Redstone Arsenal. It is felt that overland transit of a missile component with the dimensions of the upper stages of the B-1 configuration would be prohibitive cost-wise due to the myraid highway obstructions and low clearance obstacles. As a result of the highway problem and the availability of San Diego port facilities, transportation by a water route seems to be the most reasonable approach. This would necessitate approximately [illegible] miles of sea and 1500 miles of inland waterway travel.

Initially, the missile and transporter could be placed aboard a seagoing vessel of suitable capacity. The vessel would then proceed along the West Coast of the United States, Mexico, and Central America, pass through the Panama Canal and on to New Orleans. Here, the missile component would be placed aboard a barge for transportation to Redstone Arsenal via inland waterways. Transportation from Redstone Arsenal to AMR will be accomplished as mentioned above.

c. General. It should be mentioned that 14 feet is the normal highway clearance, and any protrusions beyond this height require detail planning and coordination. With increasing height, of course, the cost increases considerably.

Air and rail methods of shipping the large components were considered. It is not possible to air transport any missile component exceeding 120 inches in diameter and approximately 50 feet in length with present day aircraft, nor is it likely that such will be available within the next three years. Blimp transportation is possible,but this is an entirely new field. No currently existing blimp, without modification, has the capability to carry the B-1 upper stages. A preliminary report has been received within the last year by this Agency on the development of a super blimp capable of transporting all stages including the booster. Since the development cost of this scheme is approximately $17.5 million, it has not been given furthe consideration to date.

Consideration is presently being given to the feasibility of having small shuttle barges that could operate between St. Joseph and Redstone Arsenal and New Orleans and Redstone Arsenal. These barges would have provisions,similar to those of the SATURN booster barge,to accommodate the missile components.

It is possible that the present concept for the SATURN booster barge could be modified and that the barge could be increased in length to accommodate the first, second, and third stages, so that a complete missile, once checkout had been completed, could be shipped from Redstone Arsenal to AMR. Cost figures on such a barge are not available at this time.

5. Required Facilities

A survey has been made on the additional facilities required at Redstone Arsenal for the accomplishment of the SATURN development program, followed by an operational program with a firing rate of six vehicles per year.

Table 8 summarizes the facilities required at ABMA and AMR. They are broken down into the R&D program and the operationa program by fiscal year.

The facilities required at the upper stage contractor site have not been clearly defined to date. Convair has sufficient floor

Table 8

FACILITIES REQUIRED FOR SATURN PROGRAM

Facility Description	FY 1959 and FY 1960	FY 1961	FY 1962	FY 1963	Total
R&D					
Launch - Blockhouse	2.328				2.328
Pad and Area Development	6.556	0.120			6.676
Service Structure	4.600				4.600
Staging Building	0.450	0.600			1.050
Capital Equipment	0.372	0.712	0.100		1.184
UDOP Sites		0.090			0.090
Static Test Tower - Construction	1.642	7.500			9.142
Instrumentation and Equipment	0.981	3.300			4.281
Pressure Test Cell	0.530				0.530
Transportation (Barge, Docks, Dredging)	1.316	1.140	0.470	0.400	3.326
Hydrostatic Test Tower		0.600			0.600
Additional Manufacturing Facilities		2.450			2.450
Additional Inspection and Reliability Facilities		5.860	2.030		7.890
Minor Construction	0.678	1.000			1.678
TOTAL R&D FACILITIES	19.453	23.372	2.600	0.400	45.825
Operational					
Launch Facility - Pad and Area Development			0.250	7.250	7.500
Service Structure			0.250	9.750	10.000
Blockhouse Expansion			0.250	0.750	1.000
Capital Equipment			0.050	0.500	0.550
High Pressure Test Cell			0.060	1.000	1.060
Transportation				1.400	1.400
Hydrostatic Test Tower			0.020	0.350	0.370
Additional Manufacturing Facilities			0.115	1.850	1.965
TOTAL OPERATIONAL FACILITIES			0.995	22.850	23.845
TOTAL R&D AND OPERATIONAL FACILITIES	19.453	23.372	3.595	23.250	69.670

space and requires only a blockhouse and one or two test stands, depending on whether one or two upper stages will be tested there. This new test complex can be added to an existing test area. Martin would require the same test complex and additional checkout facilities. Additional floor space might be required, depending on the existing work load at that time period.

The cost for all the facilities required have been included in the program cost estimates discussed in more detail later.

F. MISSION AND PAYLOAD CAPABILITIES

1. Mission Spectrum

The SATURN is expected to be the only large space carrier-vehicle available to the United States from 1963 to about 1970. This requires that the basic vehicle be able to perform all missions of interest with a minimum number of changes. The following list is a summary of missions for which the SATURN could be used.

a. Orbital Missions

(1) Instrumented satellites

(2) Manned recoverable orbital space vehicles

(3) Manned engineering and scientific research satellites

(4) Orbital supply vehicles

(5) Communication satellites

(6) Astronomical satellites

(7) Navigational satellites

b. Lunar Missions

(1) Lunar TV satellites

(2) Instrumented and manned lunar circumnavigation

(3) Stationary and non-stationary lunar soft landing vehicles

(4) Permanent lunar satellite relay station

(5) Manned lunar landing and return via orbital refueling

(6) Direct supply vehicle for lunar base

c. Space Probes

(1) Solar satellites

(2) Direct solar probe

(3) Solar system escape communication probe

d. Planetary Missions (All Planets of Interest)

(1) Planetary satellites

(2) Planetary semi-hard landing vehicles

(3) Planetary soft landing vehicles

(4) Permanent planetary relay and surveillance stations

(5) Supply vehicle for planetary bases via orbital refueling

2. Typical Vehicle Weight Breakdown

Weight breakdowns have been prepared, for the purpose of performance estimates, on all vehicles under consideration. These weights must be considered preliminary, due to the limited manpower available for this effort. It is felt, however, that the weight breakdowns obtained for the B, B-1, and B-3 versions are fairly accurate since they are based on rather detailed design studies. Therefore, they are listed in this report as representative figures of the SATURN space vehicles. It should be kept in mind that the basic design approach is conservative and no major effort was made to reduce weights; it is considered unwise to attempt to improve the ample performance by weight shaving with a corresponding reduction of reliability in a costly vehicle like the SATURN.

Tables 9, 10, and 11 are summaries of the weight and propulsion data used for performance calculations. It should be noted that they represent typical configurations of B, B-1, and B-3

Table 9

SATURN B

SUMMARY WEIGHT AND PROPULSION DATA (24-HOUR ORBIT)

Stage	I	II	III
Engine	H-1	LR-87	LR-115
Propellant	Lox/RP-1	Lox/RP-1	Lox/LH
Thrust, lb	8 × 188K	2 × 181.5K	2 × 15K
Adjusted Thrust			
(Sea Level)	1,498,850		
(Vacuum)	1,691,000	363,000	30,000
I_{sp} (Sea Level)	257.0		
(Vacuum)	289.9	305	412
Flow Rate, lb/sec	5852.140	1190.164	72.186
Adjusted Exit Area, in.^{2}	13,075		
Missile Diameter, in.	257	160	120
$W_{11, 15}$, Payload, lb			5,100*
W_{16}, Guidance Compartment, lb	500		500
W_2, Guidance and Control, lb	1,100	500	1,500
W_3, Fuselage, lb	45,000	5,967	1,923
W_4, Propulsion, lb	22,400	4,692	1,127
W_5, Recovery Equipment, lb	6,000		
W_6, Trapped Propellants, lb	15,500	1,167	200
W_7, Usable Residuals, lb	7,047	2,200	MRS 500 FPR 1130
W_8, Propellant Consumption, lb	697,637	217,800	47,120
W_s, Structure Weight, lb	75,000	11,159	5,050
W_n, Stage Burnout Weight, lb	97,547	14,526	6,880
W_a, Stage Weight, lb	795,184	232,326	54,000
W_o, Lift-Off Weight, lb	1,086,610	291,426	59,100
W_c, Cutoff Weight, lb	388,973	73,626	11,980
r, Mass Ratio	2.7935	3.9582	4.9332
Δu, Characteristic Velocity, m/sec	2784	4115	6449
F_o/W_o	1.379	1.246	0.508
F_c/W_c	4.347	4.930	2.504

Vehicle Characteristic Velocity = 13,348 meters per second.

*Nominal payload for which velocity requirement was not met in this specific case.

Table 10

SATURN B-1

SUMMARY WEIGHT AND PROPULSION DATA (24-HOUR ORBIT)

Stage	I	II	III	IV
Engine	H-1	LR-87	LR-115	LR-115
Propellant	Lox/RP	Lox/RP	Lox/LH	Lox/LH
Thrust, lb	8 × 188K	4 × 220K	4 × 20K	2 × 15K
Adjusted Thrust, lb	1,498,850 (SL)	880,000 (Vac)	80,000 (Vac)	30,000 (Vac)
I_{sp}, sec	257 (SL)	312 (Vac)	420 (Vac)	420 (Vac)
Flow Rate, lb/sec	5852.140	2820.513	190.476	71.429
Exit Area, m^2	8.5634			
Missile Diameter, in.	257	220	220	120
$W_{11,15}$, Payload, lb				7800
W_{16}, Guidance Compartment, lb				500
W_2, Guidance and Control, lb	2000	500		1500
W_3, Fuselage, lb	45,623	12,315	3800	1307
W_4, Propulsion, lb	22,000	10,000	1850	990
W_5, Recovery Equipment, lb	10,000			
W_6, Trapped Propellants, lb	15,000	6800	1000	582
W_7, Usable Residuals, lb	2852	3419	345 MRS 3125 FPR	400 FPR
W_8, Propellant Consumption, lb	567,438	338,962	68,697	24,157
W_S, Structure Weight, lb	79,623	22,815	5650	4297
W_n, Stage Burnout Weight, lb	97,475	33,034	10,120	5279
W_a, Stage Weight, lb	664,913	371,996,	78,817	29,436
W_O, Lift-Off Weight, lb	1,152,962	488,049	116,053	37,236
W_C, Cutoff Weight, lb	585,524	149,087	47,356	13,079
r, Mass Ratio	1.9691	3.2736	2.4255	2.847
Δu, Characteristic Velocity, m/sec	1846	3620	3646	4303
F_O/W_O	1.300	1.803	0.6965	0.8057
F_{vac}/W_C	2.893	5.903	1.6895	2.294
Vehicle Characteristic Velocity = 13,415 m/sec				

Table 11

SATURN B-3

SUMMARY WEIGHT AND PROPULSION DATA (24-HOUR ORBIT)

Stage	I	II	III	IV
Engine	H-2	LR-87	P&W	LR-115
Propellant	Lox/RP-1	Lox/RP-1	Lox/LH	Lox/LH
Thrust, lb	8 × 250K	4 × 220K	2 × 150K	4 × 20K
Adjusted Thrust				
(Sea Level)	1,993,150			
(60,000 feet)	2,171,704			
(Vacuum)	2,185,296	880,000	300,000	80,000
I_{sp} (Sea Level)	266.1			
(60,000 feet)	289.9			
(Vacuum)	291.7	312	420	420
Flow Rate, lb/sec	7515.968	2820.513	714.286	190.476
Adjusted Exit Area, in.^{2}	13,075			
Missile Diameter, in.	257	220	220	220
$W_{11, 15}$, Payload, lb	788,810	315,440	107,610	18,300*
W_{16}, Guidance Compartment, lb				500
W_2, Guidance and Control, lb	2,000	500		1,500
W_3, Fuselage, lb	48,000	15,500	7,880	3,740
W_4, Propulsion, lb	22,000	10,150	6,700	2,100
W_5, Recovery Equipment, lb	10,000			
W_6, Trapped Propellants, lb	15,000	7,870	2,300	1,070
W_7, Usable Residuals, lb	3,250	4,350	950	MRS 400 FPR 2740
W_8, Propellant Consumption, lb	650,000	435,000	190,000	77,260
W_s, Structure Weight, lb	82,000	26,150	14,580	7,840
W_n, Stage Burnout Weight, lb	100,250	38,370	17,830	12,050
W_a, Stage Weight, lb	750,250	473,370	207,830	89,310
W_o, Lift-Off Weight, lb	1,539,060	788,810	315,440	107,610
W_c, Cutoff Weight, lb	889,060	353,810	125,440	30,350
r, Mass Ratio	1.7311	2.2295	2.5147	2.5456
Δu, Characteristic Velocity, m/sec	1518	2453	3798	5213
F_o/W_o	1.295	1.116	0.951	0.743
F_c/W_c	2.443	2.487	2.392	2.636
Vehicle Characteristic Velocity = 12,982 meters per second.				

respectively, for one payload weight and the 24-hour orbit dogleg mission from AMR.

