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GEORGE C. MARSHALL SPACE FLIGHT CENTER
HUNTSVILLE, ALABAMA

SCIENCE AND TECHNOLOGY ADVISORY COMMITTEE FOR MANNED SPACE FLIGHT

PRESENTED AT
MANNED SPACECRAFT CENTER
HOUSTON, TEXAS
JUNE 29, 1964

National Aeronautics and Space Administration

PREFACE

A ① This publication contains the Marshall Space Flight Center contribution at the Science and Technology Advisory Committee (STAC) Meeting at the Manned Spacecraft Center, Houston, Texas, on 29 June 1964. U

The printed material includes the essential points of each presentation even though there may be some deviations due to time limitations and delivery techniques of the individual presenters.

A minimum number of illustrations have been deleted from this publication to prevent classification of the entire document. Those illustrations which have been deleted may be obtained through official channels from:

Marshall Space Flight Center
Huntsville,
Alabama.

A ② Provides excellent historical background concerning Saturn development, missions, technical problems, etc.

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LAUNCH VEHICLE SYSTEMS

By Dr. William A. Mrazek

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A brief historical review of the generation of the Saturn family will only introduce to you the many constraints and circumstances which partially influenced the configuration of the three largest U. S. launch vehicles, as we know them today.

After the Russian Sputnik flight, ARPA (Advanced Research Projects Agency) of DOD approached the Army Ordnance Missile Command, specifically the Research and Development Division under Dr. von Braun. They requested a proposal for demonstration of a test-stand firing of a clustered-engine propulsion vehicle in the shortest possible time and with a relatively modest price tag. The selection of the H-1 engine from Rocketdyne, a natural choice which I will explain in the second part of this paper, obligated half of the available funds. With the other half we had to design the propulsion system, manufacture it, purchase or modify necessary tooling and assembly gear, and also convert the large test stand which had been used on former Army-Air Force program but was now available for our use. In order to stay within the budget limitation we utilized 70" and 150"/105" tooling available and several bulkheads out of our warehouses.

We succeeded almost on time; but during the design phase a program redirection of consequential impact came from ARPA: "Consider this stage as the first one of a large launch vehicle and go shopping for a second stage, preferably one which already exists in the United States' military inventory." This directive resulted in some of the restraints determining the configuration of the first stage of the Saturn I and IB. Tooling had already started and there was no way to modify our previous decisions.

Early configuration studies for an earth, and later lunar orbital operation, resulted in designation of Configuration 1, or C-1, then C-2, C-3, C-4, and, finally, with the selection of the LOR or Lunar Orbital Rendezvous mode, to the C-5. This was later redesignated the Saturn V.

Saturn I represents the results of the configuration 1, or C-1, study. The Saturn IB configuration is derived from the selection of the first stage of the Saturn I and the third stage of the Saturn V which increased almost twice the payload capability of the Saturn I. This higher payload was later requested by the Manned Spacecraft Center in order to accomplish essential lunar rendezvous exercises in near-earth orbit.

Let me explain now why we have an S-IV stage and not an S-III stage. The urgency of the program forced us to seek an early participation of the aerospace industry in building the upper stages. Studies of a C-4, planned for the earlier earth orbital rendezvous, consisted of a stack of four stages. The S-IV stage was originally the fourth stage of a C-4 and in size it fitted the Saturn I. Later, already under contract with Douglas Aircraft Company for the stage, we added two more engines to optimize it for the Saturn I. The C-4 was abolished with the selection of the LOR and the Saturn V.

The Saturn V finally called for a five-engine first stage, the S-IC; a five-engine second stage, the S-II; and an improved S-IV called the S-IVB as the third stage. This S-IVB stage uses twice the thrust of the six RL-10 type engines which were available at the earlier time. The new hydrogen engine, J-2, which I also will describe later, in a single application satisfied the requirement of 200, 000 pounds sea-level thrust.

Figure 1 shows dimensions and inboard profiles of the Saturn family.

On the right is Saturn I. We are well aware that a single tank configuration in the first stage would have saved some stage weight and conversely would have increased the payload. As I mentioned before, the time allowed and the limited funds did not permit us to use an optimized version. The interconnected multi-container approach was fast, easy, and inexpensive for manufacturing. It still is today. Part transportation is no problem. In considering tankage volume, it is weightwise immaterial whether a single tank or a bundle of any size smaller tanks is used to contain a prescribed volume under the same pressure. The weight difference experienced is due to the additional interconnect lines, the structural tie-in points, the thrust distribution into many skins, and on the upper end to the thrust transmission from many carrying containers into a single outside skin of the upper stage.

The center configuration on Figure 1 is the Saturn IB. I have described the reason for this configuration. We are confident that we will be able to improve the payload capability to approximately 36,000 pounds to help achieve multiple exercises for the astronauts in a near-earth orbit. The S-I stage is supposed to show that it has been somewhat lengthened to carry more propellant. The S-IVB stage shown is the version for the Saturn IB.

The configuration on the extreme left of Figure 1 is the Saturn V with a height of almost 400 feet. You see the first stage as a single-tank version separated by the interstage cylindrical portion. This configuration was chosen after many optimization studies considering the NPSH requirement of the pumps of the engine. Equally important were weights of gaseous pressurants, unusable propellants, etc. In the Saturn V you really see an assembly of different design principles. First, an unsophisticated arrangement of the LOX front container and the RP-1 in the rear resolved production as well as propulsion problems.

The common bulkhead now used in both upper stages resembles very much the basic design of an S-IV stage. There were, of course, some difficulties which had to be solved.

Now even with the S-IVB in Saturn IB ahead of the Saturn V schedule, the contracted hardware will be delivered on time.

Mrs. Neisler

In order to achieve a compatible engineering approach for all stages of the Saturn V, a document "Saturn V Design Ground Rules" was made applicable for all stages. A few of the rules of a more general nature directed:

ORDER

a. Vehicle orientation on the pad and engine location, including stage and engine alignment.

b. Separable research and development equipment and network

c. Standby time on pad

d. Separation modes

e. Applicable safety factors

1. Research and Development flights

2. Manrated vehicle

3. Pressure Vessels

4. Valves, filters, switches, etc.

f. Ground wind standard

g. Ninety-five per-cent (95%) quasi-steady state flight wind profile.

h. Umbilical access

For stages of the Saturn V, certain specific requirements were established:

a. Separation technique

b. Interface control

c. Test requirement

d. Basic configuration

e. Propellant fill rate

There were many more.

One of the items connected with this review is the mode of separation. The launch release of the Saturn V under approximately 95-98% thrust takes place automatically after the proper ignition is established. In order to suppress any adverse dynamic effects on the vehicle, we chose a controlled release which has a decoupling and damping effect, thus avoiding the triggering of longitudinal oscillation which would possibly exceed static and flight loads in certain structural areas.

The separation of the S-II stage from the S-IC first stage occurs in two steps. We do not allow the "fire-in-the-hole starting", but require a minimum of 6 to 10 feet physical distance before igniting the engines. This will assure us that we have no unnecessary protection of the forward stage due to pressure and temperature peaking in the closed or partially vented interstage.

Another parameter considered at the second stage separation was the problem encountered during clearing of the unneeded interstage. To avoid possible collision during separation, we decided to introduce a staged separation. The first separation occurs just above the S-IC stage. The second stage is moderately accelerated by the ullage rockets which keep the liquid propellant settled and the suction ports covered during this short near zero "g" period. The first stage is retarded at the same time by larger retrorockets to compensate the tailoff thrust of the five F-1 engines and retard the stage. This approach of a clear break and elimination of bumping increases the separation reliability.

After ignition of the J-2 engines of the second stage, we allow about 30 seconds time for recovery from residual pitching or yawing due to control maneuvers just before separation or resulting from the unavoidable unsymmetrical ignition of the five engines.

At this time a simple explosive release charge will separate the interstage skirt which until this moment remains attached to the S-II stage. The skirt will move backwards because of acceleration differences and will, finally, after impingement by the expanded jets, be propelled past the engines. In the case of one engine not igniting, the auxiliary electro-hydraulic actuator system will tug the engine in, thus again avoiding a possible collision. A secondary mission can be flown with four engines alive.

Another built-in feature of the stages is the relatively large ullage space above all propellants. Vent behavior and pressurization transient during start-up are of concern to us. As soon as better data are available, slight adjustment of propellant loading in one or more stages will permit an improvement of the overall performance of the vehicle. The parameter affected by increasing the weight is the take-off acceleration which is constrained by safety measures for the launch pad and the flight dynamics of the first few seconds. The clearance requirements between vehicle and the nearby umbilical tower must be fulfilled.

Considering the compressed research and development flights of the Saturn V, it will doubtless be necessary to intensify ground test programs. We will not allow unqualified hardware for flights. In addition to a full functional testing of components, a well-designed qualification program will assure the elimination of "random" failures.

A safe spread between loads in general and capabilities will eliminate even the overlap of 3 sigma values, allowing for side effects not understood.

As an example, I would like to briefly discuss the S-II Program efforts in the qualification of the second stage. The following tests are scheduled:

a. Hazardous suppression test which will determine the maximum allowable propellant leak into the enclosed interstage (connected with N_2 flush provisions to avoid explosions.

b. Common bulkhead tests:

Tests of 8-inch diameter and 55-inch diameter models to establish extrapolation factors to 33 feet diameter.

c. Separation joint and mating tests.

d. LH_2 tank wall insulation test.

1. Program for establishing K factors

2. To establish shell buckling strength and fixity assumption.

e. Functional separation tests.

f. Propellant sloshing and vortexing tests, also, baffle evaluation to establish efficiency.

g. Environmental control system testing.

h. Structural tests of different subsystems and systems.

i. Propulsion system tests to establish early evaluation of the pressurization requirement.

j. Battleship testing with gradual engine firing introduction.

k. Full-fledged qualification program of all active components.

l. Zero leakage test program.

m. All-systems vehicle with gradual transfer to fully automatic firing mode.

n. Dynamic testing of full configuration with participation of all flight hardware under empty and many other different flight conditions.

o. Ground wind oscillation program (flutter effect).

p. Facility checkout program.

1. Transportation oscillation

2. Wind excitation

3. Umbilical fit

4. Automatic tanking procedures

q. Ullage and retrorocket tests

r. Service platform tests

s. Electromechanical mockup test for qualifying automatic checkout and launch procedures, establishing tapes as part applicable to the launch tape.

t. Continuous stage acceptance firing.

u. High force shock and vibration and acoustics testing of subassemblies and critical mounting arrangements.

v. Combination environmental testing including force, vibration, and cryogenics.

This concludes my general launch vehicle discussion, and I would like to spend a few more minutes discussing the engine programs used in the Saturn program. I will not include the small attitude motors.

In my presentation I have several times referred to engine designations such as the F-1, J-2, etc. In the Saturn I/IB program we use the engines depicted in Figure 2.

The H-1, produced by Rocketdyne, is an outgrowth of the IRBM and ICBM programs. It has been several times uprated and gradually improved; new component designs have been introduced. The IRBM-ICBM engine was repacked. This was obviously necessary to simplify the clustering of eight engines. The reliability of this engine is high. In connection with a limited engine-out program, inadvertently demonstrated during the SA-6 flight last month, the H-1 will be a workhorse that will stay a long time with the IB program. As a second-stage engine of the Saturn I, the only available high-energy hydrogen/oxygen engine available at that time, the RL10-A3 by Pratt and Whitney, was used.

The Saturn V program is fully dependent on the Rocketdyne F-1 engine and the upper stage high-energy engine, the J-2.

The next slides will show you pertinent data and performance achievements. They will also explain some of the critical problem areas. Of course, we still have problems with both engines. They just do not have the maturity of the H-1 workhorse.

Figure 3 shows the key milestones in our engine development programs. The engines are the longest lead-time items in the entire vehicle, therefore it is necessary to define and start their development ahead of other phases of our program. Note that five to eight years are required from program initiation to qualification.

The major problems we have encountered in our engine development programs are associated primarily with these three areas: performance, hardware maturity, and manufacturing, Figure 4. As I will show later, we are well along in the solution of many phases of these problems.

A typical example of problems associated with developing engine maturity is the current development of the J-2 engine control system. Fast engine starting tends to decrease the hydrogen pump stall margin while slow starting aggravates the problem of side loads encountered in an attempt to minimize both of these effects. A promising solution is the use of a "step" opening main LOX valve, Figure 5.

Another example of a troublesome area which is being resolved by further development is the balance piston cavity pressure on the J-2 engine fuel pump. The pump must be constructed so that passageways to the pressure balancing cavities permit proper axial loading of the pump, Figure 6.

This figure illustrates the problem mentioned on the previous figure. Redesign of seals and internal pump surfaces are being effected to reduce this problem, Figure 7.

Turbopump failures in the F-1 engine LOX impeller area have led to the corrective actions shown on Figure 8.

Figure 9 illustrates possible troublesome areas on the F-1 turbopump. It is anticipated that corrective action through continued development will alleviate these problem areas.

Figure 10 shows recent major accomplishments for each of the four launch vehicle engines.

Figure 11 shows the last of the 188K H-1 engines being checked out prior to shipment. Six 200K engines have also been delivered to date.

The stainless steel thrust chamber for the H-1 engine (replaces nickle thrust chamber) has been qualified and is now being delivered on production engines, Figure 12.

Figure 13 shows the launch of Centaur vehicle AC-2 on November 27, 1963. The two RL10 engines on the Centaur stage delivered an average specific impulse of 429.7 seconds and an average thrust of 29950 pounds. The RL10 engine has also performed satisfactorily on the flights S-IV-5 and S-IV-6.

Figure 14 shows one of the last RL10A-3 production engines as it is being prepared for shipment. This completes the delivery of 90 A-3 engines on contract NAS8-2690. There are 32 additional A-3-1 engines on order under contract NAS8-5607 for delivery through 1967. The engine successfully passed a pre-qualification test program in December 1963 and a qualification test program will be completed by September 15 of this year.

Figure 15 is the second dual-position J-2 engine system test stand which was activated in November. This stand has a 500-second run duration capability.

Figure 16 depicts the first 500-second (full-duration) J-2 engine system firing on Delta II test stand.

Two ground test J-2 engines were delivered. Figure 17 is the J-2 engine production line.

Figure 18 is the Rocketdyne assembly area at Canoga Park. They are capable of assembling four engines simultaneously in this area.

Figure 19 is the first of three single engine test stands for acceptance firing of production engines.

I hope that I have given you a better understanding of the motivating influences in the design of the Saturn vehicle from its earliest conception in C-1 studies to the Saturn V Lunar Orbit Rendezvous spacecraft of the future. We have every confidence that the Saturn vehicles have been adequately designed, are being adequately tested, and will perform the missions to which they have been assigned.

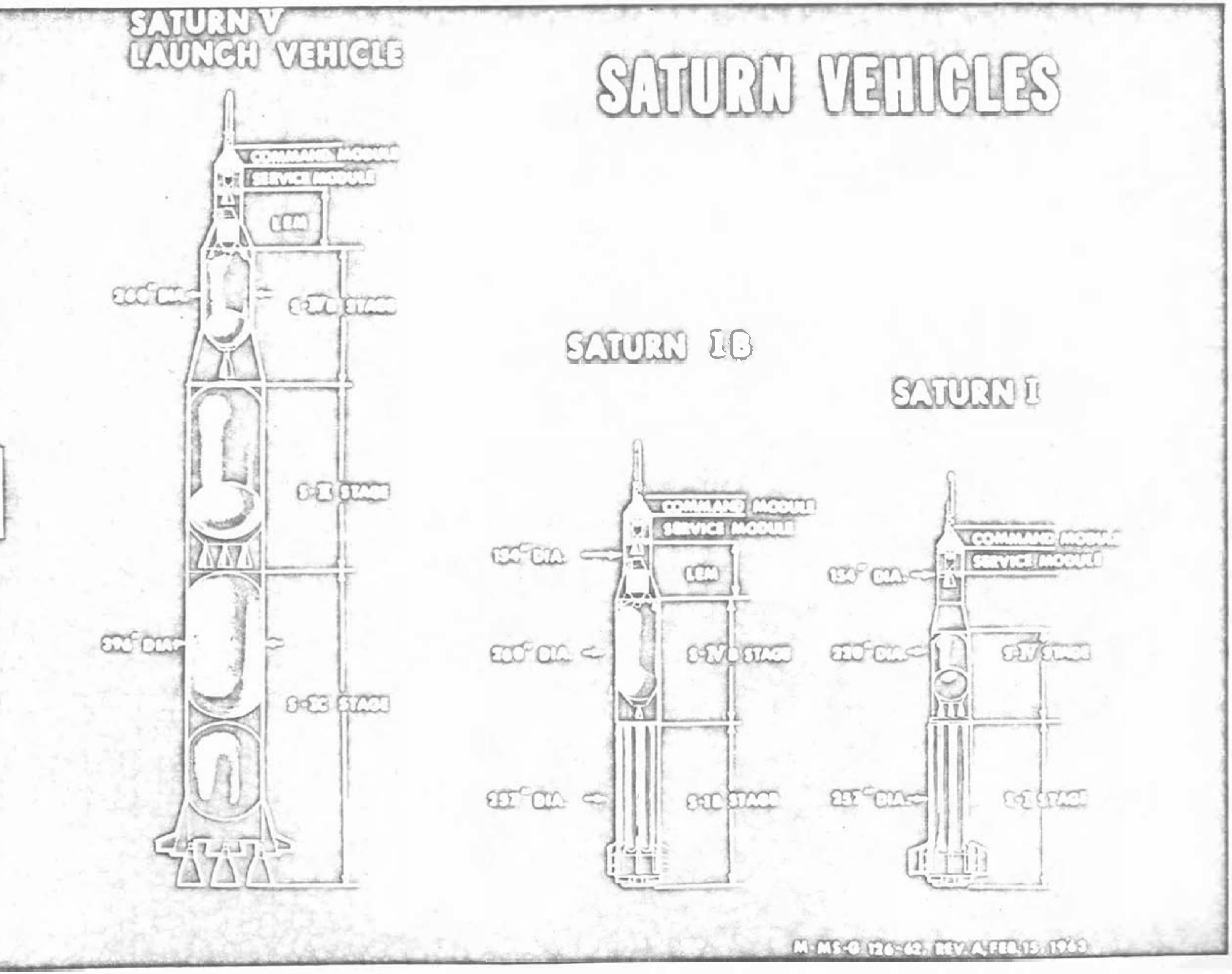
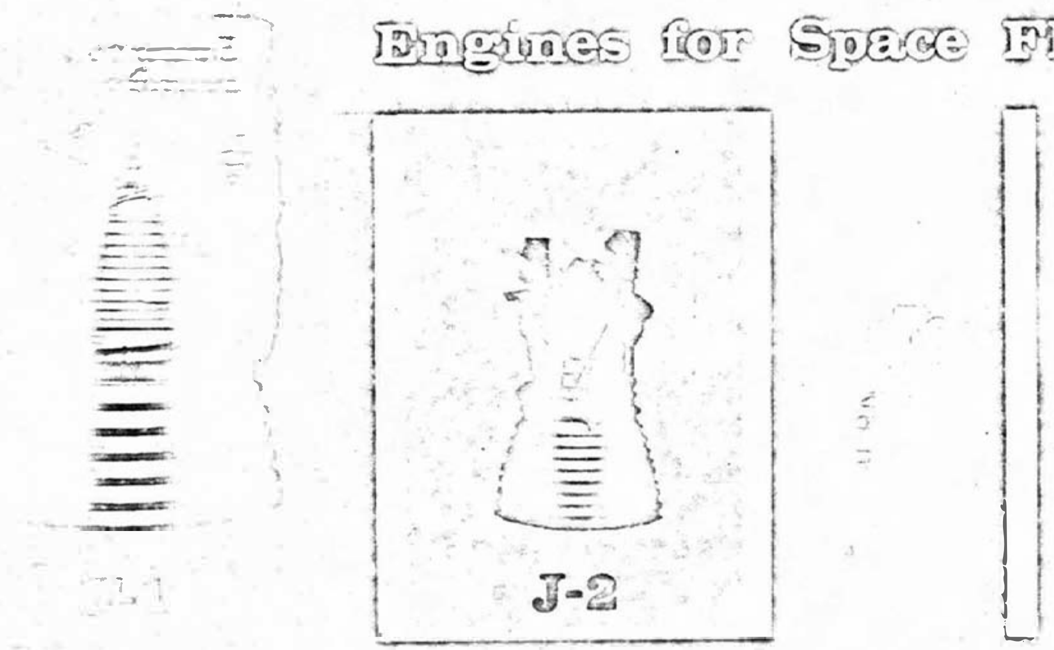

SATURN V
LAUNCH VEHICLE
SATURN VEHICLES
COMMAND MODULE
SERVICE MODULE
LEM
260" DIA.
S-IVB STAGE
S-II STAGE
396" DIA.
S-IC STAGE
SATURN IB
154" DIA.
COMMAND MODULE
SERVICE MODULE
LEM
260" DIA.
S-IVB STAGE
257" DIA.
S-IB STAGE
SATURN I
COMMAND MODULE
SERVICE MODULE
154" DIA.
220" DIA.
S-IV STAGE
257" DIA.
S-I STAGE
M-MS-G 126-62, REV. A, FEB. 15, 1963

Fig. 1

Engines for Space Flight



THRUST	1,500,000 lb	200,000 lb	188,000 lb	15,000 lb
	(SEA LEVEL)	(ALTITUDE)	(SEA LEVEL)	(ALTITUDE)
FUEL	KEROSENE	HYDROGEN	KEROSENE	HYDROGEN

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Fig. 2

Fig. 3

THIS CHART EXCLUDED BECAUSE OF CLASSIFICATION.

MAJOR PROBLEMS

1. PERFORMANCE

2. HARDWARE MATURITY

3. MANUFACTURING

Fig. 4

J-2 CONTROL SYSTEM

SOLUTIONS TO CURRENT DIFFICULTIES:

(1) RAMP MAIN LOX VALVE OPENING

SIMULATION STUDIES INDICATE "RAMP" OPENING WOULD INCREASE STALL MARGIN BUT WOULD NOT HELP SIDE LOAD PROBLEM.

(2) 16°/STEP MAIN LOX VALVE SEQUENCE

RESULTS TO DATE SHOW THAT STALL MARGIN IS INCREASED.

Fig. 5

Fig. 6

J-2 FUEL PUMP

BALANCE PISTON CAVITY PRESSURE INCREASES STILL ENCOUNTERED ON SOME TURBOPUMPS. NO HARDWARE DAMAGE RESULTING FROM THE BALANCE PISTON PROBLEM TO DATE. REDLINE FOR R&D TESTING HAS BEEN REMOVED. EXTENSIVE EFFORT BEING CONTINUED TO ELIMINATE THE PROBLEM.

Fig. 7

THIS CHART EXCLUDED BECAUSE OF CLASSIFICATION.

F-1 TURBOPUMP

POSSIBLE EXPLANATION FOR FAILURES:

(1) FRETTING AT IMPELLER SPLINE

(2) IMPELLER FATIGUE

(3) CONTAMINATION

CORRECTIVE ACTION:

- LOWER ΔP IN SYSTEM TO ALLOW LOWER RPM ON TP
- TIGHT SIDE FIT SPLINE INCORPORATED BETWEEN LOX IMPELLER & COUPLING
- CHANGE COUPLING MATERIAL
- SHOT-PEENING OF IMPELLERS
- DISASSEMBLY, INSPECTION & CLEANING OF ALL PUMPS

Fig. 8

Fig 9

THIS CHART EXCLUDED BECAUSE OF CLASSIFICATION.

MAJOR ACCOMPLISHMENTS

H-1	1. COMPLETED DELIVERIES OF 188K ENGINES (JAN. '64, 35 ENGINES) 2. DELIVERED 6 200K ENGINES 3. STAINLESS STEEL THRUST CHAMBER 4. FLOWN SUCCESSFULLY ON 3 FLIGHTS DURING THIS PERIOD
RL10	1. AC-2 LAUNCH 2. SA-5 AND 6 LAUNCH (S-IV-5) 3. COMPLETED PRE-QUAL (DEC. '63) 4. COMPLETE DEL OF 90 A-3 ENGINES OF CONTRACT NAS8-2690 (DEC. '63)
J-2	1. ACTIVATED DELTA TEST STAND (NOV. '63) 2. FIRST 500 SECOND ENGINE TEST (DEC. '63) 3. DEMONSTRATED GIMBAL CAPABILITY (APR. '64) 4. DELIVERED 2 GROUND TEST ENGINES (APR. '64-MAY '64)
F-1	1. DELIVERED 2 GROUND TEST ENGINES 2. DEMONSTRATED 6 GIMBAL CAPABILITY (OCT. '63) 3. DEMONSTRATED NOZZLE EXTENSION FOR FULL DURATION 4. COMPLETED CONSTRUCTION OF TEST STAND 10 AT EDWARDS 5. COMBUSTION INSTABILITY UNDER CONTROL