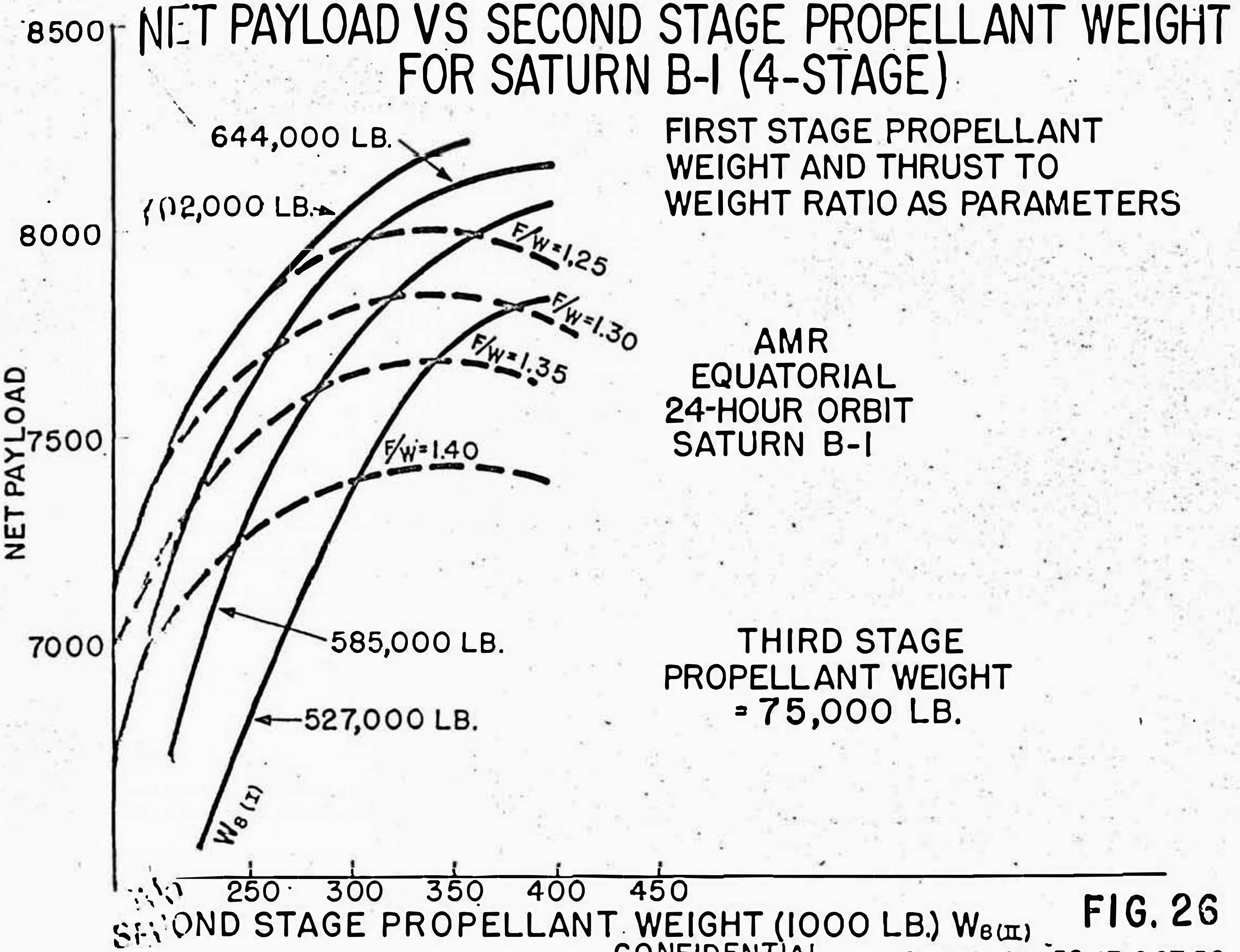
3. Payload Capabilities

a. Trajectories. A performance study was conducted for the SATURN B-1 and B-3 vehicles. Trajectory data were calculated for various propellant loading combinations, and the performance was evaluated and optimized to determine the optimum propellant weight in each stage and the most desirable trajectory shape. The 96-minute orbital mission, the escape mission, and the 24-hour equatorial mission were considered in these studies, with all launchings from AMR.

Various trajectory shapes were considered for the various missions. For the 96-minute orbital mission, a three-stage vehicle was assumed, with perigee injection for the necessary kick at the apogee of the Hohmann transfer ellipse. The perigee injection conditions were a flight path angle of 90° with the local vertical, an altitude of 100 statute miles, and a velocity of 7929 m/sec (26,014 ft/sec). The apogee velocity of the transfer ellipse was 7464 m/sec (24,488 ft/sec) at an altitude of 568 km (353 statute miles or 307 nautical miles). After the apogee kick, a circular velocity of 8045 m/sec (26,394 ft/sec) was attained.

The trajectory sequence for the 24-hour equatorial orbital mission is somewhat different from that of the 96-minute orbit. Investigations were conducted for both the three-stage and four-stage versions of the SATURN. The principal difference between the 24-hour equatorial and the 96-minute orbit is the necessity for a coasting or parking orbit. For the three-stage vehicle, the third stage would be started a total of three times: first, after completion of the second stage; second, near the equator after a time in the parking orbit, which was assumed to be a circular orbit; and third, after going through the transfer ellipse for the apogee kick. Substantial improvement in performance was attained by the addition of the standard CENTAUR as a fourth stage. The first three stages were utilized for injection of the fourth stage and its payload into the parking orbit. This is attractive from the viewpoint of higher payload capability and also for operational reasons. The 96-minute orbit, three-stage vehicle is near optimum for parking orbit injection. The fourth stage is used for perigee and apogee kick, with one restart required.

For the escape mission, a trajectory shape similar to that described for the 24-hour orbit is used, except that the coasting



NET PAYLOAD VS SECOND STAGE PROPELLANT WEIGHT FOR SATURN B-1 (4-STAGE)

FIG. 26

c. Payload Capabilities of SATURN Vehicles for Direct Flight Missions. The payload capabilities obtained for the SATURN are summarized in Table 12 for selected earth and lunar missions and in Table 13 for planetary missions based on the escape payload given in Table 12.

Net payloads and gross payloads are listed. The net payload is defined as the actual payload plus the payload container, including all equipment required to operate the payload, such as power sources and control systems. The gross payload is defined as the net payload plus all payload shrouding required during the ascent trajectory, the instrument compartment containing all instrumentation, guidance,and control equipment required for injecting the payload into the desired trajectory or orbit, and the standard propellant residuals left in the last powered stage (equivalent to 3% of the total velocity requirement).

The payload capabilities quoted for the B version are based on a 160-inch diameter, two-engine second stage and a two-engine third stage (interim B). A four-stage SATURN B-1 configuration is not considered feasible for the low orbit (300 mile) mission and is not listed.

It can be seen from Table 12 that the low altitude orbit capability of the SATURN for a single flight can be anywhere between 27,000 and 90,000 pounds depending on the configuration selected. This growth potential is even more apparent if more demanding missions, like the 24-hour orbit or lunar soft landing are compared. The initial payload capabilities are increased by a factor of four.

It should be noted that the lunar soft landing capabilities are based on the use of high-energy propellants for the landing maneuver, an assumption which has to be verified. If lower specific impulse propulsion systems are required, a lower payload capability will result.

The relationship used to derive lunar soft landing capabilities from escape payload capabilities is shown in Fig. 27, which gives data for 300 seconds and 420 seconds specific impulse landing stages. The major assumptions made for the derivation of these curves are listed on this figure.

The difference in payload capabilities of the individual SATURN configurations is even more pronounced in planetary missions.

Table 12

PAYLOAD CAPABILITIES FOR SATURN VEHICLE CONFIGURATIONS

Vehicle	Number of Stages	96-Minute Orbit 307 NM	Escape	Equatorial 24-Hr Orbit from AMR	Lunar Soft Landing
B	3	27,000*	8,400	5,000	2,650
		31,500**	12,000	8,100	3,100
B-1	3	35,000	10,250	5,200	3,350
		40,100	14,100	8,650	3,850
	4	Not feasible	11,900	7,800	4,000
			15,500	10,200	4,550
B-2	4	55,200	16,000	8,500	5,300
		62,000	20,500	12,900	6,250
B-3	4	71,400	26,000	17,500	9,000
		78,000	31,200	22,000	10,300
B-4	4	58,500	21,400	14,000	7,350
		65,500	26,200	18,000	8,400
C	4	90,000	34,000	23,000	11,600
		99,000	39,500	28,000	13,300

* Net Payload

** Gross Payload

Table 13

SATURN CAPABILITIES FOR SELECTED PLANETARY MISSIONS

Vehicle	Payload, In Lb	Mars Satellite	Mars Soft Lander	Venus Satellite	Venus Soft Landers	Reference Escape
B	Net	933	1435	225	1350	8, 400
	Gross	1290	3755	595	4180	12, 000
B-1	Net	2365	3400	945	2830	11, 900
	Gross	2780	7240	1385	7410	15, 500
B-2	Net	3145	5000	1550	4775	16, 000
	Gross	3580	9140	2075	9575	20, 500
B-3	Net	5445	10, 465	2430	10, 705	26, 000
	Gross	6120	16, 020	3450	16, 645	31, 200
B-4	Net	4440	8055	1745	8605	21, 400
	Gross	5045	13, 860	2690	14, 045	26, 200
C	Net	7790	15, 825	3390	15, 220	34, 000
	Gross	8720	22, 725	4550	22, 880	39, 500

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LUNAR SOFT LANDING CAPABILITY

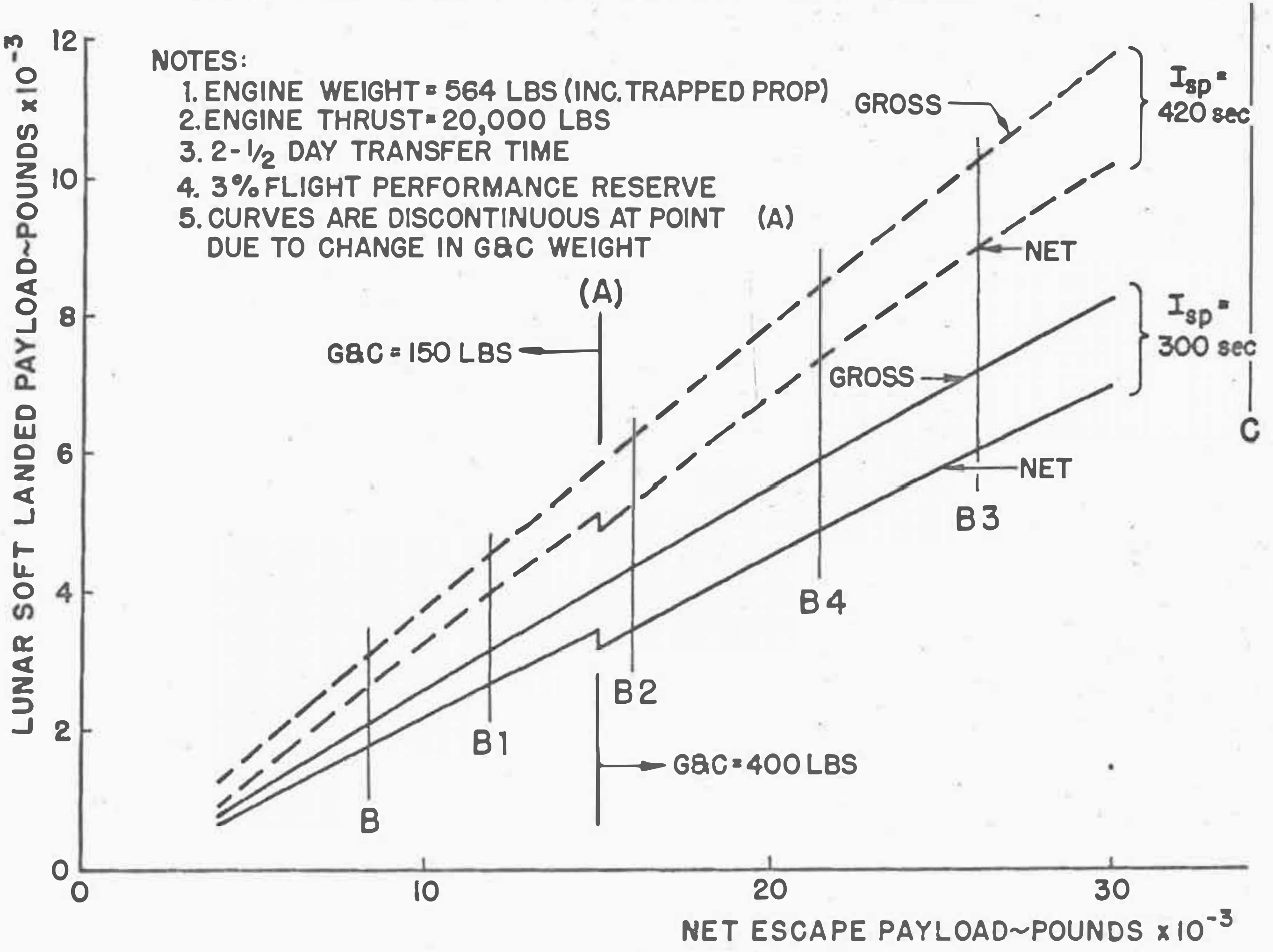

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FIG. 27

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In the most extreme case, the Venusian satellite, the net payload capability of the C vehicle is about 15 times that of the interim B version. The low value of 225 pounds net payload for the interim B version makes it questionable whether or not this configuration has a capability which is useful. Comparing B and B-1 only, it is very convincing that the B-1 is a much more desirable choice since its capabilities are 2 to 4 times better than those of the B for this type mission. This results in a considerably greater quantity of scientific information in one flight for approximately the same amount of money.