Fig. 10

Fig 11

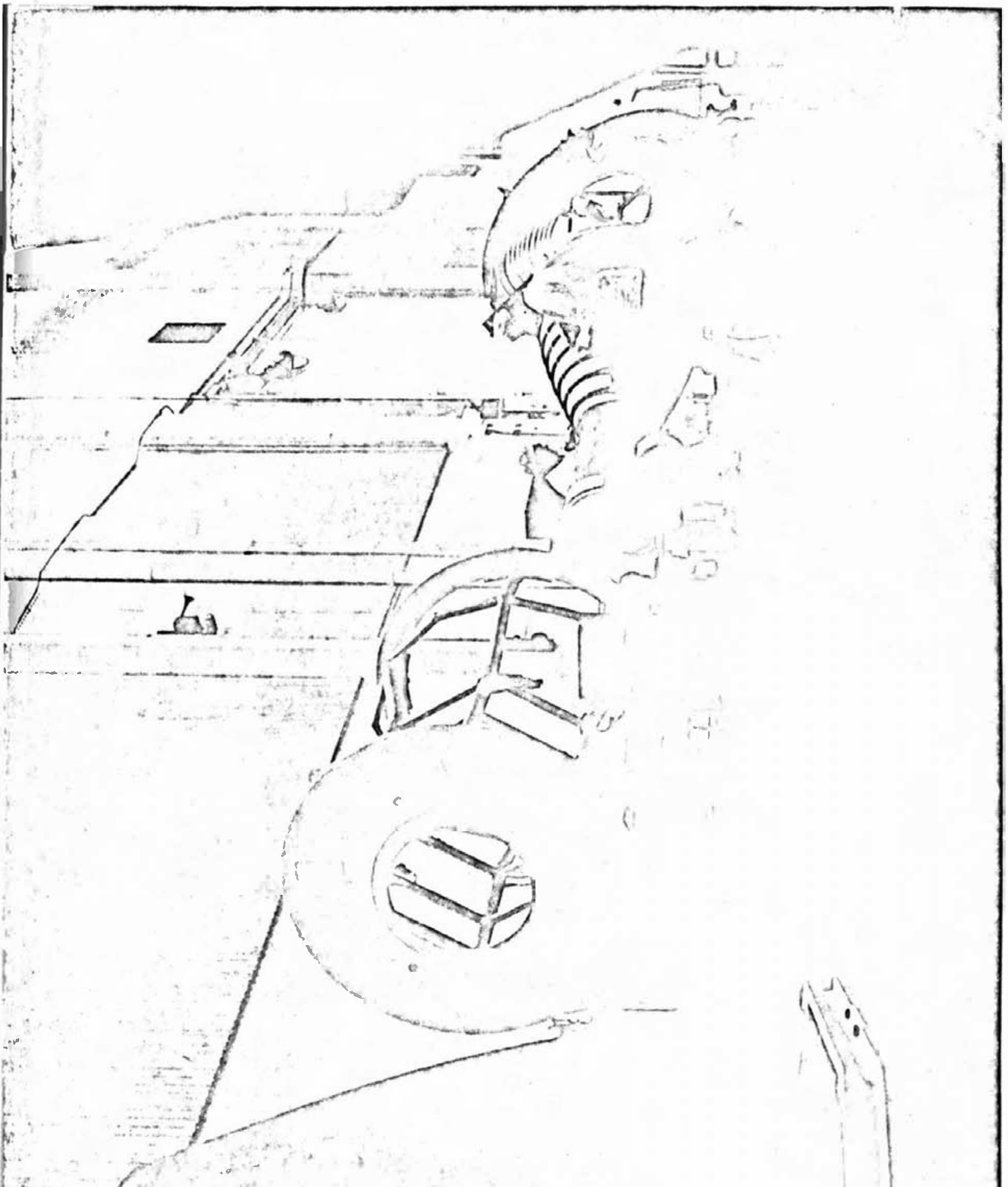

Fig 12

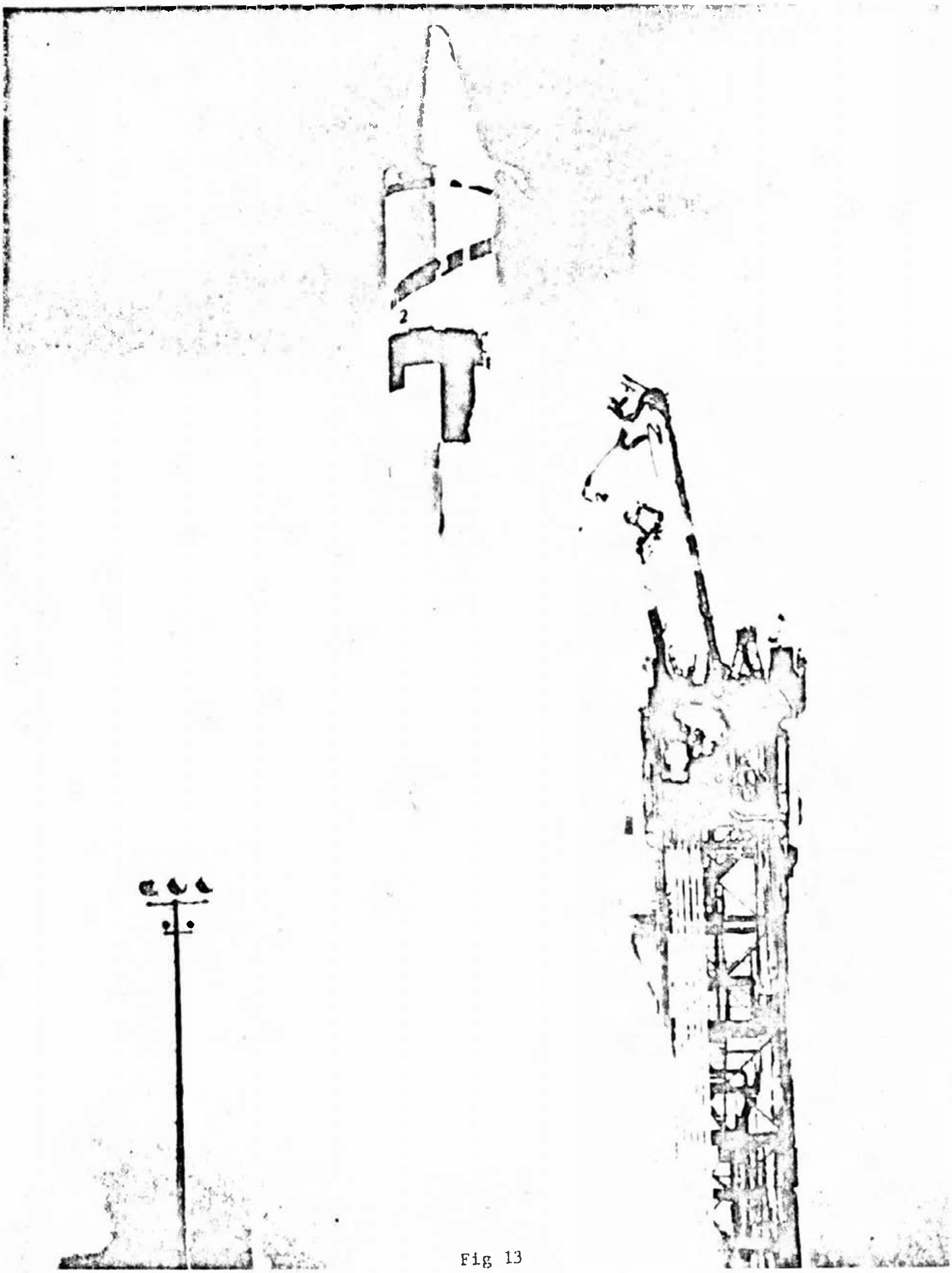

Fig 13

Fig 14

J-2 DELTA 2 TEST STAND - SANTA SUSANA

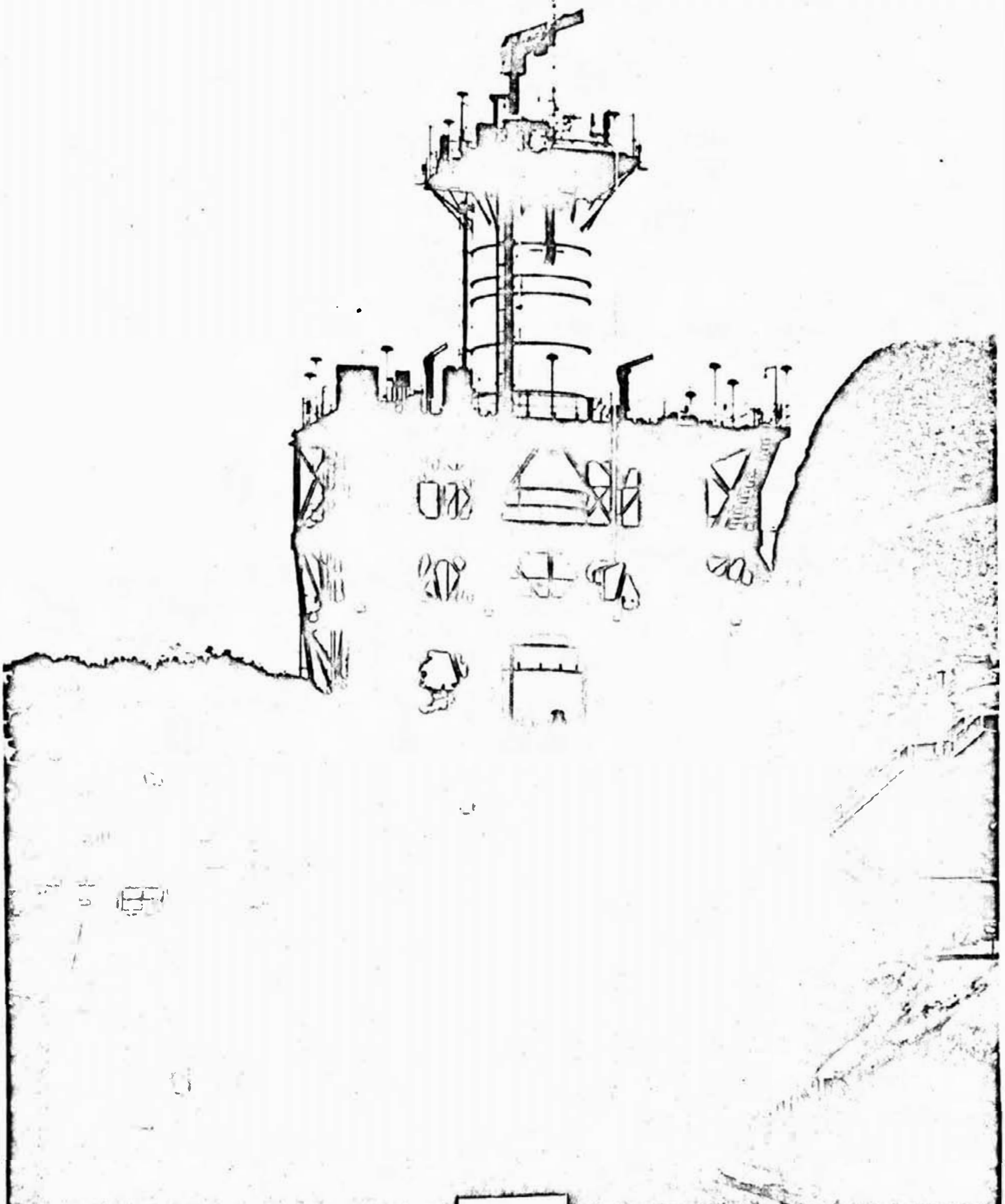

Fig 15

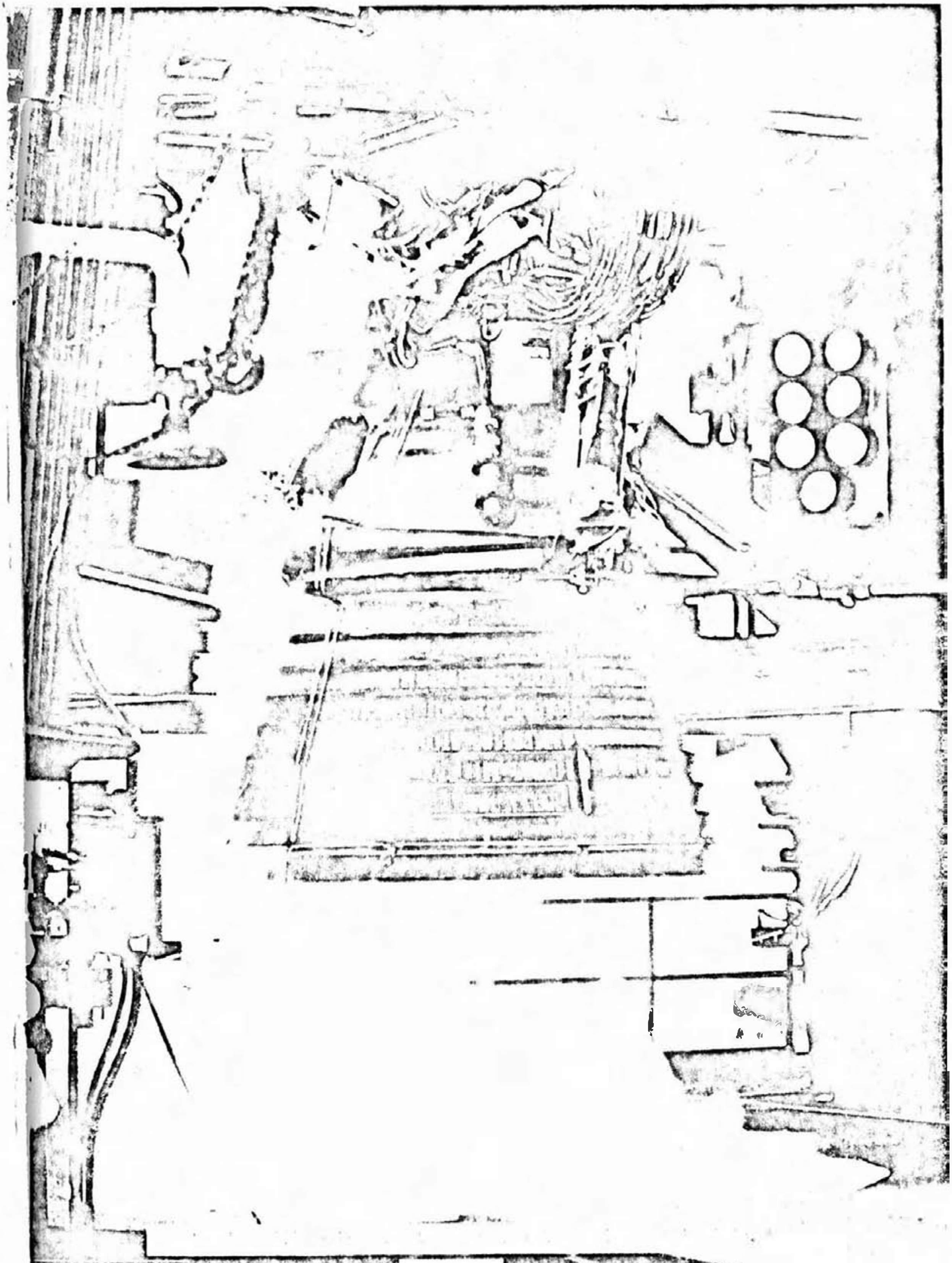

Fig 16

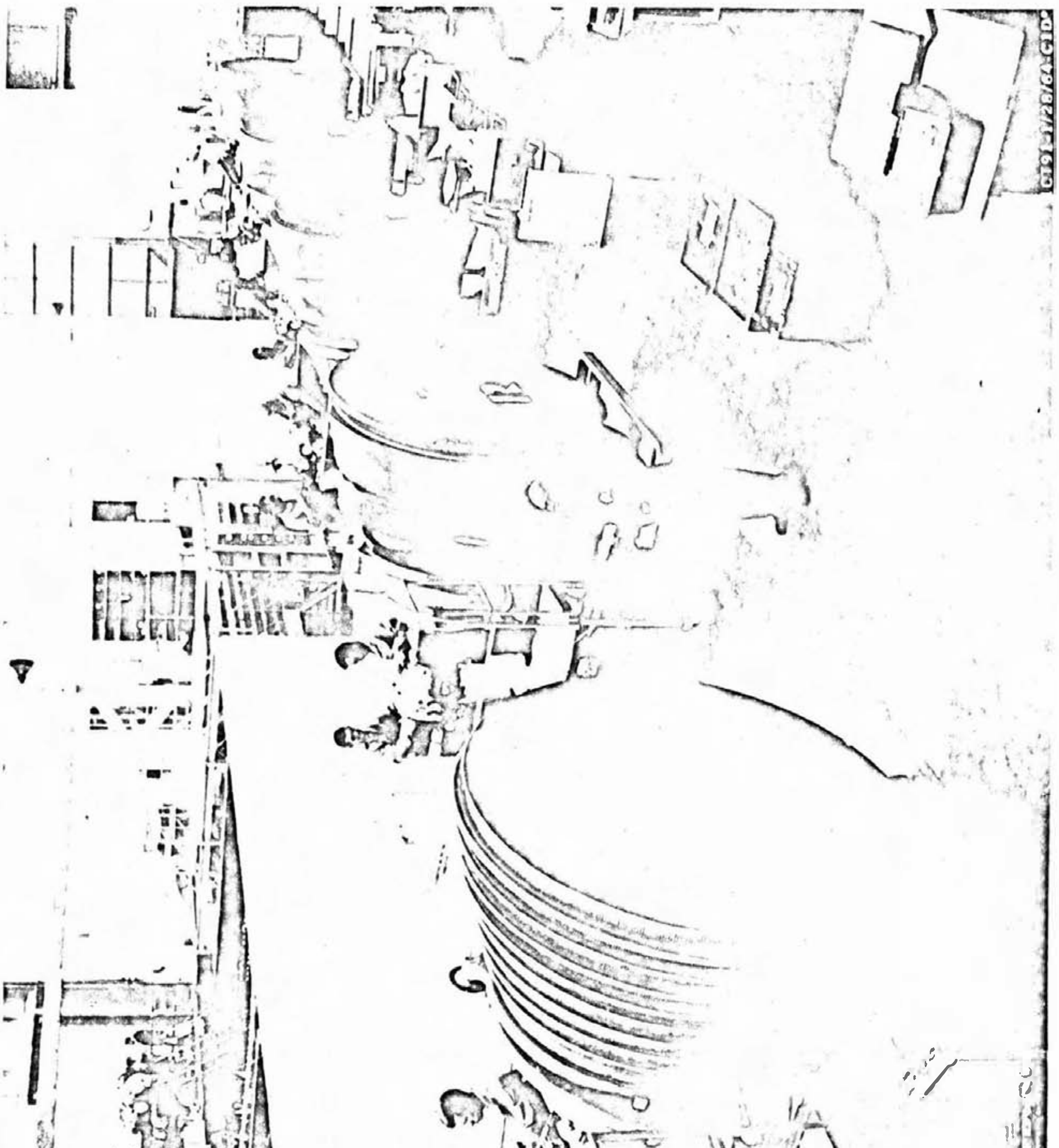

Fig 17

Fig 18

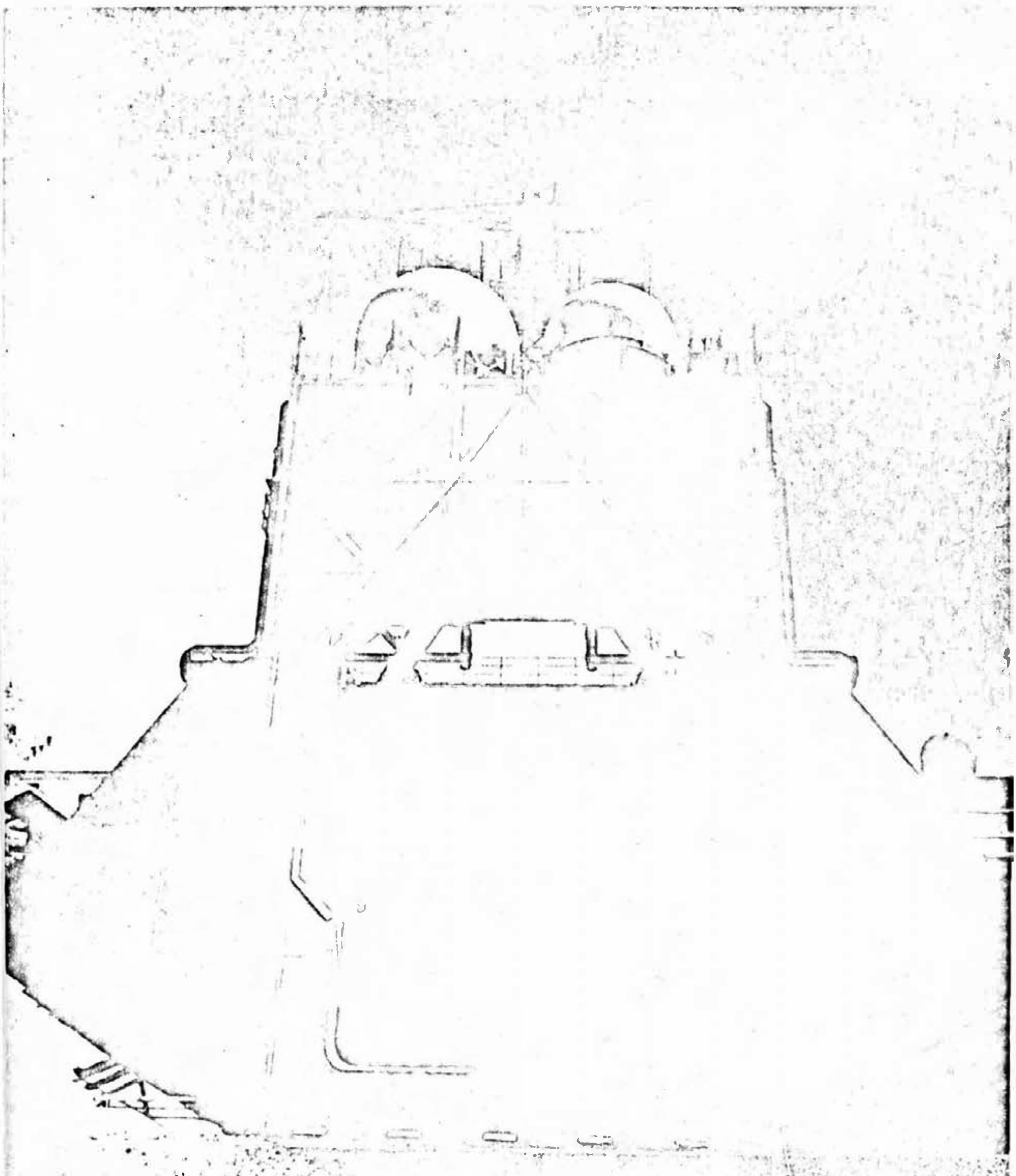

Fig 19

STRUCTURES AND PROPULSION

By Mr. E. A. Hellebrand

STRUCTURES AND PROPULSION

By Mr. E. A. Hellebrand

My presentation consists of six topics, each self-contained, but related to structural problems encountered in the development of a reliable Saturn/Apollo Launch Vehicle. The main problems as of today are in the prediction of dynamic responses of large, relative "soft," vehicle structures, in the analysis of orthotropic shell stability and in the development of a reliable reproducible high strength, ductile aluminum weldments of a great variety of weld thicknesses.

To tie in with the Apollo mission requirements, I will start out with Saturn V loads and structural limits. Figure 1 shows the design limit loads of the Saturn V vehicle. The three curves shown are the loads that dominate in the design of the vehicle.

The top curve shows the bending moment distribution over the vehicle while standing on the pad and subjected to a 99.9% wind condition. It should be noted that the pre-launch bending moment is higher than the in-flight bending moment in the aft areas of the first stage and thus contributes greatly in the design. The empty unpressurized configuration proves to be the most critical.

The middle curve represents the maximum in-flight bending moment and the associated axial load distribution along the vehicle. This $q\alpha$ Max condition occurs at t = 70 seconds.

The angle of attack and the gimbal angle which combine to give this moment curve are determined from design wind conditions aloft. Also included in the moment distribution are the dynamic effects of a free-free body.

The axial load distribution is the result of combining drag, thrust, and inertia forces. At $q\alpha$ Max the longitudinal acceleration is approximately 2 g's.