d. Engine-Out Influences on Performance From the Very Beginning. The SATURN booster has been designed for an engine-out capability; i.e., to continue its flight despite the failure of one engine. The loss of thrust in one engine will have two pronounced effects: first, it will increase the total burning time, increasing the velocity loss due to earth's gravity; and second, it will increase the trajectory angle slightly at cutoff. The resulting velocity losses vary with the individual missions, as well as with stage arrangement. The four-stage version using high-energy propellants in the third and fourth stages will be affected very little. A three-stage version for low altitude missions not using high energy in the upper stages will be affected to a higher degree.

To illustrate this point, two diagrams have been developed, one for a low altitude mission (Fig. 28) and one for the 24-hour mission (Fig. 29).

Figure 28 shows the payload losses expressed as percentages of the payload weight with no engine failures for a three-stage SATURN B-1 configuration performing a 96-minute orbit mission. The lowest straight line illustrates, as a function of the time that the failure occurs after lift-off, the losses incurred with one engine not operating. It shows that a 2 to 3.5% loss would occur if the engine failed during the first 30 to 60 seconds of flight time. This potential loss, in the present plan will be compensated (4 to 6 times for the case above) by the 3% velocity propellant reserve. Consequently, the calculated payload penalty, even in the unlikely case of one engine failing during take-off, is adequately covered by the velocity, or propellant reserves in the payload stage.

Figure 29 illustrates the same influence for a 24-hour mission of a B-1 configuration. In this case, the penalty can be even

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PAYLOAD PENALTIES AT ENGINE FAILURE (S)

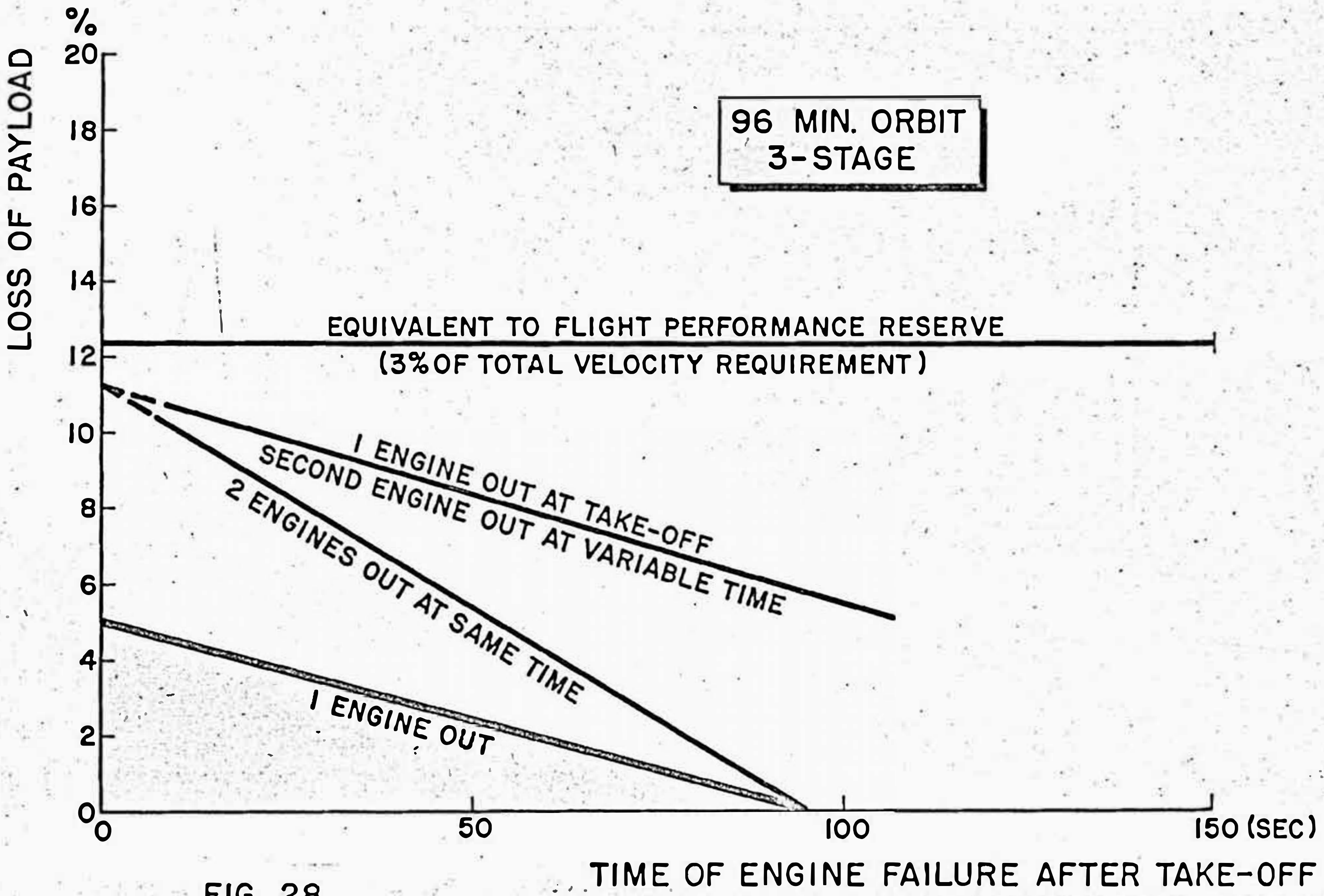

FIG. 28

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PAYLOAD PENALTY (ΔW_1) DUE TO CARRYING RESERVES VS THE PERCENTAGE OF SUCCESSFUL 24 hr SATELLITE MISSIONS INSURED

(WITH RESPECT TO 1st STAGE MOTOR FAILURE ONLY)

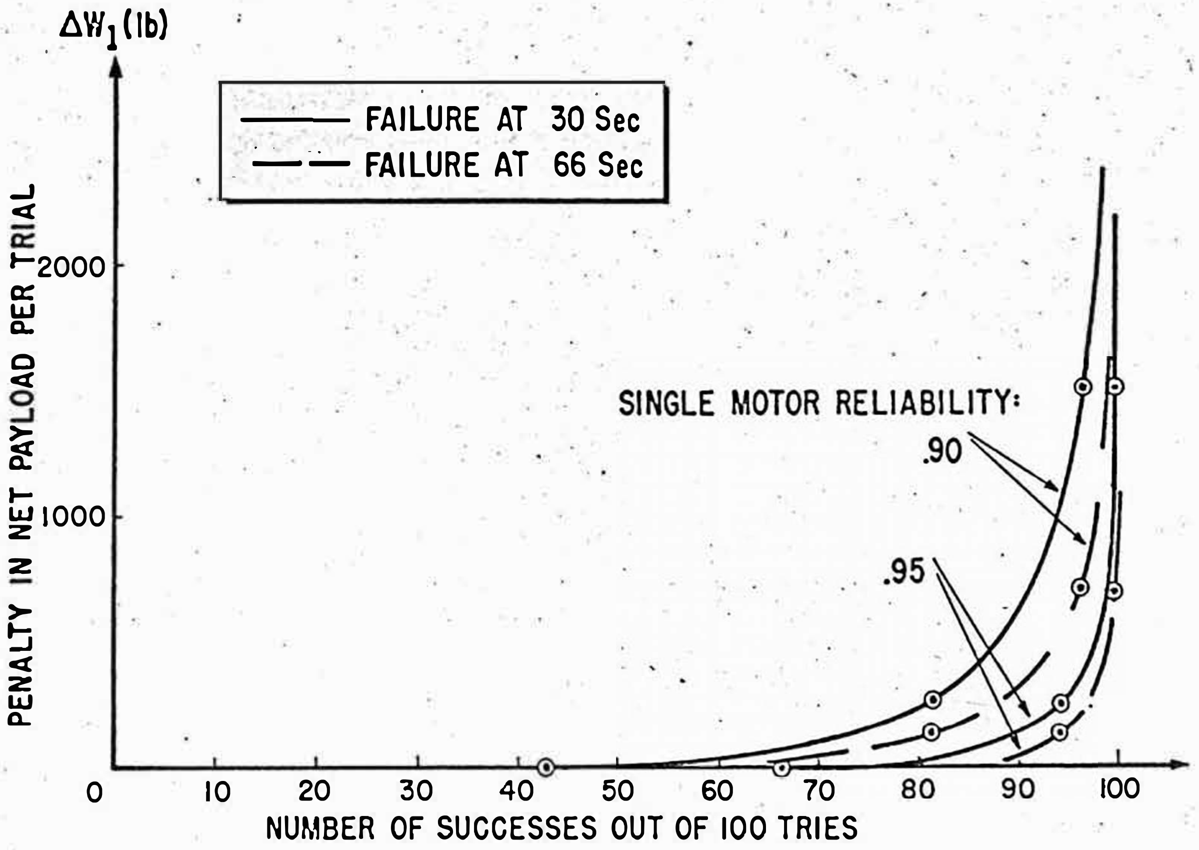

FIG. 29

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more easily absorbed (at least 4 times for one engine failure) by the flight performance reserve. The payload losses in this case are given in absolute figures and the engine failure time element is introduced as an additional variable. It was assumed that the failure would occur at 30 seconds and 66 seconds. The additional parameter of engine reliability was assumed to be 0.95 and 0.90 (the first one is closer to the expected reliability). This diagram shows that a failure occurring at 30 seconds (a very pessimistic assumption) would cause a 1200-pound payload reduction if a 95% probability for successful accomplishment of the mission is required.

These diagrams illustrate that the engine-out capability is not necessarily a loss of payload capability if proper increases are provided.

e. Payload Capabilities With Orbital Refueling. One means for increasing the single flight payload capability for missions requiring high velocities is by orbital refueling. The capability of this method is greatly reduced for the B version due to the small size of the third stage. Detailed studies have been made on the orbital-refueling capabilities of various SATURN vehicle configurations. The results of these studies indicate that a configuration with a third stage having a large propellant capacity offers the most promising solution to orbital departing vehicles. For example, when the SATURN B-1 is flown into a low orbit, it not only provides a net payload of 35,000 pounds, but also places in orbit an empty third stage with a complete guidance and control system. This empty stage could be refueled and used as an orbital departure stage, while the 35,000 pounds of net payload could be used to provide additional stages for such missions as lunar landing or other deep space or planetary maneuvers. These additional stages could either be flown into orbit, filled with propellant, or for even greater payload capability, flown into orbit empty and refueled like the standard third stage.

The example used above could also be accomplished by the SATURN B; however, the resulting capability would be reduced considerably since the last propulsion stage for the orbital flight is much smaller as is the net payload capability.

The B-1 and B-3 orbital refueling and mission capabilities will be given here as examples. Orbital launched vehicles, based on the B-1 and B-3 performance figures, have orbital lift-off weights of 360,000 and 875,000 pounds, respectively. The payload capabilities are summarized in Table 14. The payload figures shown in this

Table 14

NET PAYLOAD CAPABILITIES OF ORBITAL LAUNCHED SPACE VEHICLES BASED ON SATURN B-1 AND B-3 CONFIGURATIONS

Mission	B-1	B-3
Lunar Soft Landing	44,000	107,600
Lunar Soft Landing and Return	11,380	29,240
Martian Satellite	63,300	155,000
Martian Soft Landing	130,600	321,300
Martian Satellite and Return	28,000	69,540
Martian Soft Landing and Return	10,350	26,750
Venusian Satellite	41,400	102,600
Venusian Satellite and Return	11,350	28,700

table are net payloads based on a vacuum specific impulse of 420 seconds except for the return flights for which a specific impulse of 300 seconds was used.

For fueling the orbiting vehicle, a total of ten tanker flights is required for complete fueling of a B-1 type orbital vehicle and 14 for the B-3 type.

Depending on the mode of operation and time period involved, one additional flight may be required to provide the living quarters for the fueling and checkout crew prior to orbital departure. Other flights may be required for crew rotation, unless one crew can complete the preflight work, in which case they could return in their own reentry vehicle.

Although the missions selected for Table 14 were kept in a spectrum of relatively large payload weights, there is no reason why smaller vehicles could not be used if it is desirable to reduce not only the number of supply flights but also the single flight payload capability. The table of payload capabilities shows that practically any desired single payload capability can be obtained by this technique. For example, the lunar soft landing payload weight can be increased by a factor of 11. This is sufficient to place on the lunar surface a vehicle capable of returning a 10,000-pound capsule to earth. Thus, this technique provides the earliest possibility of carrying out a manned lunar landing and return mission. Increasing the single flight payload capability by use of the orbital assembly technique could be accomplished; however, it is considered the least desirable of the two solutions for the early time periods under consideration.

f. Propellant Capacities. The importance of designing each of the SATURN stages for optimum propellant capacity cannot be minimized. Yet, from the economy and reliability standpoints, it is important to have only one basic SATURN configuration. To meet these two paradoxical requirements, while at the same time providing complete mission flexibility as far as maximum performance is concerned, the following approach has been used in designing the propellant capacity for each stage: first, the optimum propellant loading is determined for each stage, based on anticipated missions; and then each stage is designed for the maximum propellant loading indicated by the investigation.

This approach provides one basic configuration, which increases vehicle reliability and decreases system cost, with complete mission flexibility, since propellant loading can be determined at a very late date in the schedule, and at the same time resulting in an overall maximum performance capability. For some missions, each stage will not be filled to capacity which results in carrying a small amount of additional tankage weight. This weight is less than 0.5% of the stage weight when the tank is filled to 90% of capacity. The loss in payload for this 90% propellant capacity loading in the booster would be approximately 1% for a low orbit mission and less than 0.4% for a 24-hour orbit.

Table 15 gives the recommended standard design capacity for the individual stages and compares these with the (approximate) optimum propellant loading for the individual missions.

G. COST AND SCHEDULE

1. Introduction

A summary of the cost and schedule data for the initial SATURN program was presented in Chapter III. This summary contained development and funding plans for the B and B-1 vehicle configurations based on several R&D schedules. Also presented earlier was a distribution of R&D and operational money by the constituents of the program and a breakdown of ABMA and contractor cost for the complete R&D program.

Presented and discussed in this section are additional initial program development and funding plans, follow-on program development and funding plans, a typical complete SATURN program funding requirement , a mission chart for the initial R&D program, and the effect of the SATURN configuration decision on the National Space Program.

2. Ground Rules and Study Requirements

In addition to the ground rules and study requirements listed earlier in the report another request was made. ARPA requested that cost and schedule data illustrating the transition from the SATURN B, information presented during the SATURN and TITAN C review on 16 through 18 September 1959 to the new R&D program resulting from this study,be included in this report. The data presented to the SATURN - TITAN C review committee, chaired by Dr. York (DDR&E) and Dr. Dryden (NASA), was based on the first six vehicles of the R&D program and was specifically aimed at the

Table 15

RECOMMENDED SATURN B-1 TANK CAPACITIES AND PROPELLANT LOADINGS

Mission	Number of Stages	$(F/W)_I$	I	II	III	IV
Design Propellant Capacity (lb)			650, 000	330, 000	100, 000	29, 000
DYNA-SOAR 100-Nautical Mile Orbit	2	1. 3	650, 000*	330, 000*		
300-Nautical Mile Orbit	3	1. 3	600, 000	300, 000	80, 000	
300-Nautical Mile Orbit	3	1. 25	615, 000	325, 000	90, 000	
Escape	3	1. 3	600, 000	300, 000	100, 000*	
Escape	4	1. 3	600, 000	300, 000	75, 000	25, 000
Escape	4	1. 25	600, 000	330, 000*	90, 000	29, 000*
24-Hour Orbit	3	1. 25	625, 000	330, 000*	100, 000*	
24-Hour Orbit	4	1. 3	600, 000	300, 000	75, 000	24, 000

*Indicates use of maximum capacity.

DYNA-SOAR type mission. The data presented at the subject meeting is given in Table 16 and is referred to as Plan A. The schedule used for Plan A is based on an early availability of an operational SATURN to meet the requirements of the DYNA-SOAR program. The new R&D program is based on the "Original Schedule" defined in Chapter III, and its corresponding cost breakdown is the same as that presented in Fig. 5. The major differencies in cost between Plan A and the new R&D program is as follows:

a. The vehicle R&D cost forPlan A is for the first six flights of a ten-vehicle program (as requested by ARPA).

b. The GSE R&D cost in Plan A includes some hardware consumed during the program which is now included under Engine and Vehicle R&D in the new R&D funding breakdown.

c. The launch facilities included in Plan A provide the capability of one launch per month; whereas, in the new R&D program a maximum rate of only three per year is required. (Plan A - two blockhouses and four pads, new R&D - one blockhouse and one pad.)

d. The other facilities include a production capability of 12 per year for Plan A and six per year for the new R&D program.

e. The increase in cost for the stage hardware in the new R&D program is due to four more units of flight hardware.

f. No G&C hardware is included in Plan A, as requested by ARPA, since the DYNA-SOAR payload which was under consideration contained a G&C system.

g. When Plan A was submitted to the evaluation committee, it was stated that no cost had been included for vehicle transportation, mission and payload integration, and supporting research.

3. Initial Program

In addition to the development and funding plans and information presented in Chapter III on the initial SATURN program, several other schedule and cost variations were investigated.

The development and funding plan for the SATURN B, based on the funding limitations, is presented in Fig. 30 and results in a R&D completion date delay of six months when compared to the

Table 16

COMPARISON OF SATURN B SCHEDULE AND FUNDING PLANS

	CY 1961	CY 1962	CY 1963	CY 1964	Total
Plan A Schedule[1]	3	3			6
New R&D Schedule	2	2	3	3	10

ITEM	PLAN A	NEW R&D
Engine R&D	\$ 48.2 x 10^6[2]	\$ 49.0 x 10^6
Vehicle R&D	221.6	256.3
GSE R&D	21.7	14.5
Propellants	10.2	14.5
Launch Facilities	29.1	15.7
Other Facilities	33.2	28.5
Hardware		
First Stage	41.6	69.8
Second Stage	6.9	12.1
Third Stage	3.8	7.1
G&C		18.6
GSE	11.4	13.8
Launch Operation	8.0	20.7[3]
Mission and Payload Integration		13.5
Supporting Research		20.0
TOTAL	\$ 435.7 x 10^6	\$ 554.1 x 10^6

[1] Submitted to Dr. York (DOD) 16 September 1959 with copies to ARPA.

[2] Includes propellant for second and third stage engine R&D.

[3] Includes transportation to AMR.

SATURN "B" SCHEDULE & FUNDING PLAN

(BASED ON LIMITED FUNDING PLAN)

CY	1961	1962	1963	1964	1965	1966	1967	1968	1969	1970	TOTAL
OPERATIONAL FLIGHT SCHEDULE					1 2 1	2 1 2 1	2 1 2 1	2 1 2 1	2 1 2 1	2	30
R & D FLIGHT SCHEDULE	1	1 1	1 1	1 1 1 1	1						10

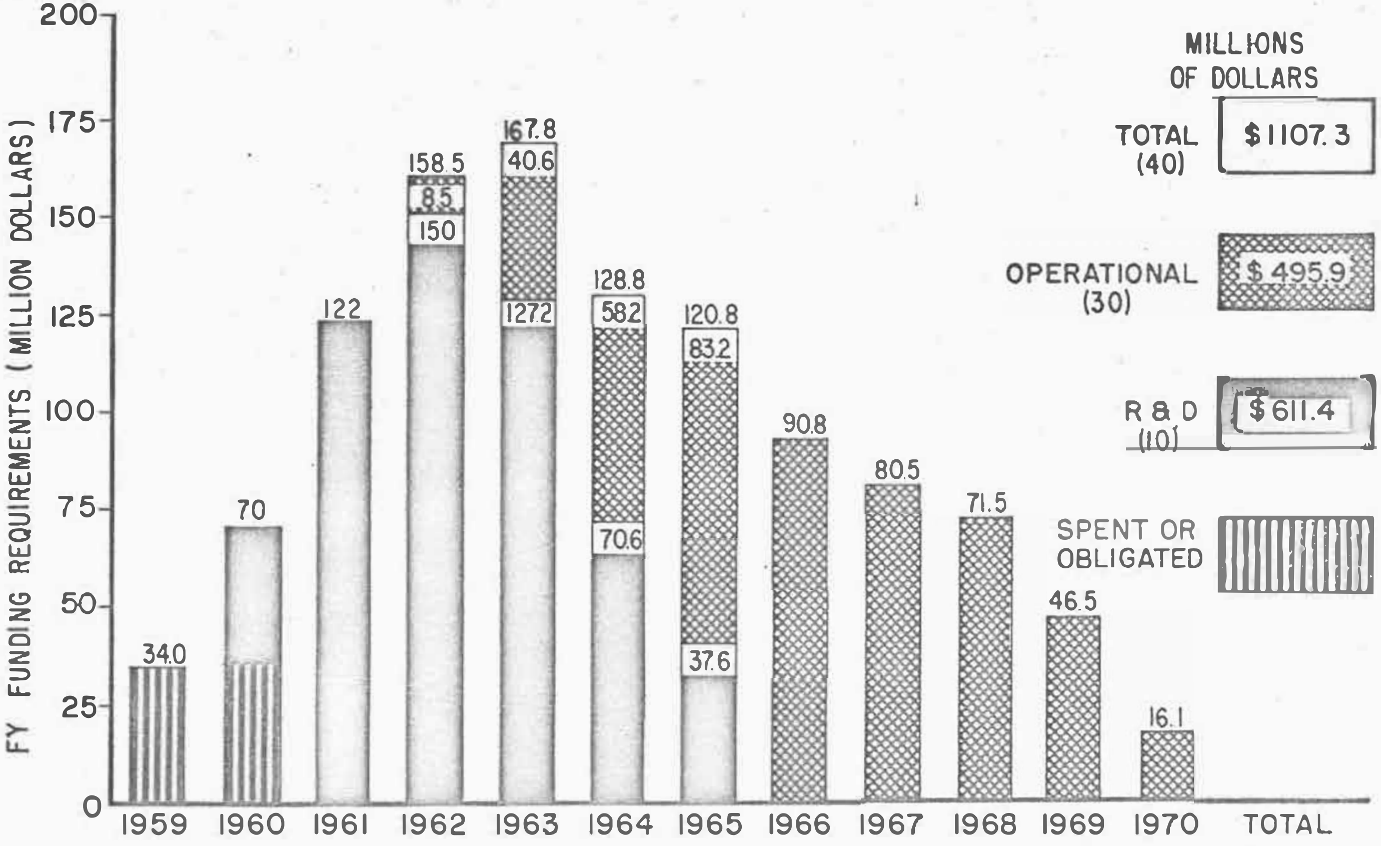

FIG. 30 GE 140-30-59 REV A 17 OCT 59

the original schedule. In addition to the schedule delay, the R&D cost is increased from $554.1 million to $611.4 million. The operational program cost for either of the schedules is the same, $495.9 million; however, each flight would be delayed by six months when compared to the original schedule.