The bottom curve shows the axial load distribution for the maximum logitudinal acceleration during first stage boost. This condition occurs at maximum thrust just prior to cut-off. Due to the high altitude, the dynamic pressure is very small thus making the drag almost negligible. The preponderance of the loading is then due to thrust and weight. This maximum longitudinal acceleration is approximately 5 g's.

Figure 2 compares ultimate, failing, and design bending moment distributions along the Saturn V vehicle, at one time point in flight (70 seconds, q Max).

The top curve shows the ultimate bending moment of the vehicle in conjunction with the normal axial load. If these moment values are equaled or exceeded, a structural failure will occur in the vehicle. The S-IC stage is designed mainly by ground wind and also internal pressure conditions which raise the bending moment capability considerably.

The second curve shows the moment distribution for an angle of attack of 7.6° and a gimbal angle of 6°, i.e., all control engines hard-over. Engine hard-over is not a design condition, but is a failure condition (the guidance package may fail such that the engines gimbal to 6°).

As can be seen, this curve touches the ultimate moment curve at station 2800. The significance of this will be shown on Figure 3.

The design bending moment curve is presented as a static moment with body bending added. The lower curve defining the shaded area is the rigid body bending moment, while the upper curve is the rigid body moment with the dynamic effects superimposed. This curve is the design bending moment.

Figure 3 shows the critical angle of attack associated with certain gimbal angles for booster flight time.

The point indicated by the triangle on the lower curve is derived from the critical moment curve where it touches the ultimate moment curve.

The lower of the two curves gives the critical angle of attack in conjunction with all control engines hard-over ($\beta = 6^{o}$).

The upper curve shows the same thing for a gimbal angle of $\beta = 4^{o}$.