Development and funding plans, based on the original and funding limitation schedules, for the SATURN B-1, with different prime contractors for the second and third stages, are given in Figs. 31 and 32, respectively. These figures show that an increase in R&D cost of $67.2 million and a delay of 15 months results when the FY budgets are restricted as defined by the ground rules of the study. A review of the data presented on one prime contractor accomplishing all upper stage development and production (Figs. 5 and 6) as compared to having two prime contractors (Figs. 31 and 32) shows that: Approximately 8 to 10% of the total R&D cost and 6 to 7% of the total operational cost could be saved if all the upper stages were developed and manufactured by one prime contractor.

Although the accelerated schedule indicated in Fig. 8 is possible, the desirability of attempting such a program is questionable unless the earliest practical SATURN availability is recognized as a national space program requirement. The funding requirements to accomplish the accelerated schedule are as follows:

	R&D Program	Operational Program
FY 1959	$34 million	
FY 1960	134 million	$16.1 million
FY 1961	234 million	67.0 million
FY 1962	104 million	86.8 million
FY 1963	61 million	91.3 million
FY 1964	8 million	80.6 million
FY 1965		72.2 million
FY 1966		46.6 million
FY 1967		27.8 million
FY 1968		7.5 million
	$575 million	$495.9 million

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SATURN "B-1" SCHEDULE AND FUNDING PLAN

(BASED ON ORIGINAL SCHEDULE-DIFFERENT CONTRACTORS FOR 2nd & 3rd STAGE)

CY	1961	1962	1963	1964	1965	1966	1967	1968	1969	TOTAL
OPERATIONAL FLIGHT SCHEDULE				1	2 1 2 1	2 1 2 1	2 1 2 1	2 1 2 1	2 1 2	30
R&D FLIGHT SCHEDULE	1 1	1 1	1 1 1	1 1 1						10

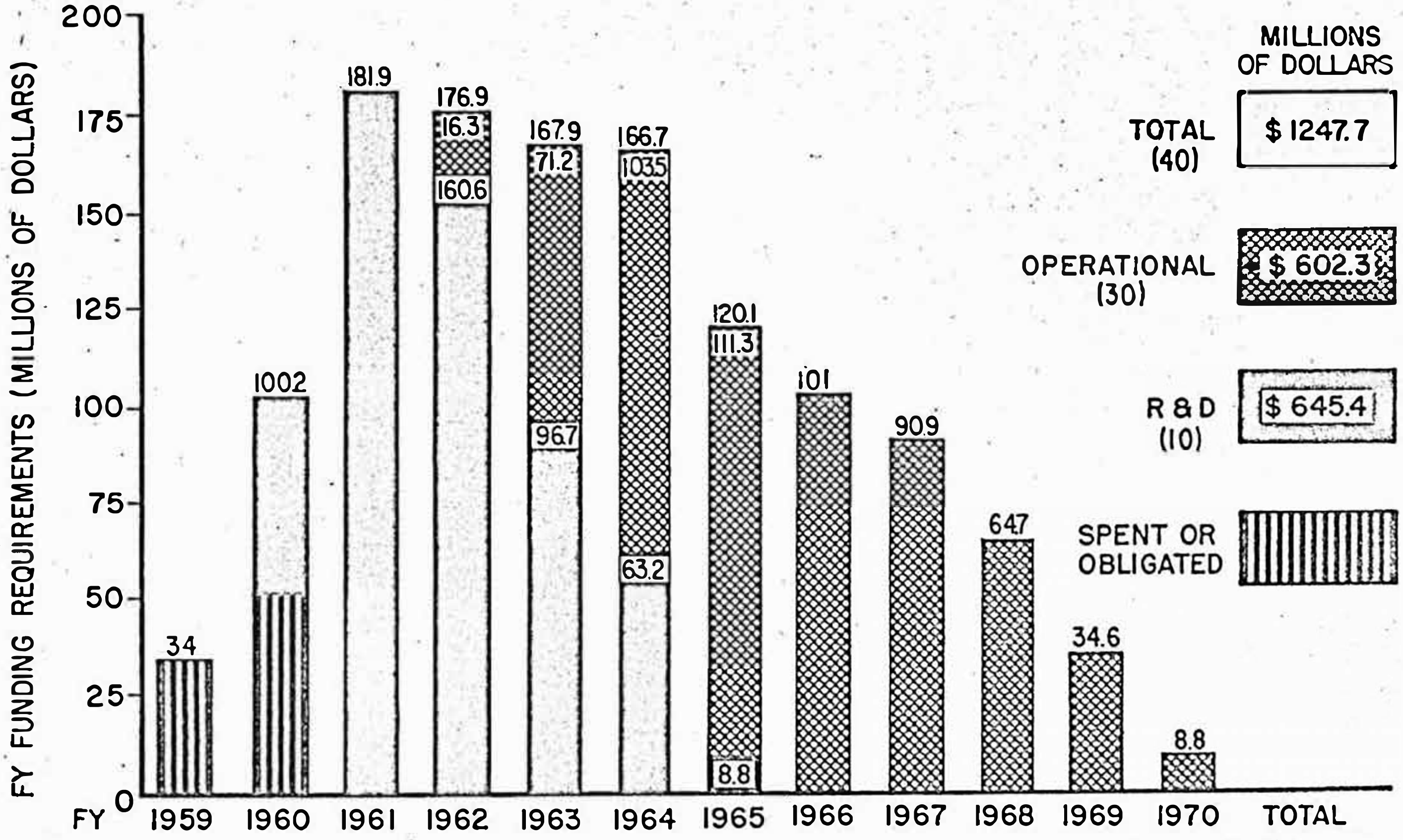

FIG. 31 GE 140-31-59
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SATURN "B-I" SCHEDULE AND FUNDING PLAN

(BASED ON LIMITED FUNDING PLAN-DIFFERENT CONTRACTORS FOR 2nd & 3rd STAGE)

CY	1961	1962	1963	1964	1965	1966	1967	1968	1969	1970	TOTAL
OPERATIONAL FLIGHT SCHEDULE					1	2 1 2 1	2 1 2 1	2 1 2 1	2 1 2 1	2 1 2	30
R & D FLIGHT SCHEDULE	1	1 1	1	1 1 1	1 1 1						10

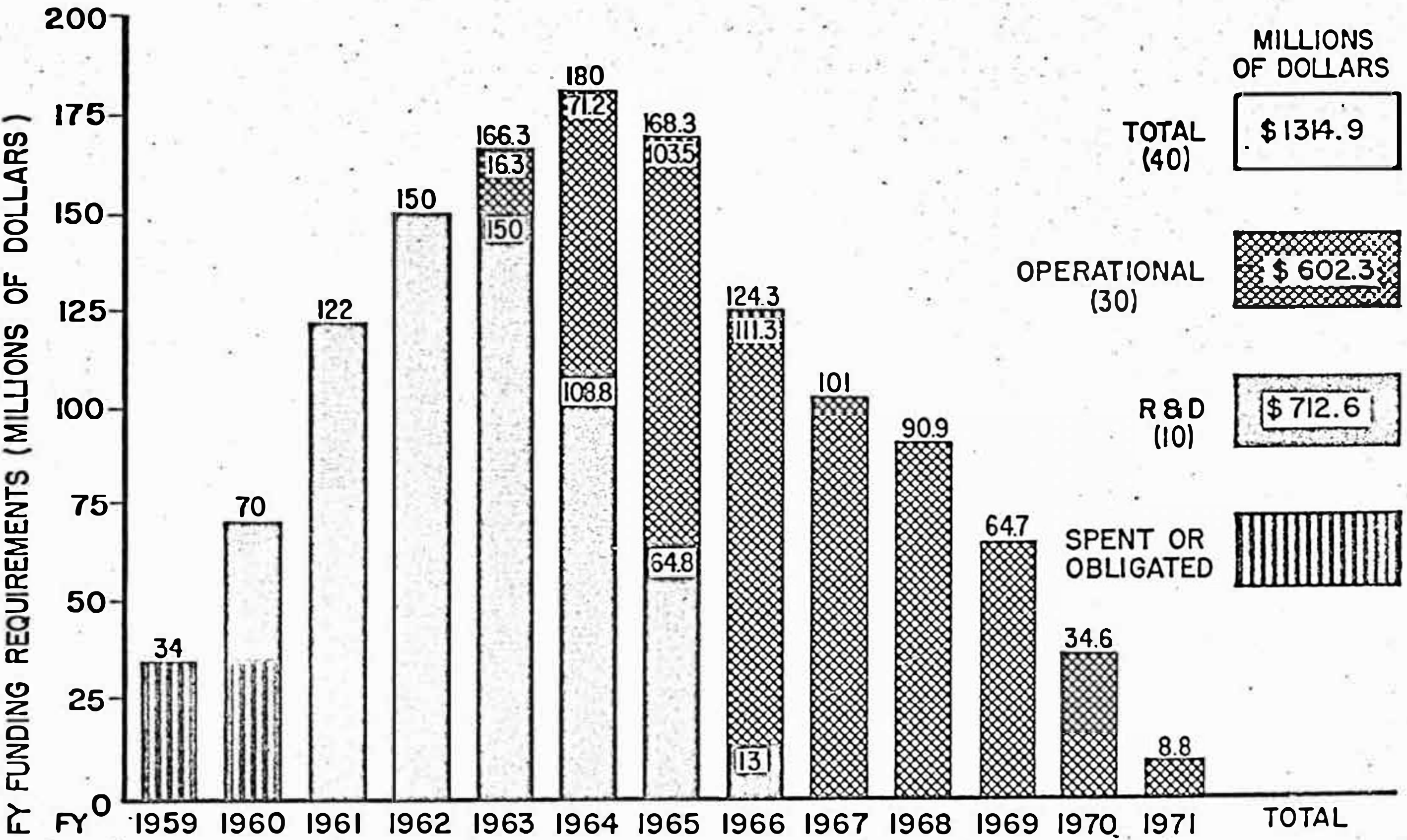

FIG. 32

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In order to illustrate the lead times involved in preparing a SATURN booster, in particular, the first flight vehicle (SA-1), a detailed schedule is presented in Fig. 33. The data included on this figure reflects the present planning for SA-1 and indicates the delays resulting from not having an approved upper stage configuration. As shown on this schedule, the first flight cannot occur until 21 months after an upper stage diameter decision is made.

Figure 34 gives a tentative mission chart for the proposed SATURN B-1 ten-vehicle R&D program. The fourth stage on Vehicles 9 and 10 is optional, and these two flights could be made with a three-stage configuration.

4. Follow-On Program

The follow-on or growth potential program for the SATURN B-1 is presented for the three configurations discussed earlier (B-2, B-3, and B-4). The primary purpose for this portion of the study is to indicate trends, orders of magnitude in cost, and schedules for various follow-on programs. To illustrate the relative cost of development for the three different follow-on vehicle configurations, each was added to the SATURN B-1 based on the original schedule (Figs. 35, 36, and 37). To illustrate the variations in operational availability of the follow-on vehicle, the B-2 was chosen as a typical configuration and added to the SATURN B-1 based on the optimum and funding limitation schedules (Figs. 38 and 39, respectively).

In determining the schedules and funding requirements, it was assumed that the combined development cost annual rate would be approximately the same as the maximum rate required for the B-1 program alone. In reviewing the lead times involved for an optimum follow-on development, it was found that the funding requirements were compatible with the combined maximum funding rate assumption. Therefore, the schedule data presented in Figs. 35 through 39 are considered near optimum and a shifting of initiation date has little or no effect on total R&D cost. Figure 40 summarizes the results of the follow-on program schedule and cost study and gives the development sequence for the R&D flight vehicles.

NO.	YEAR	VEHICLE DESCRIPTION	FLIGHT MISSION
1. 2. 3. 4.	1961 1961 1962 1962	8 X 165 K BOOSTER DUMMY UPPER STAGES NO PAYLOAD	PROPULSION TEST STRUCTURAL TEST CONTROL SYSTEM TEST ATTEMPT OF RECOVERY
5. 6.	1963 1963	8 X 188 K BOOSTER 4 X 220 K 2nd STAGE DUMMY THIRD STAGE OR NOMINAL PAYLOAD	SYSTEMS TESTS AS ABOVE FOR BOTH STAGES - ATTEMPT OF ORBITING NOMINAL PAYLOAD IF DESIRABLE
7. 8.	1963 1964	8 X 188 K BOOSTER 4 X 220 K 2nd STAGE 4 X 20 K 3rd STAGE PAYLOAD	SYSTEMS TESTS AS ABOVE FOR ALL STAGES - GUIDANCE TEST FOR ESCAPE TRAJECTORY OR ORBIT
9. 10.	1964 1964	8 X 188 K BOOSTER 4 X 220 K 2nd STAGE 4 X 20 K 3rd STAGE 2 X 20 K 4th STAGE PAYLOAD	SYSTEMS TESTS AS ABOVE FOR ALL STAGES - GUIDANCE TEST FOR 24 hr. ORBIT OR PLANETARY MISSION

FIG. 34

TENTATIVE MISSION CHART (SATURN R&D)

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SATURN B-I PLUS B-2 R&D SCHEDULE & FUNDING PLAN

BASED ON ORIGINAL SCHEDULE

VEHICLE / CY	1961	1962	1963	1964	1965	1966	1967	1968	1969	TOTAL
B-1 SCHEDULE	I I	I I	I I I	I I I						10
B-2 SCHEDULE					I I	I I	I I			6

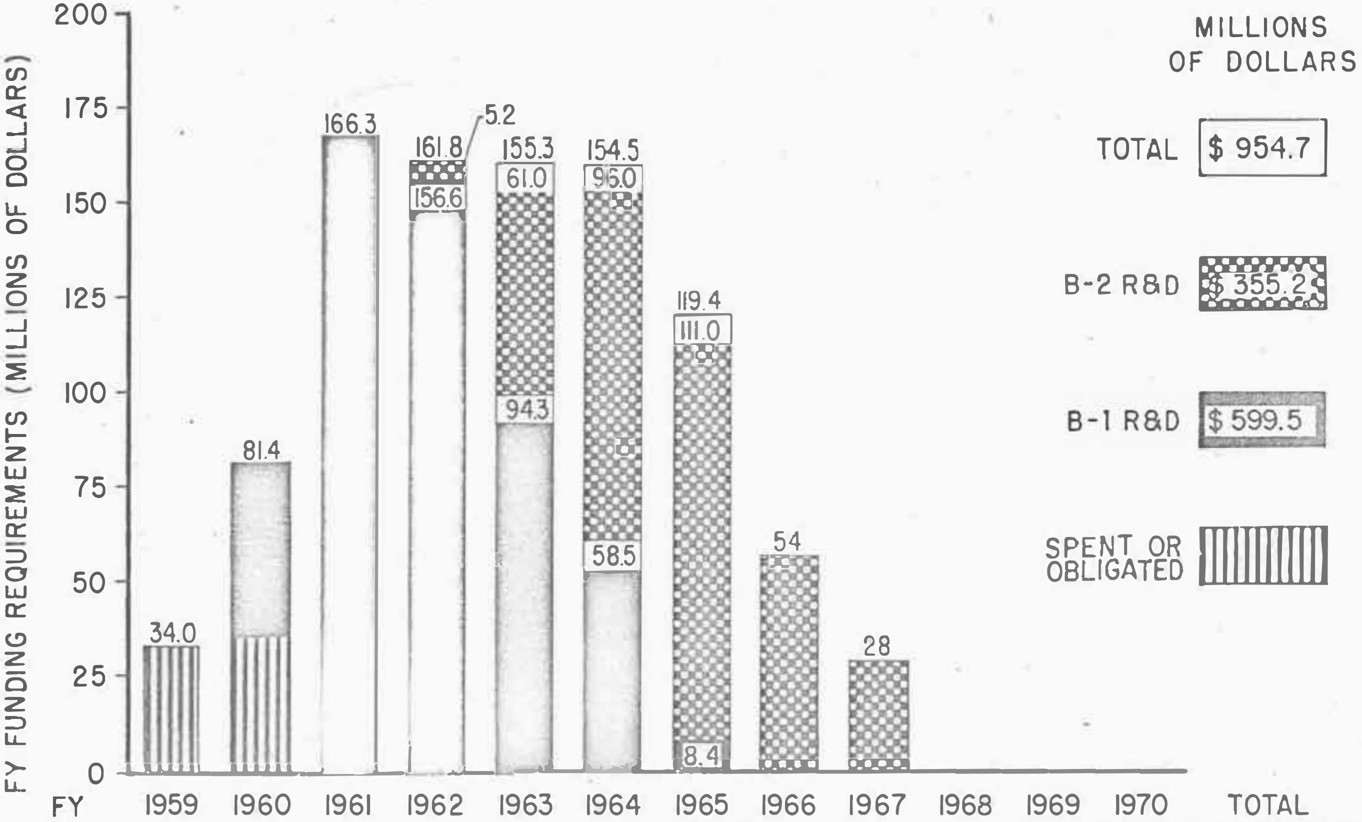

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FIG. 35

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SATURN B-1 PLUS B-3 R&D SCHEDULE & FUNDING PLAN
(BASED ON ORIGINAL SCHEDULE)

VEHICLE / CY	1961	1962	1963	1964	1965	1966	1967	1968	1969	TOTAL
B-1 FLIGHT SCHEDULE	I I	I I	I I I	I I I						10
B-3 FLIGHT SCHEDULE					I I	I I	I I			6

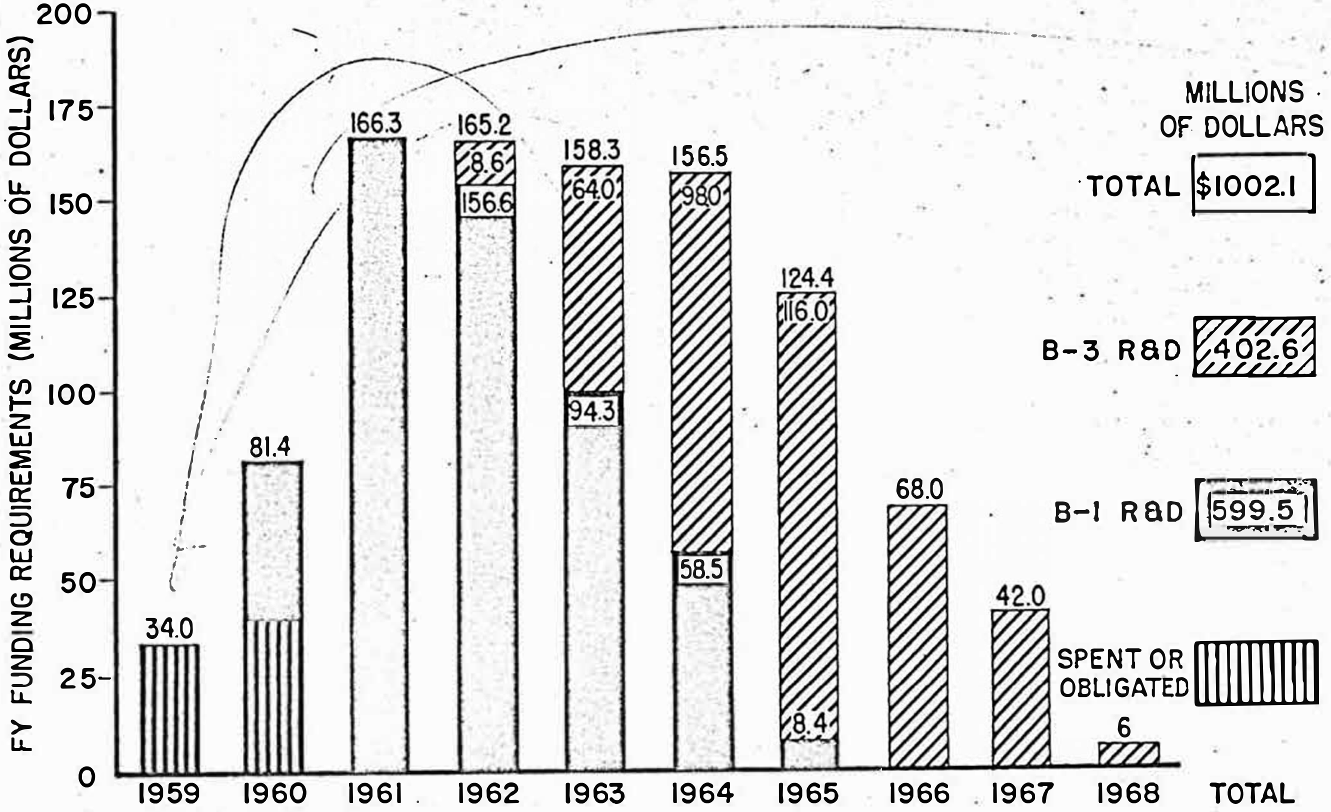

FIG. 36 GE-140-40-59 REV A 17 OCT 59

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SATURN "B-1" PLUS "B-4" R&D SCHEDULE & FUNDING PLAN

(BASED ON ORIGINAL SCHEDULE)

VEHICLE \ CY	1961	1962	1963	1964	1965	1966	1967	1968	1969	TOTAL
B-1 FLIGHT SCHEDULE	I I	I I	I I I	I I I						10
B-4 FLIGHT SCHEDULE				I	I I	I I	I I			7

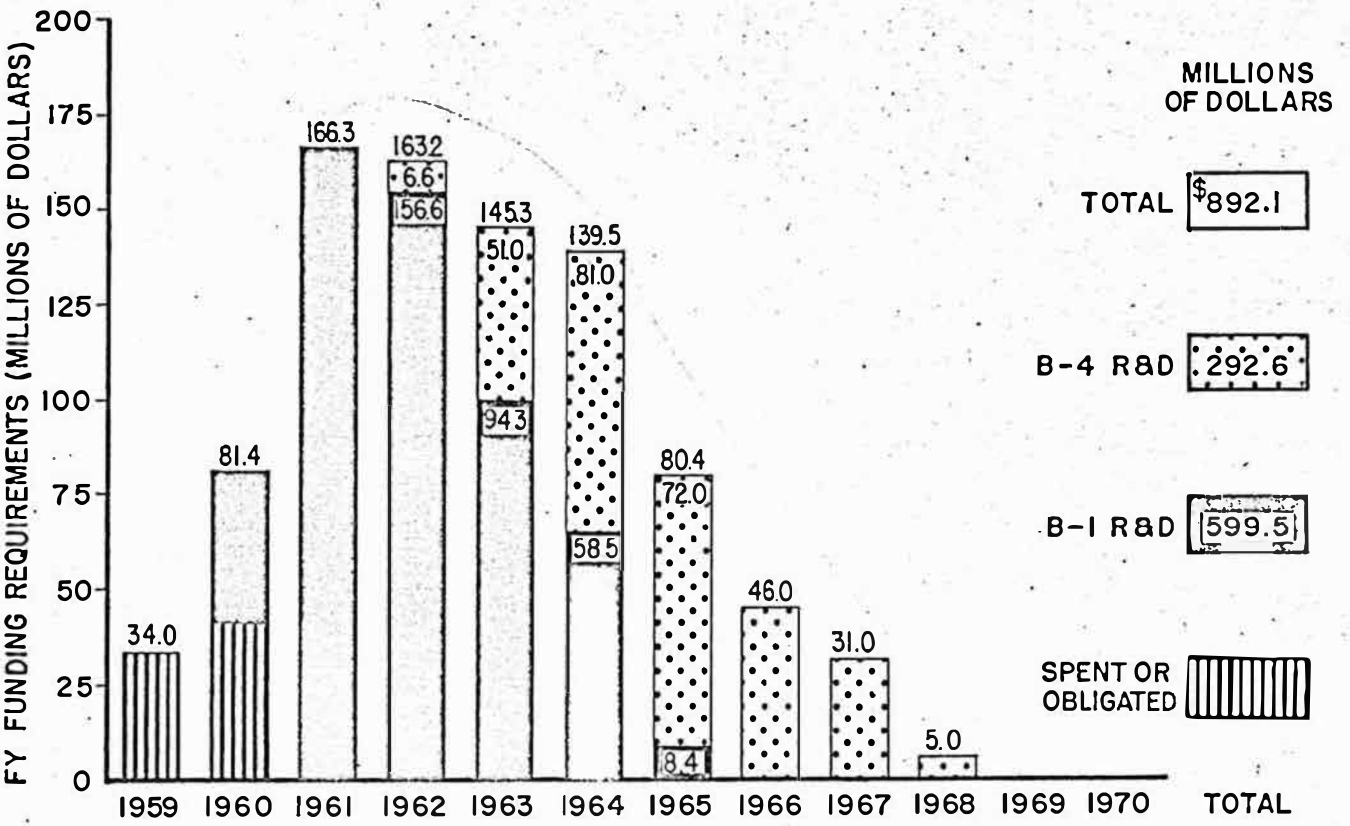

FIG. 37 GE 140-41-59 17 OCT 59 REV A

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SATURN "B-1" PLUS "B-2" SCHEDULE & FUNDING PLAN