The purpose of this chart is to show that lower values of gimbal angle will increase the critical angle of attack. The same is true if negative gimbal angles are used with positive angles of attack. However, in such a case, extreme caution must be used since the buildup of lateral acceleration may then make the structure critical.

In considering these curves two facts must be noted:

1. The curves are not particularly smooth because different points on the vehicle may be critical at different flight times.

For instance, at 55, 61, and 70 seconds, station 2800 is critical; at 74 and 78 seconds, station 2520 is critical, and at 101 and 114 seconds station 2740 is critical.

2. After 90 to 100 seconds the curves become less significant because the dynamic pressure is so low that large changes in angle of attack will not change the moment to any great degree and leave it in the critical range.

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Now, let me discuss safety factors.

If it were possible to confidently identify the minimum strength of a structure as an event occurring only, say, once in 200 strength tests and to identify the highest expected = design = limit load as an event occurring only once in 200 flights, the DF could be one.

The failure rate of this structure would be less than 10^{-6}, a reasonable goal, even for manned space flight.

Limited knowledge and ground simulation of the environment, of the structural and material behavior of a vehicle and uncontrollable or undetected shortcomings in fabrication and assembly demand a DF $>$ 1.

An efficient design, however, is possible only if we properly separate the areas of limited knowledge from the uncontrollable influences and split the overall DF into definable load factors LF $>$ 1 or strength reduction factors RF $<$ 1 and confine the "remaining DF" to uncontrollables or undetectables.

The goal must be to minimize DF, but to identify uncertainties of assumptions exactly where they are known to exist.

For instance, instead of using an arbitrary DF = 1.5 across the board, apply individual LF $>$ 1 to gust loads, air density, engine thrust, engine start and cut-off transients and other dynamic influences, pressures and working temperatures.

Also apply individual reductions factors RF $<$ 1 to the strength of welds, of parts with geometrical, thermal or material discontinuities, and to the buckling strength of panels and shells of complex design. Also use fitting and bearing factors to alleviate stress concentrations.

Then the overall DF has to cover only uncontrollable or undetectable influences such as slight fabrication defects, residual stresses and local thermal and dynamic effects. It would cover discrepancies between ground simulation and the actual flight environment. Refined fabrication methods and rigorous quality control are prerequisites for this approach.

The DF would also cover discrepancies between the assumed and actual means and standard deviations of loads and strengths if a statistical design approach is employed. In order to provide this additional confidence a DF = 1.25 is considered adequate.

The whole question of design factors is a matter of proper and detailed bookkeeping.

It is more economical, efficient and safer to assign, for instance, a reduction factor of 0.80 to the average buckling strength of a complex honeycomb panel, to add 10% to the thickness of a weld island for discontinuities, to add X% to the static loads for dynamic influences and 20% to the average air density, but keep the overall DF as small as possible instead of using a high DF to take care of a multitude of unidentified uncertainties. A high overall DF adds considerable weight and prohibits optimum design.

As more knowledge through ground and flight testing increases the confidence in the structural soundness of a vehicle, these "individual safety factors"' may then be logically reduced and strength reduction factors brought back towards unity for a lighter, more efficient "second block" vehicle structure.

Extensive use should be made of statistical methods to utilize ground and flight test results, also of other vehicles, to estimate standard load and strength distribution curves and to predict failure rates rather than use general design factors.

The aforementioned buckling strength reduction factor of 0.80, for instance, could have been the result of three compression tests with relative strengths of 1.0, 1.1 and 1.2, a mean of 1.1 and a standard deviation S = 0.1. The reduction factor applied to the mean would then cover 2.2S or more than 97% of all probable values of panel strength.

The confidence in this prediction, however, should be increased by additional buckling tests. Money spent on well planned and properly evaluated ground tests is very worthwhile and is only a tiny fraction of the money lost in flight failures.

A proper choice of materials is most important. Ductile, tough and pliable materials can certainly be handled within the range of a MIN. OVERALL DF = 1.25. With brittle materials even an overall DF = 2 would not give us any guarantee of success.

As of today we have three major structural trouble areas: Quality of weldments, dynamic response, buckling of complex panels and shells. Individual LF or RF may be applied generously to these areas, but don't design pressure vessels to an overall DF = 1.4.

In those cases where a higher overall DF is chosen to increase mission flexibility, it is suggested to introduce a mission factor MF >1 if desirable.

A reliable emergency detection system and abort system (with a DF ≥ 1.5) eliminates the need for an overall DF > 1.25.

A reduction of overall DF changes the dynamic response characteristics. The dynamic thrust overshoot might require a LF = 1.5 with DF = 1.4 but will increase to LF = 1.7 with DF = 1.3. Still we reinforce only the structure effected by thrust and not the whole vehicle. We still reduce the overall structural weight.

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Configuration optimization facilitates man rating in two ways, by making maximum use of internal pressure to reduce compressive forces that lead to structural instability, and by reducing overall vehicle length to increase bending, torsional and longitudinal frequencies to minimize control feedback and coupling of longitudinal structural modes with liquid column modes and pump induced thrust oscillations, the "pogo effect." Structural instability is sensitive to a great many influences such as slight deviations in geometry, residual stresses and local discontinuities and a standard deviation of as much as 15% of the strength mean is not uncommon. Internal pressure increases the mean buckling strength and reduces deviations.

Increased lateral and longitudinal frequencies also reduce dynamic overshoots due to gust forces, thrust buildup, and rebound at engine shut off especially under pad abort conditions.

Vehicle length to maximum diameter ratios of from 8 to 10 lead to minimum weight designs. If this ratio drops below 8, bulkheads and thrust distribution structures become too large and too heavy and their spring rates become low enough to induce local dynamic effects on structures and liquids that again interfere with the functions of propulsion and vehicle control. A ratio above 10 increases lateral dynamic beam effects, reduces lateral frequencies and increases control feedback.

Figure 4 shows a two-stage Saturn class vehicle with typical configurations and some basic design details. The right half shows separate propellant tanks and one center engine. This arrangement will not yield minimum structural weights because large portions of the outer shell, not pressurized, are exposed to compressive forces from thrust and bending moments and only a small area in the upper tank region is supported by internal pressure. Relatively deep ellipsoidal bulkheads with an axis ratio of $\sqrt{2}$:1 to avoid compressive hoop stresses at the equatorial region and the long thrust cone are responsible for the excessive length of the unpressurized shell areas.

The left half shows a much shorter vehicle of equal performance capability. The length reduction is caused by a multiple engine arrangement, common bulkheads and ellipsoidal heads with an axis ratio of 2:1. Almost 70% of the outer shell is supported by internal pressure, against only 35% on the right. Internal pressure not only reduces compressive stresses in the shell from thrust and bending moments but also increases the buckling stress itself by as much as 70%.

The shorter vehicle also has higher bending -, torsional- and longitudinal-vibration frequencies, and is less susceptible to dynamic overshoots.

There are certain penalties to pay for this ideal configuration. The engines require aerodynamic fairings exposed to aerodynamic heating. The common bulkheads require insulation and must be reinforced to withstand possible pressure differences causing an unfavorable buckling condition. Shallower bulkheads must be reinforced at the equator because any axis ratio greater than $\sqrt{2}:1$ leads to compressive hoop stresses in that region.

Propellant tanks should be designed so that the cylindrical wall thickness required for hoop tension due to internal pressure and the dynamic head pressure coincides with the wall thickness required to withstand axial compressive forces from thrust and bending moments. Propulsion requirements, however, might necessitate pressures different from "optimum" conditions.

An upper stage with lighter propellants but a higher axial load factor under full tank conditions requires approximately the same diameter for minimum tank weight as does the lower stage. This results in a fairly streamlined aerodynamic configuration.

Different tank configurations are shown on Figure 5. On the left you see a conventional cylindrical tank. The multicell tank shown in the middle reduces the vehicle length by providing shallow bulkheads. It also allows for thinner skin gages with higher weld efficiencies. Sloshing frequencies are increased and effective sloshing masses are reduced.

If engines can be arranged such that the thrust is led into longerons between cells, the weight of thrust distribution members is minimized. A single engine, attached to the center post, counteracts the inertia forces of the center portion of the liquid; however, additional stability against "falling through" of the center portion of the tank has to be provided by reinforcing portions of the radial walls to convert them into rigid shear webs. A multicell tank with radial walls is stable under internal pressure but unstable under hydrostatic or liquid inertia forces. This holds true also for a toroidal tank or a cylindrical tank with semitoroidal bulkheads shown on the right. Both tank configurations help reduce overall vehicle length and weights and reduce the length of unpressurized intertank shells.

- o -

You have probably heard of the Pogo program.

Several large vehicles, notably Titan and Thor Agena, have developed longitudinal oscillations during flight. Characteristically these vibrations have started at a definite time in flight, reached a maximum amplitude at a frequency corresponding to the fundamental mode of vibration of the vehicle, and then died out before the end of flight.

Because the natural frequency of a vehicle is continuously altering during flight, due to loss of LOX and fuel mass, it is hardly surprising that some phases of the flight conditions have been just right for any small perturbation in thrust to be amplified by the structure and give quite large variations in LOX or fuel suction pressure which in turn have modulated the thrust to start the cycle all over again at ever increasing amplitude.

Analysis of observed POGO oscillations have conformed reasonably well with the type of instability just described. Theory has tied up very well with the onset of instability, less well with the fade out of instability and not at all with observed amplitudes. This is because vibration will start at the same time as instability starts, will not stop as soon as the system becomes stable again because the free vibration needs time to decay, and the amplitude is governed by non-linear effects which are not known.

It must be noted that all analyses to date have been performed on vehicles in which POGO has been observed - i.e., after the fact analysis, and has dictated the fixes which have been employed to eliminate the effect without complete insight into the theoretical problem.

On the Saturn V we are, for the first time, attempting an analysis on a vehicle which has not flown and we must rely entirely on theory. The presentation that follows gives the status of inhouse studies being made by the Dynamics and Loads Branch at MSFC.

Figure 6 shows the mass model presently being used to analyze POGO for first stage burning of Saturn V. The locations of mass blocks are shown in their relative positions with respect to the vehicle. The circles represent propellants and the rectangular blocks indicate dry structure. Figure 7 shows a block diagram with different areas of the vehicle shown in the proper block to describe the system. The first block represents the structure which is described with the aid of generalized coordinates utilizing the mode shapes of the structure. The second block represents the transfer function of the feed line for the LOX tanks. The mathematical model equation is derived from a figure like the one at the bottom. The spring on mass represents a cavitation bubble while the mass describes a column of uniform compressible fluid. This model uses the engine acceleration X_e, and the LOX tank pressure P_L as forcing functions which are obtained from the structure transfer function. Therefore, from the LOX line model an engine suction pressure is obtained for driving the engine transfer function. The engine transfer function was obtained from Rocketdyne and can be applied when the engine is operating at approximately maximum thrust. Therefore, this model will not simulate engine start up or shut down. From the engine transfer functions a change in thrust is obtained and can be applied to the structure as a forcing function which closes the loop of the system. The same block diagram is shown in **Figure** 8 with symbols which represent various transfer functions. The transfer functions can be put in the form of operational products, where the equation $\Delta T = \emptyset_3 \emptyset_2 \emptyset_1 T = GT$ is obtained. The term $\emptyset_3 \emptyset_2 \emptyset_1$ is equal to the open loop gain G. For the closed loop system the equation $T = T_1 + GT$ is obtained, from which follows: $T = \frac{T_1}{1 - G}$ The system is unstable when $G > 1$ and real. Figure 9 shows plots of the gain for various values of the line frequencies for the Saturn V vehicles. The dashed curve is for a line frequency of 6 1/2 cps. For this graph the line frequency is assumed to be constant throughout flight. A similar plot, Figure 10, was made for a line frequency that varies from 1 cps at lift-off to 5 cps at flight cut-off. The results show that the system is stable throughout flight for these line frequencies. By comparing the diagrams the importance of obtaining accurate propellant line frequencies becomes apparent. The line frequencies used are estimates only and the accuracy of these figures is not known at the present time. Final answers for these frequencies should be obtained from test results.

I would now like to discuss our progress in welding techniques.

After the selection of the 2219-T87 material for the S-IC stage of the Saturn V, a welding program was conducted to compare the MIG and TIG welding processes. Our welding engineers have determined that TIG welding is better for welding thicknesses through one inch because of many factors. Among these are better weld quality, better strength consistency, higher ultimate strength, and better arc penetrating ability permitting the use of square butt joint designs.

Following selection of the welding process, efforts were devoted to optimizing the selected process and to determining maximum levels for weld joint fitup tolerances such as misalignment and joint gap. In addition, quality control parameters were established, describing the limit of internal and external weld defects permissible in a weldment without a sacrifice in strength. In order to keep the defects within the maximum allowable limits, it was necessary to establish a vigorous weld joint cleaning operation. This entailed mechanically scraping the joint to remove all traces of aluminum oxide in the abutting edges and adjacent surfaces.

Following this and incorporating all discontinuities, such as porosity, undercut, joint fitup tolerances, etc., weld strength design allowables were determined statistically at a 99% lower tolerance limit having 95% confidence.

The major problem, which is still unresolved, is the excessive amount of porosity being encountered in production weldments. The mechanical scraping of weld joints is a manual operation, and because of the size of the component parts is awkward and time consuming. The inability to adequately scrape the joint clean and the time lapse between cleaning and welding result in less than a perfectly clean joint. This, of course, is one of the major reasons for the excessive porosity. This problem will be resolved as we develop better production techniques for manufacturing vehicles of this size.

North American Aviation, Space and Information Systems Division, prime contractor for construction of the S-II stage, conducted a program similar to the MSFC program. They, too, determined that TIG welding was superior to MIG welding in nearly the same aspects, in spite of the different alloy (2014-T6) and thickness.

North American Aviation has had no severe problem with porosity but has encountered an excessive amount of stringer defects which appears on a radiograph as a crack-like indication. It was disclosed that here again inadequate weld joint cleaning processes were the cause of the majority of these defects. Subsequently, North American has implemented our cleaning procedure. That is, the manual mechanical scraping of the weld joint in lieu of their procedure of using an abrasive wool. The stringer defect has since been essentially eliminated.

The S-IV stage contractor, Douglas Aircraft, also uses the alloy 2014-T6, but because of their many years experience with the MIG welding process, have chosen that process for welding fabrication. There is no serious objection to the use of the MIG process, and it has certainly demonstrated structural integrity during two live firings. However, our general belief is that TIG welding would improve the ductility of the weld and generally eliminate the weld cracking problem predominantly experienced in the common bulkhead.

Douglas is now investigating the TIG welding process in both a laborato ry program and a manufacturing development program. This program will consist of a comparison of performance of MIG and TIG weldments and a study of the ease of welding by each process.

In addition to the welding technology already developed for the major structure of the launch vehicles, other work is being done inhouse at MSFC and under contract to various organizations to develop welding techniques for other materials and for difficult design configurations.

Figure 11 shows statistical weld strength values for the 2219 aluminum.

- o -

Finally, I would like to report on another interesting matter.

The intertank structure, an unpressurized, corrugated shell with ring frames carries a large compressive load from both ground winds and inflight thrust and bending moments.

The original design was based on the Shanley-Tomoshenko method of analysis. This analysis has been the accepted method throughout industry for sizing circumferential frames for stiffened cylindrical shells of sheet-stringer design. The equations are based on test results of shells of relative small diameter subjected only to pure bending.

Four 1/8 scale model shells were tested. The first two tests indicated an average buckling stress of only about 65% of the analytically predicted value. Assuming the scale model to be nearly perfect, part of the weakness of the structure was found to lie in the choice of a "channel" ring frame with an effective moment of inertia of only 70% of the calculated value. Since the ring frames form the elastic foundation for the load carrying corrugations, this moment of inertia reduction in effect increased the buckling length of the corrugations, thus reducing the buckling stress.

The third specimen with heavier I-section rings failed at 80% of the predicted buckling stress.

The fourth specimen with I-rings of twice the moment of inertia of those used in the third specimen failed at slightly more than 80% of the predicted buckling stress. This time the failure mode was buckling of the end bay which is exposed to severe local discontinuities and eccentricity of the primary compression load.

Present day analytical methods are still not quite adequate to predict complex local failure modes and a solid ground test program is indispensable. It is of interest to note here that the 1/8 scale model, relatively speaking, had greater geometric imperfections than a series of flat, full scale panels, simulating part of the front skirt of the S-IC stage. These panels buckled on the average at 104% of the predicted load. Pending full scale test results, an increase of ring stiffness and wall thickness of corrugations especially in the end bays of the intertank structure will provide a 100% load capability with a minimum of weight increase.

Let me mention here also that other more recent methods of analysis of the intertank structure were tried out.

In the Hedgepeth method a stiffness matrix is constructed which is the sum of three components, the cylinder stiffness, the frame stiffness, and the driving force matrix. Tractions Y and Z, which are defined as the resultants of unbalanced forces on an element in the hoop and radial directions, are taken at the radius of the centroid of the corrugations and are eccentric to the frames. The tractions are related to the displacements in terms of the material and geometric properties and the wave length parameters. Since, at the time of buckling, the displacements become infinitely large, the determinant of the stiffness matrix must vanish and the problem reduces to calculating the end load that makes the determinant equal to zero.

The Almroth method is based on orthotropic shell theory with the ring frame area and moment of inertia being smeared over the length. By applying the shell theory a classical load is obtained. Almroth then uses van der Neut's formulation for the ring eccentricity which reduces the classical load in some cases as much as 50%.

The Hellebrand method of analysis gives an equation for the critical load assuming each pitch of corrugated skin to act as a beam on elastic foundation. The foundation constant used in this equation corresponds to the spring constant of a circular ring acted on by a concentrated radial load and reacted by tangential shear in the ring. This method was used to design the Jupiter corrugated tail section and is conservation for light rings and small radii.

A comparison with test results indicates a large spread. Assuming the 1/8 scale model results to be represented by 100, we find for the four analyses results indicated on figure 12. This figure shows the variability of analytical results with several cross-over points of the different methods employed. Extensive ground testing is a must.

SATURN V LOR
LIMIT LOADS VS STATION

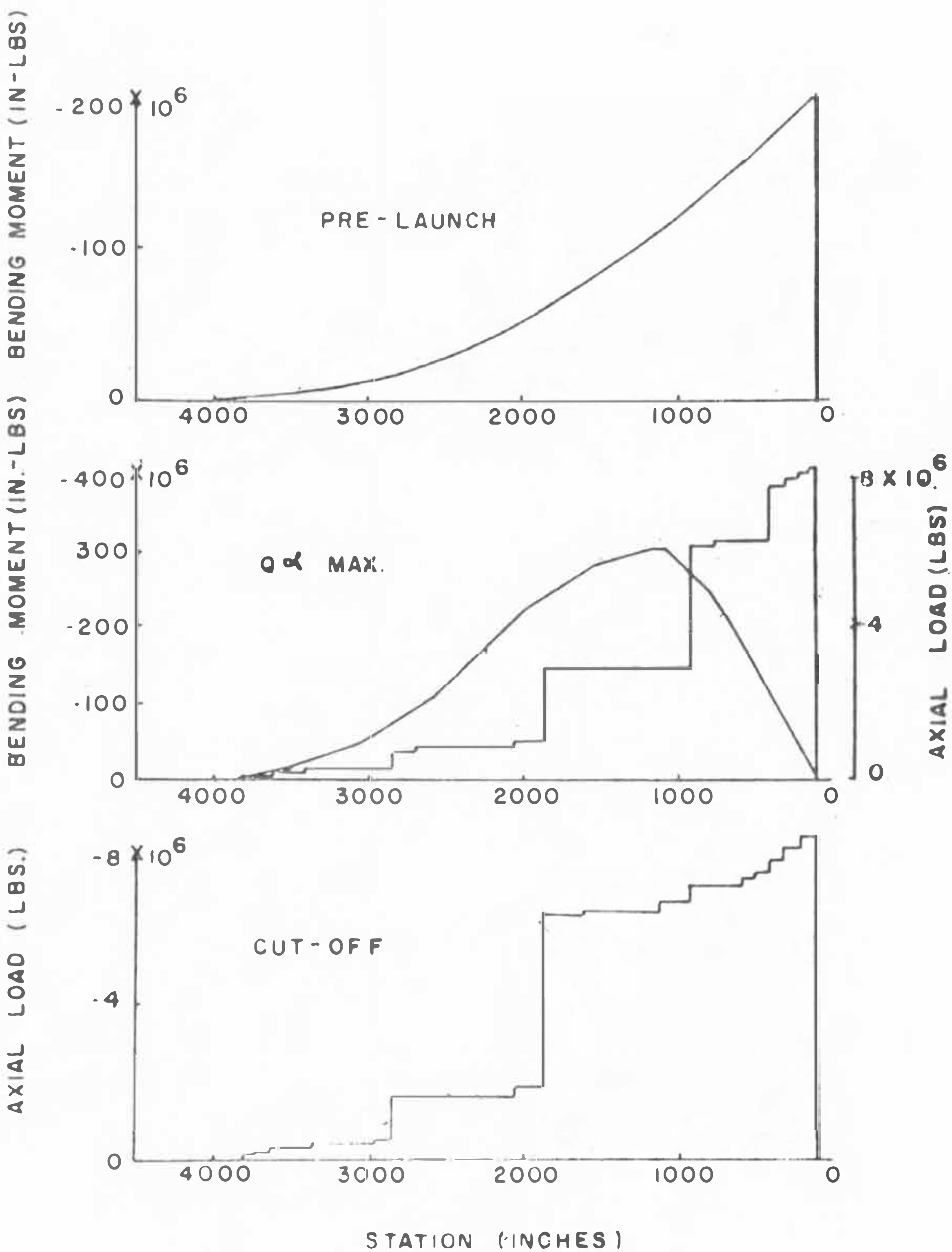


Fig 1

SATURN V LOR
BENDING MOMENT VS. STATION
t = 70 SEC.