(BASED ON OPTIMUM SCHEDULE)

CY	1961	1962	1963	1964	1965	1966	TOTAL
R& D B-2 FLIGHT SCHEDULE				1 1	1 1	1 1	6
R& D B-1 FLIGHT SCHEDULE	1 1	1 1 1	1 1 1 2				10

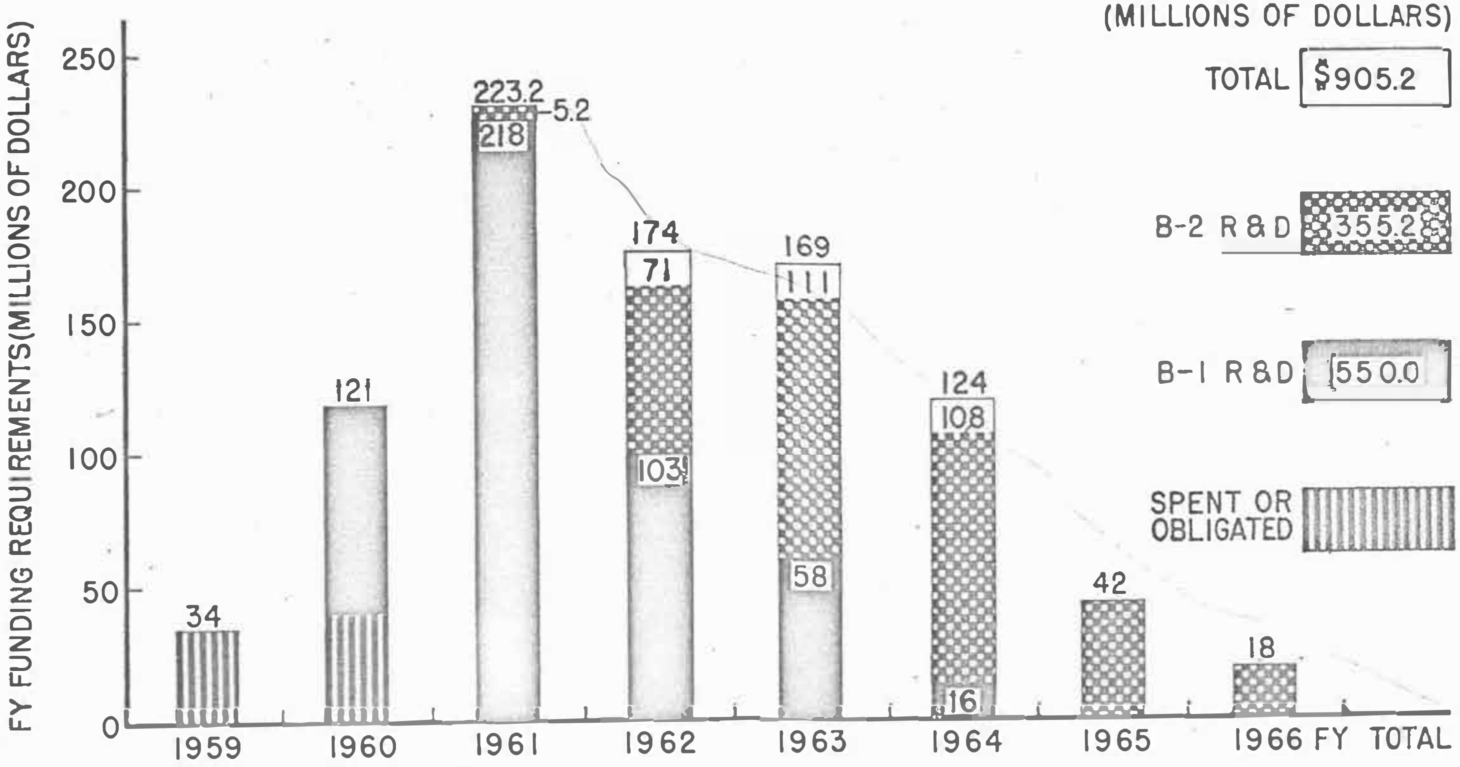

FIG. 38

GE140-43-59 17 OCT 59
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SATURN "B-1" PLUS "B-2" SCHEDULE & FUNDING PLAN

(BASED ON LIMITED FUNDING PLAN)

CY	1961	1962	1963	1964	1965	1966	1967	1968	1969	TOTAL
R & D 'B-2' FLIGHT SCHEDULE						I I	I I	I I		6
R & D 'B-1' FLIGHT SCHEDULE	I	I I	I	I I I I	I I					10

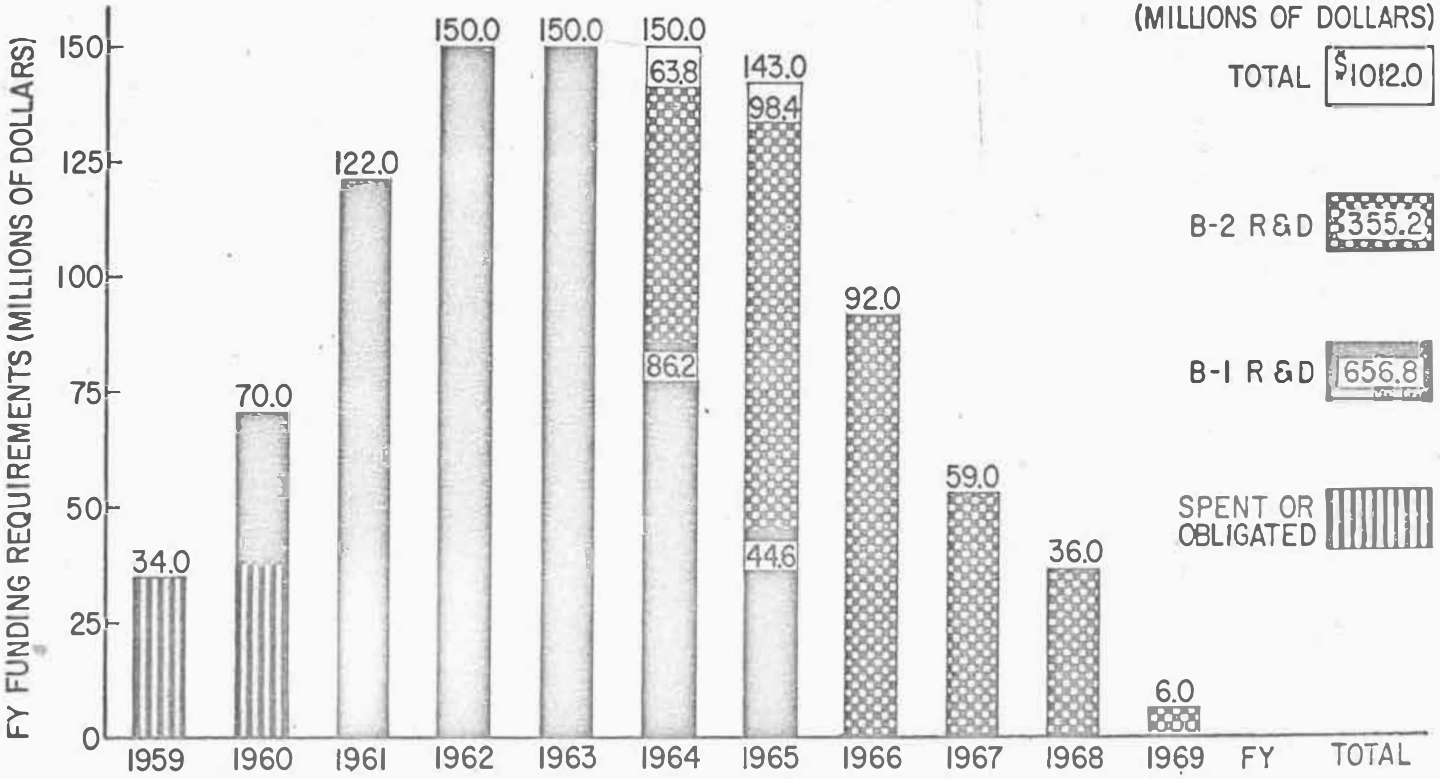

FIG. 39

GE140-42-59
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R & D SCHEDULE FOR SATURN SYSTEMS B-2, B-3 & B-4

SYSTEM	TYPE OF SCHEDULE	FLIGHT VEHICLE DESCRIPTION	CY						TOTAL FLIGHTS	TOTAL R & D FUNDING REQUIREMENTS (MIL. DOLLARS)
			1963	1964	1965	1966	1967	1968		
B-2	ORIGINAL	BOOSTER ONLY WITH DUMMY UPPER STAGES			I I				6	355.2
		2 STAGE VEHICLE WITH DUMMY 3RD STAGE				I				
		3 STAGE VEHICLE				I	I			
		4 STAGE VEHICLE					I			
B-2	LIMITED FUNDING	BOOSTER ONLY WITH DUMMY UPPER STAGES				I I			6	355.2
		2 STAGE VEHICLE WITH DUMMY 3RD STAGE					I			
		3 STAGE VEHICLE					I	I		
		4 STAGE VEHICLE						I		
B-2	OPTIMUM	BOOSTER ONLY WITH DUMMY UPPER STAGES		I I					6	355.2
		2 STAGE VEHICLE WITH DUMMY 3RD STAGE			I					
		3 STAGE VEHICLE			I	I				
		4 STAGE VEHICLE				I				
B-3	ORIGINAL	BOOSTER ONLY WITH DUMMY UPPER STAGES			I I				6	402.6
		3 STAGE VEHICLE				I I	I			
		4 STAGE VEHICLE					I			
B-4	ORIGINAL	2 STAGE VEHICLE WITH DUMMY 3RD STAGE		I	I I				7	292.6
		3 STAGE VEHICLE				I I	I			
		4 STAGE VEHICLE					I			

FIG. 40 GE 140-56-59 17 OCT 59

Figure 41 illustrates the order of magnitude for a complete SATURN program. This is considered to be typical and is based on the following:

a. SATURN B-1 R&D, based on the original schedule (Ref. Fig. 6).

b. SATURN B-1 operational program of 30 vehicles with a six flight per year rate (Ref. Fig. 6).

c. SATURN B-2 R&D program (Ref. Fig. 34).

d. SATURN B-2 operational program of 15-plus vehicles at a rate of six flights per year. As shown, such a program would approach a maximum annual funding rate of $240 million.

5. Effect of Initial SATURN Configuration Decision on National Space Program

The choice of the initial SATURN configuration will have a direct impact on the early United States space flight capability; however, the long range effect of the SATURN program is considered to be of even more importance. To illustrate this point, one typical mission for which the SATURN could be used has been chosen to show the limits of U. S. capability and the effect of the B-1 versus B configuration decision on the initial program (Fig. 42). The mission is that of manned lunar exploration. The assumptions used in this example are as follows:

a. A vehicle of 50,000 (earth) pounds takeoff weight is required to return two men from the lunar surface to the earth.

b. Two such vehicles are required on the moon to provide an adequate safety factor for the return flight (or a total of 100,000 pounds on the moon).

c. A total of 300,000 pounds is required to establish the first six-man facility on the lunar surface. This includes return transportation as well as material to construct the facility.

d. Using the B-1 configuration for the initial program, a launch rate of six operational firings per year through 1967 is available. At this time the B-3 would become operational and have a launch rate of 12 flights per year.

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TYPICAL TOTAL FUNDING REQUIREMENTS FOR SATURN "B-1" PLUS "B-2" R&D PLUS OPERATIONAL PROGRAM

CY		1961				1962				1963				1964				1965				1966				1967				1968				1969				TOTAL
B-1	R&D		1		1		1		1		1	1	1	1	1	1																						10
	OPERATIONAL																1	2	1	2	1	2	1	2	1	2	1	2	1	2	1	2	1	2	1	2		30
B-2	R&D																	1		1		1		1		1	1											6
	OPERATIONAL																											1	2	1	2	1	2	1	2	1	2	15+

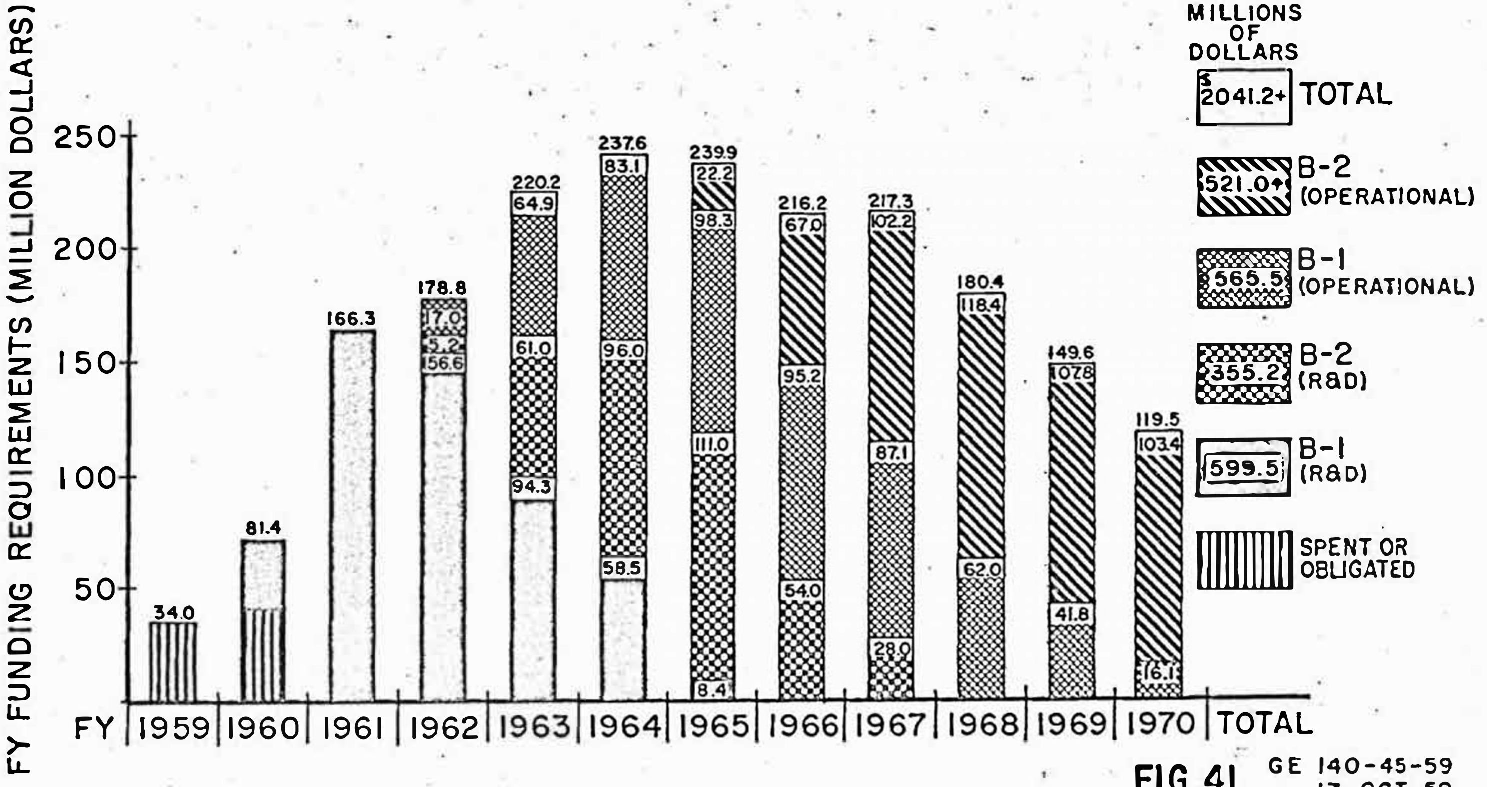

FIG. 41

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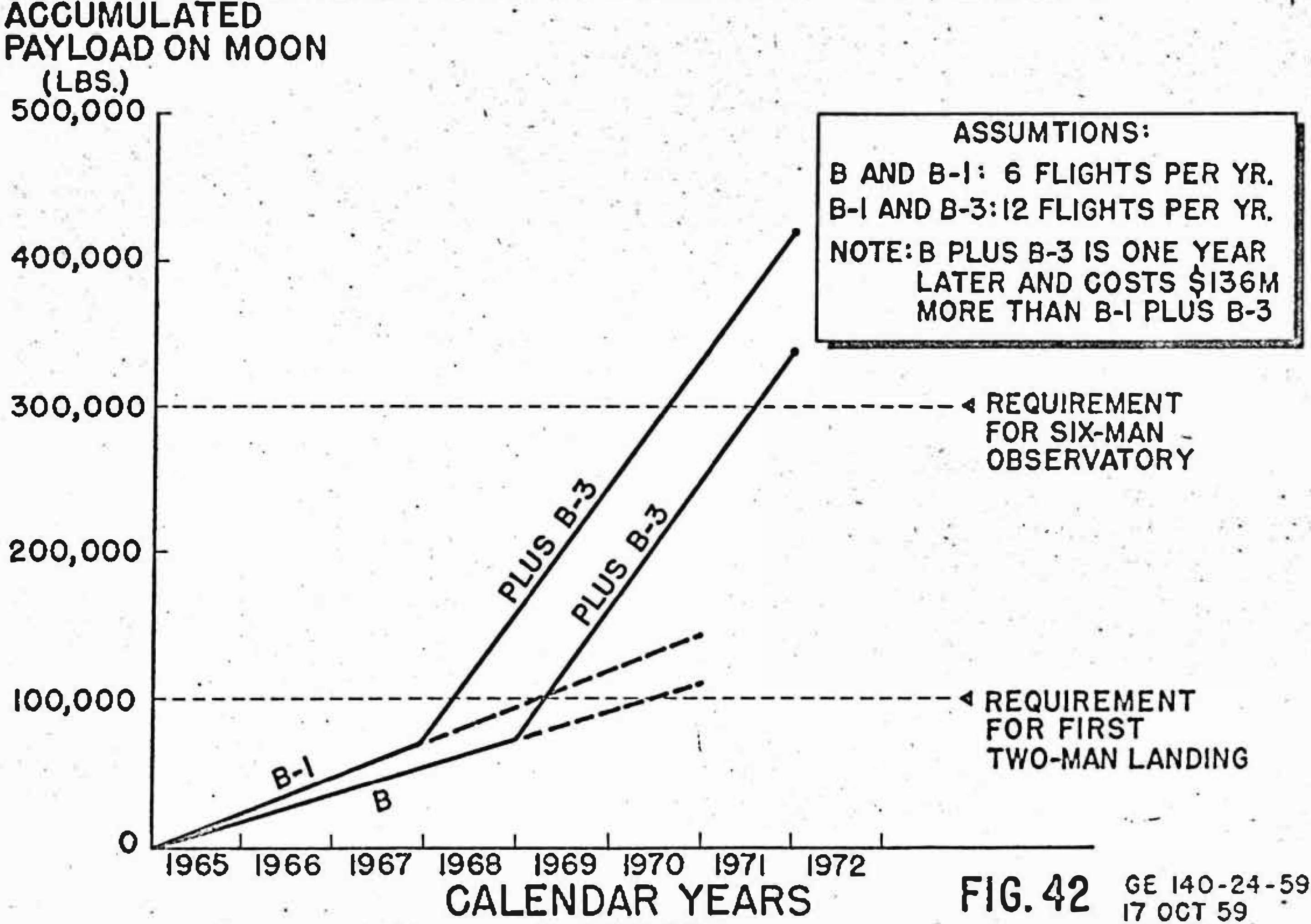
EFFECT OF INITIAL SATELLITE CONFIGURATION ON MANNED LUNAR LANDING CAPABILITY



ASSUMTIONS:
B AND B-1: 6 FLIGHTS PER YR.
B-1 AND B-3: 12 FLIGHTS PER YR.
NOTE: B PLUS B-3 IS ONE YEAR LATER AND COSTS $136M MORE THAN B-1 PLUS B-3

FIG. 42

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REV. A

e. Using the B configuration for the initial program; a launch rate of six operational flights per year through 1968 is available. At this time the B-3 would become operational and have a launch rate of 12 flights per year.

f. In establishing the operational date of the B-3 vehicle a maximum of $160 million per year is available for vehicle R&D (B-1 plus B-3 or B plus B-3).

g. All available SATURN vehicles are used for the manned lunar exploration mission. This, of course, is completely unrealistic, however, it helps to convey the point of truly "maximum capability."

h. All vehicles are 100% successful. This is again unrealistic.

i. Orbital refueling will be used, as described earlier in this report.

It is felt that this example brings out several important points which should be considered, not only for making a choice of the initial SATURN configuration, but also in the area of future planning for the United States space program.

a. The initial choice of the SATURN B configuration would delay the country's capability for sending two men to the moon and returning them by at least one year, mid 1969 versus mid 1968 for the B-1, and at a cost of approximately $136 million more.

b. The development of the B-3 vehicle as a follow-on program to the B configuration will, in addition to costing $136 million more and providing an operational vehicle at a later date, result in a lower initial operational reliability due to the drastic change in upper stages.

c. Taking into consideration, the fact that the mission reliability of the SATURN will not be 100% and that most probably not all of the SATURN vehicles will be used for this specific mission, it becomes readily apparent that if the United States wishes to accomplish even a limited manned lunar exploration program by approximately 1970, the SATURN B-1 configuration should be chosen and the development program accelerated.

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APPROVAL: DSP-TM-13-59

E. L. Williams

for H. H. KOELLE
Chief, Future Projects Design Branch

W. A. Mrazek

W. A. MRAZEK
Director, Structures and Mechanics Laboratory

Wernher von Braun

WERNHER VON BRAUN
Director, Development Operations Division

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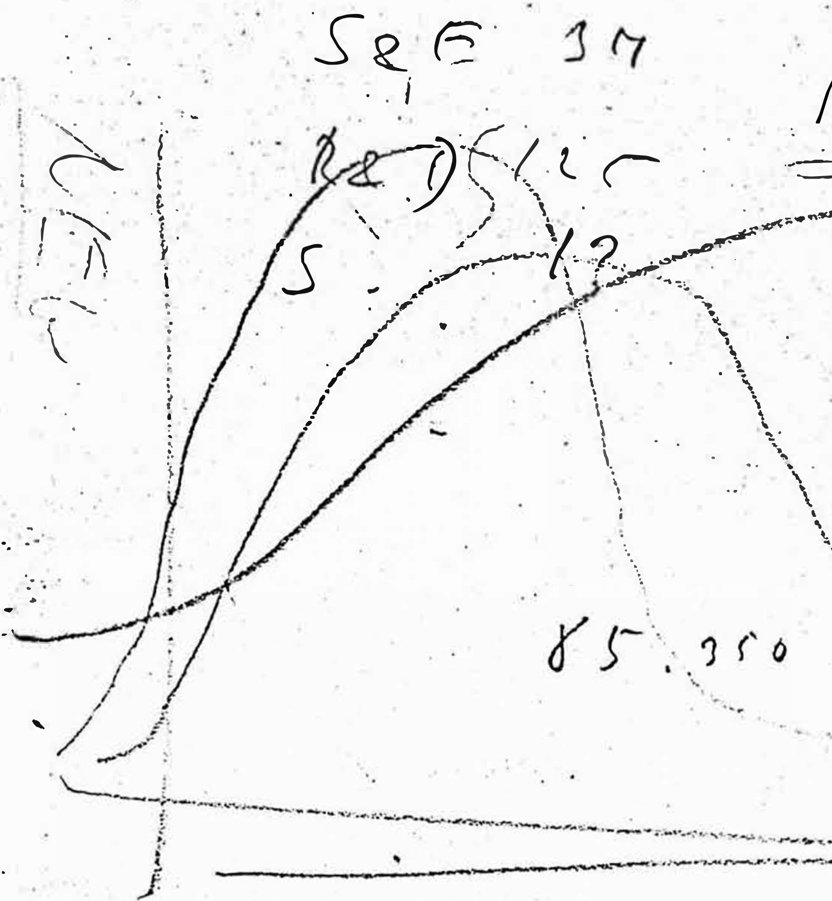

60
125 180

150
45

C&E

27.0	14.9
164.1	65.9
	150.0
659	230.8

1.20
90.0

F-162

	SATURN	TOTAL
C&E	Sol 14.9 ✓	26.220 ✓
S&E } 65.9	65.9	58.350 ✓
S of PL		27.000 ✓
R&D	150.00	150.000 ✓
	230.8	261.540 261