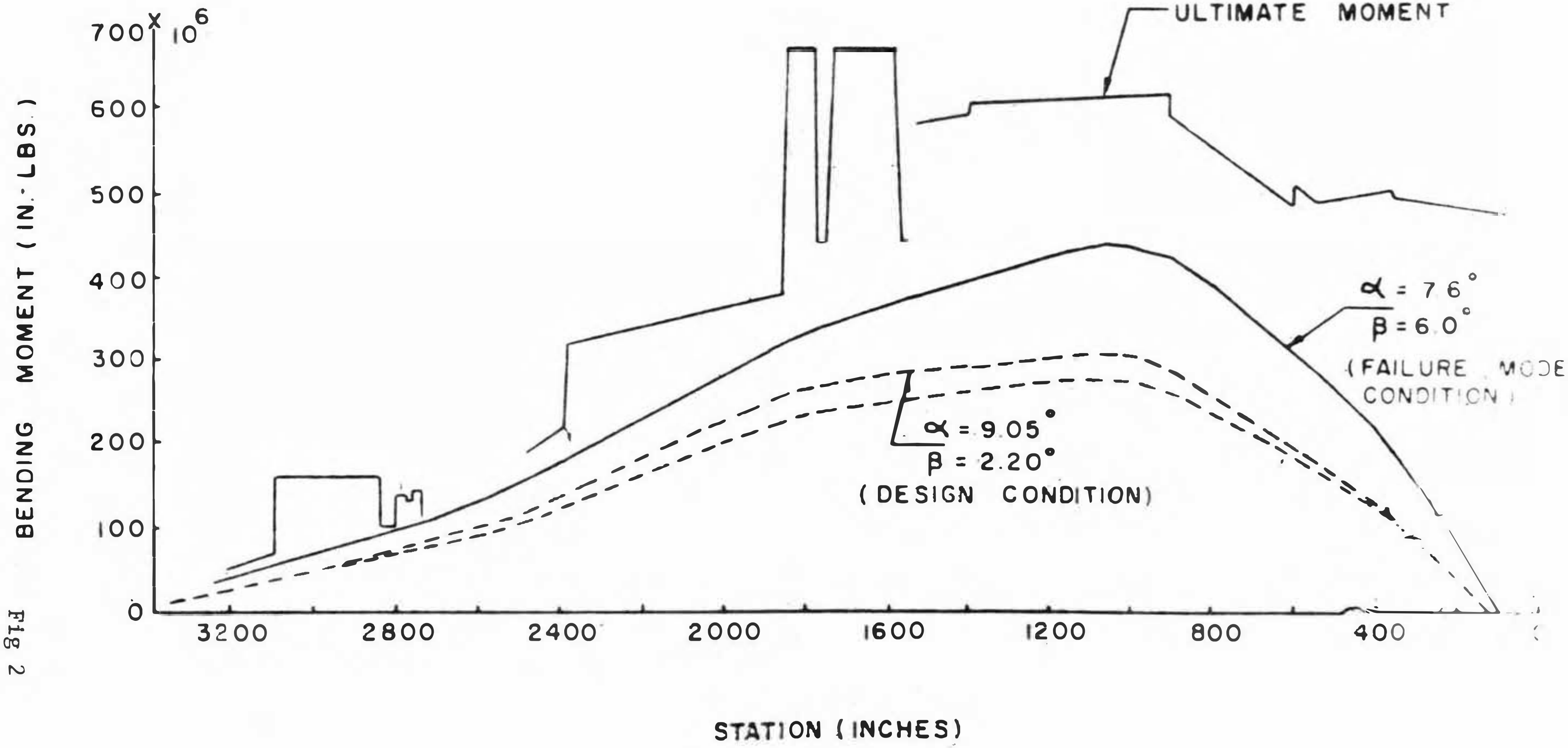
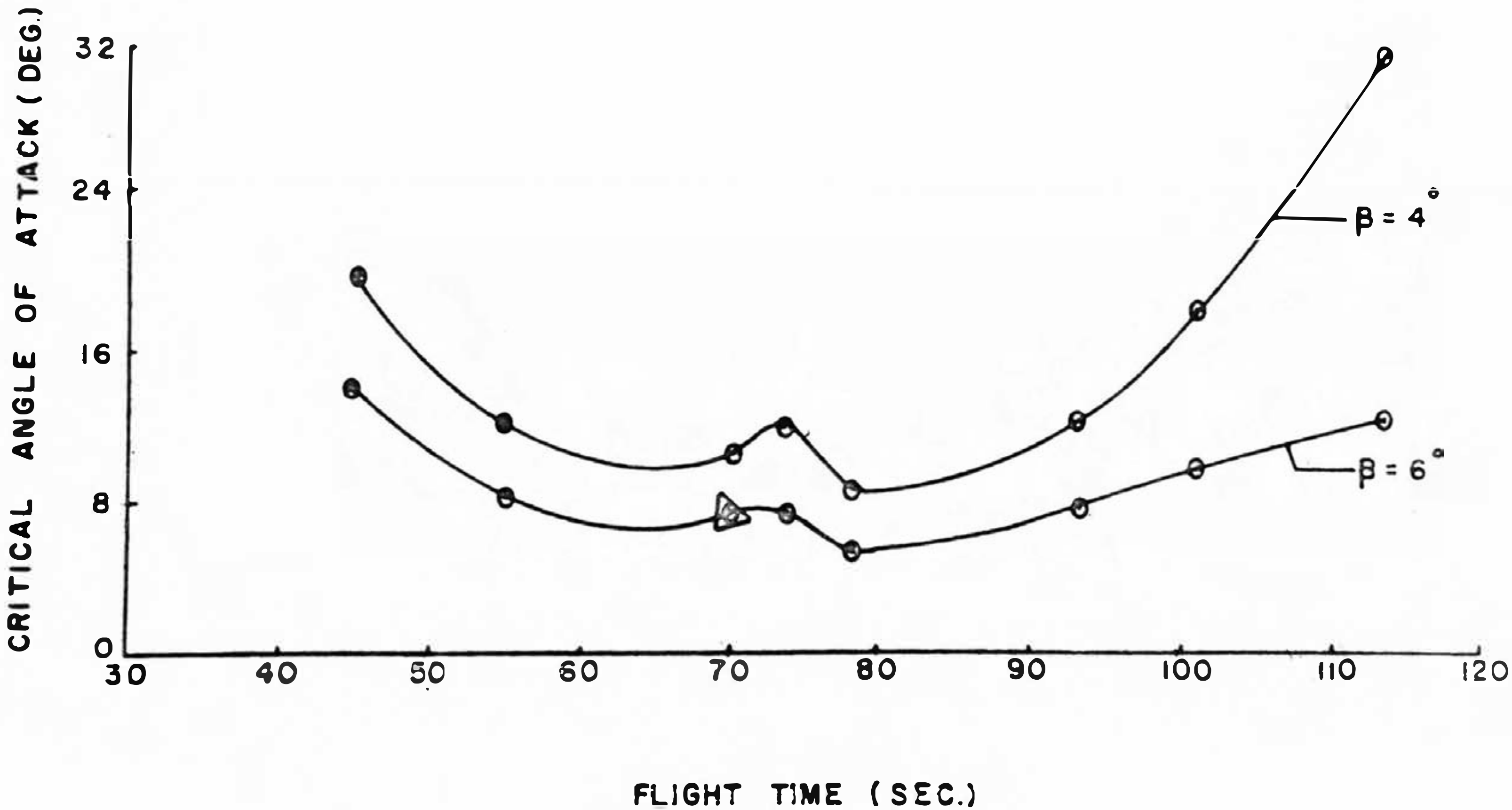
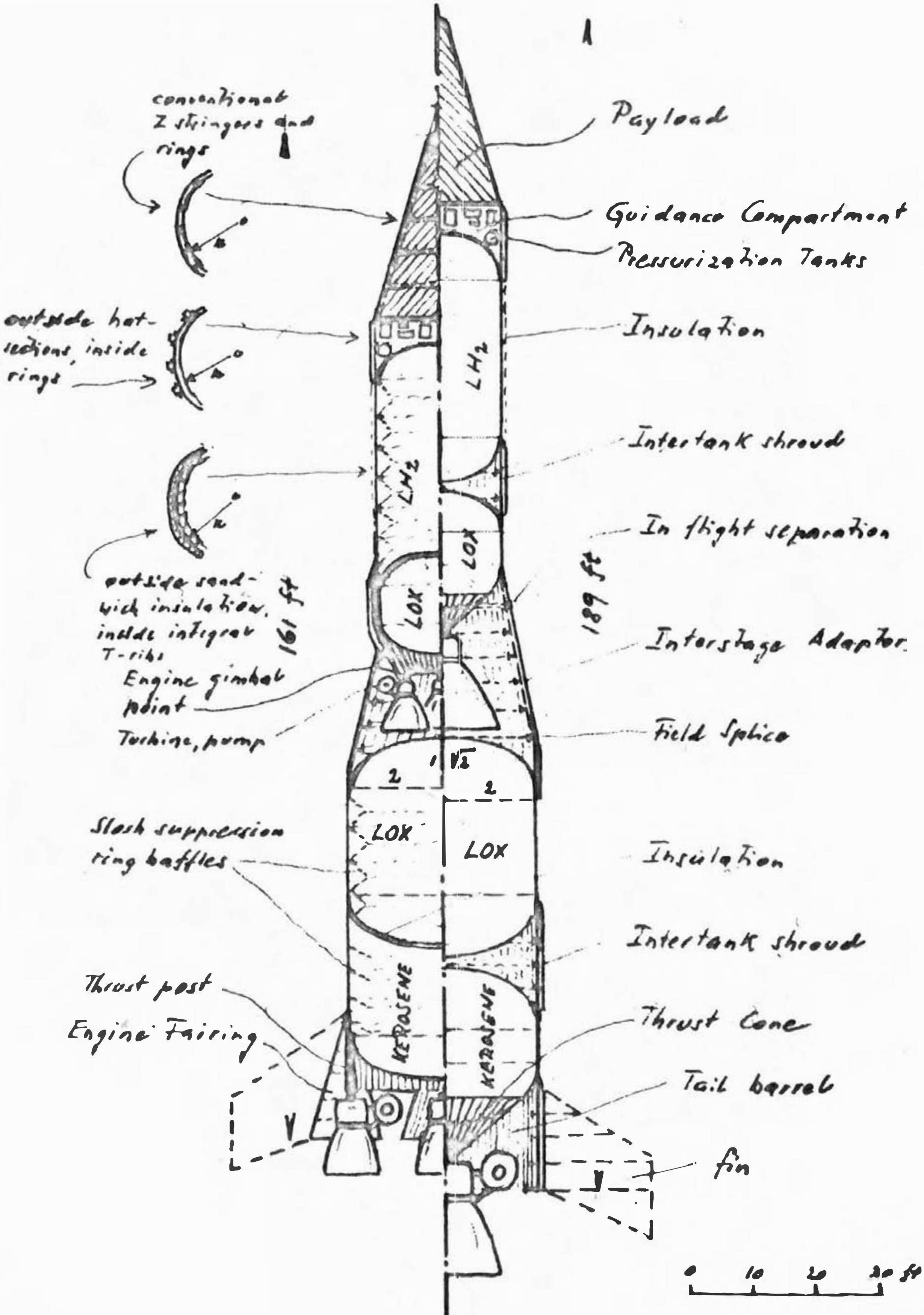


Fig 2

SATURN V LOR

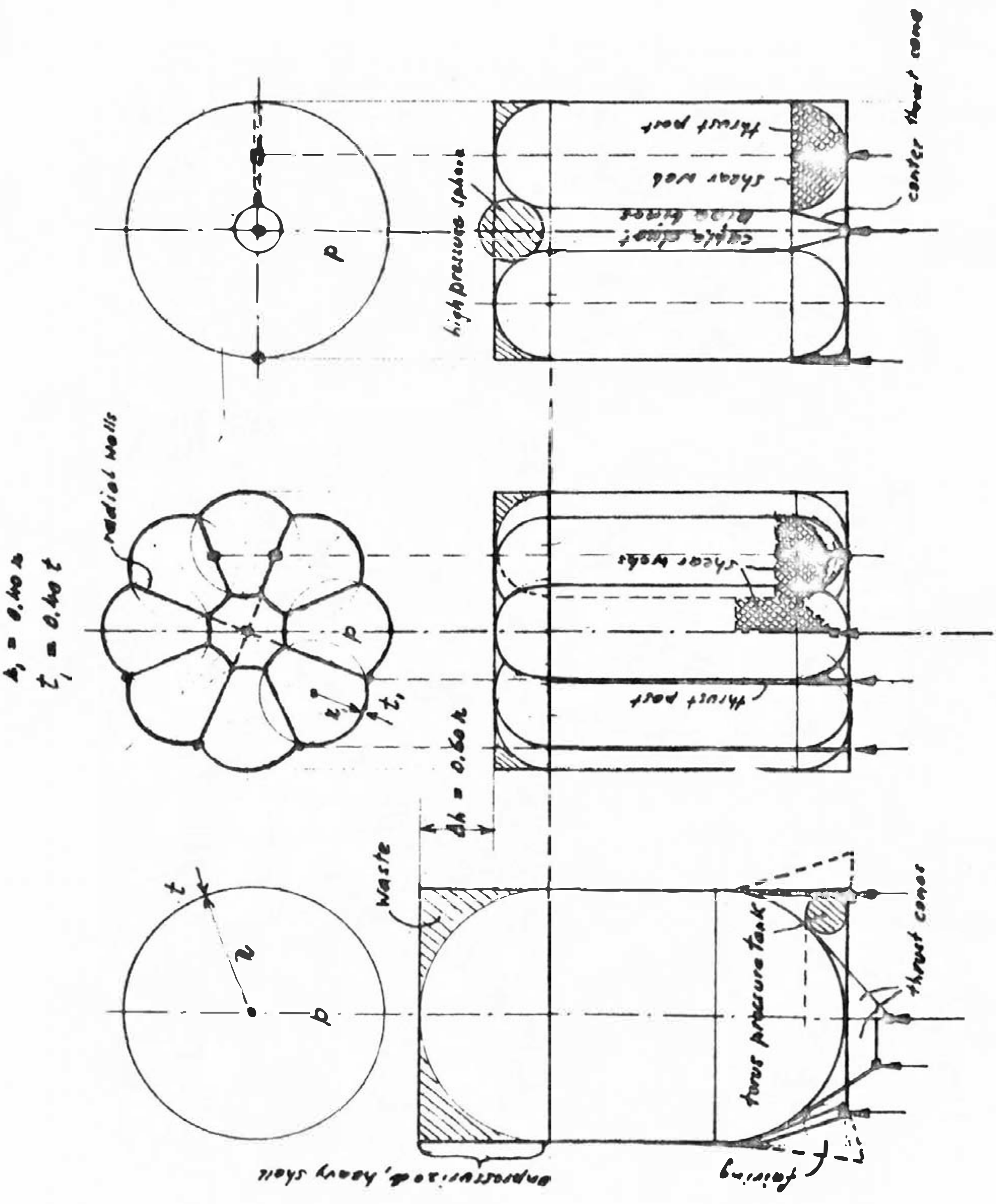
CRITICAL ANGLE OF ATTACK VS. FLIGHT TIME





I
conventional Z stringers and rings
Payload
Guidance Compartment
Pressurization Tanks
outside hat-sections, inside rings
Insulation
LH_2
LH_2
Intertank shroud
In flight separation
LOX
LOX
outside sandwich insulation, inside integral T-ribs
161 ft
189 ft
Interstage Adaptor
Engine gimbal point
Turbine, pump
Field Splice
2
1 √2
2
Slosh suppression ring baffles
LOX
LOX
Insulation
Intertank shroud
KEROSENE
KEROSENE
Thrust post
Engine Fairing
Thrust Cone
Tail barrel
fin
0 10 20 30 ft

Fig 4



$h_1 = 0.40\,h$
$t_1 = 0.40\,t$
radial walls
$\Delta h = 0.60\,h$
waste
unpressurized, heavy shell
torus pressure tank
thrust cones
fairing
shear webs
thrust post
shear web
high pressure sphere
center thrust cone

Fig 5

PRELIMINARY MASS MODEL OF SATURN V LOR

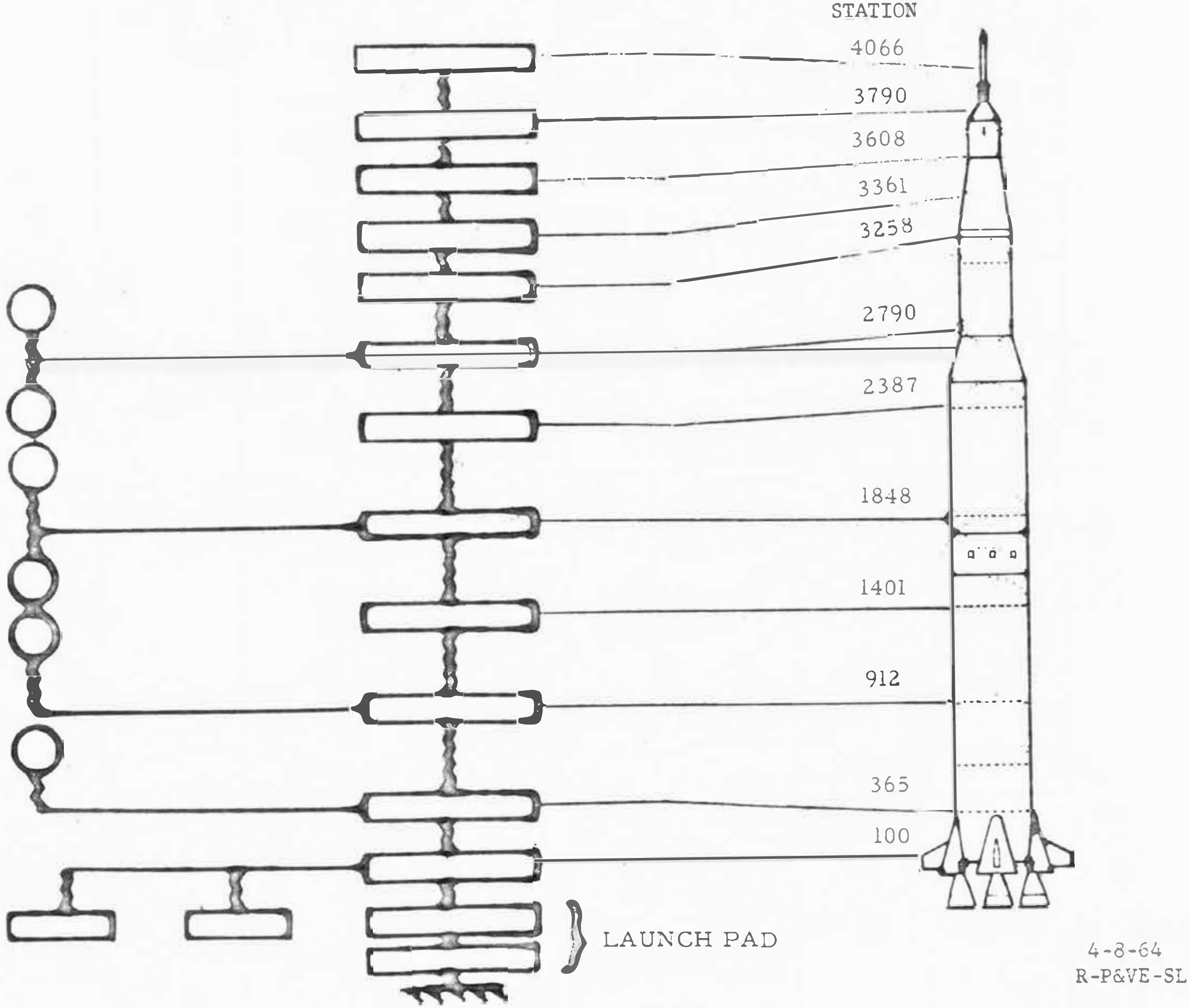
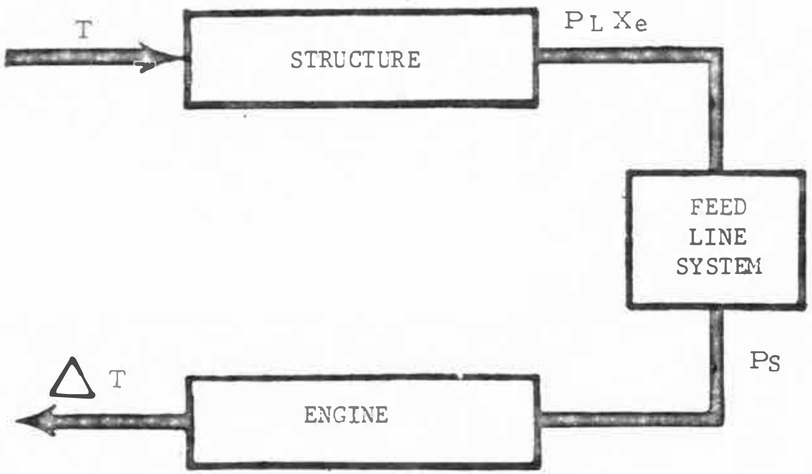


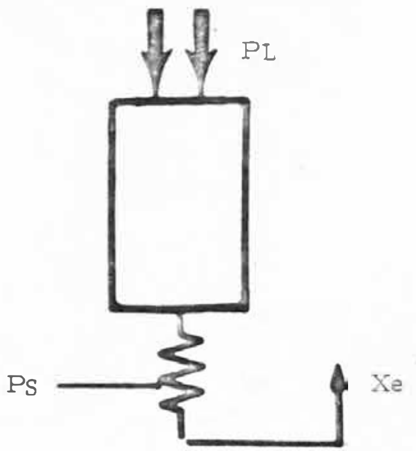
Fig 6

PRELIMINARY SATURN V MODEL FOR
OPEN LOOP POGO STUDY

SIMPLIFIED BLOCK DIAGRAM



FEED LINE SYSTEM MODEL



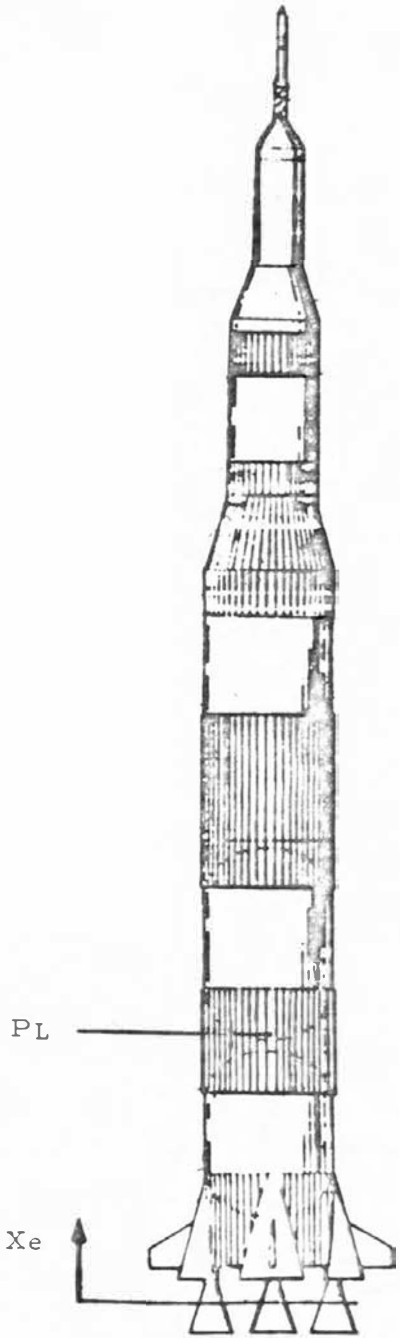


Fig 7

4-7-64
R-P&VE-SL

EQUATIONS FOR SATURN V AND IB OPEN LOOP POGO STUDY.

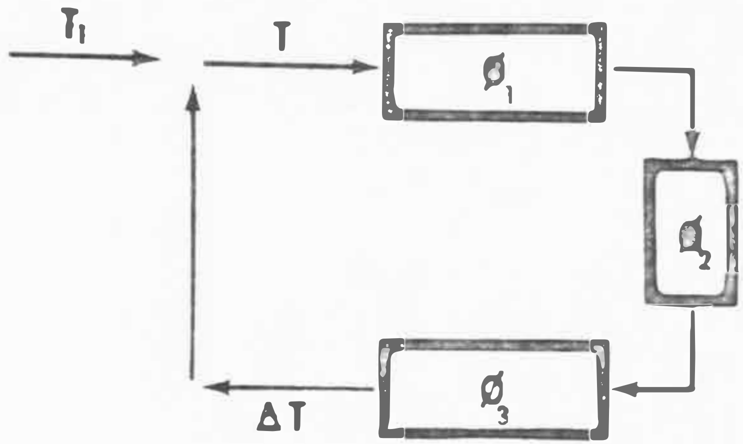


Fig 8

STRUCTURE TRANSFER FUNCTION $= \varnothing_1$

FEED LINE TRANSFER FUNCTION $= \varnothing_2$

ENGINE TRANSFER FUNCTION $= \varnothing_3$

$$\Delta T = \varnothing_3 \varnothing_2 \varnothing_1 T \equiv GT$$

$G(iw) =$ OPEN LOOP GAIN

ON CLOSING LOOP

$$T = T_1 + \Delta T = T_1 + GT$$

$$T = \frac{T_1}{1 - G}$$

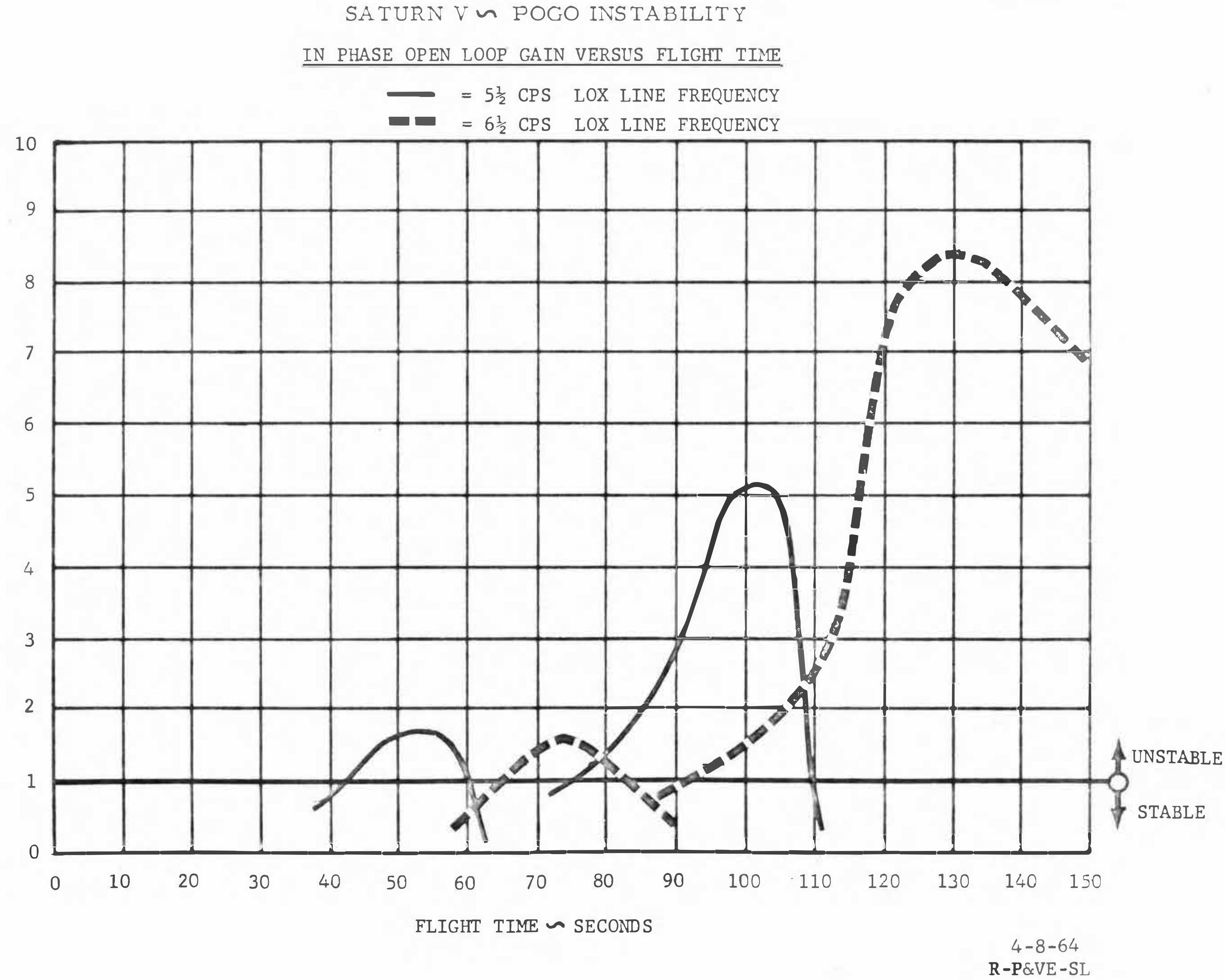
IN PHASE OPEN LOOP GAIN, G_P, IS VALUE OF $G(i\omega)$

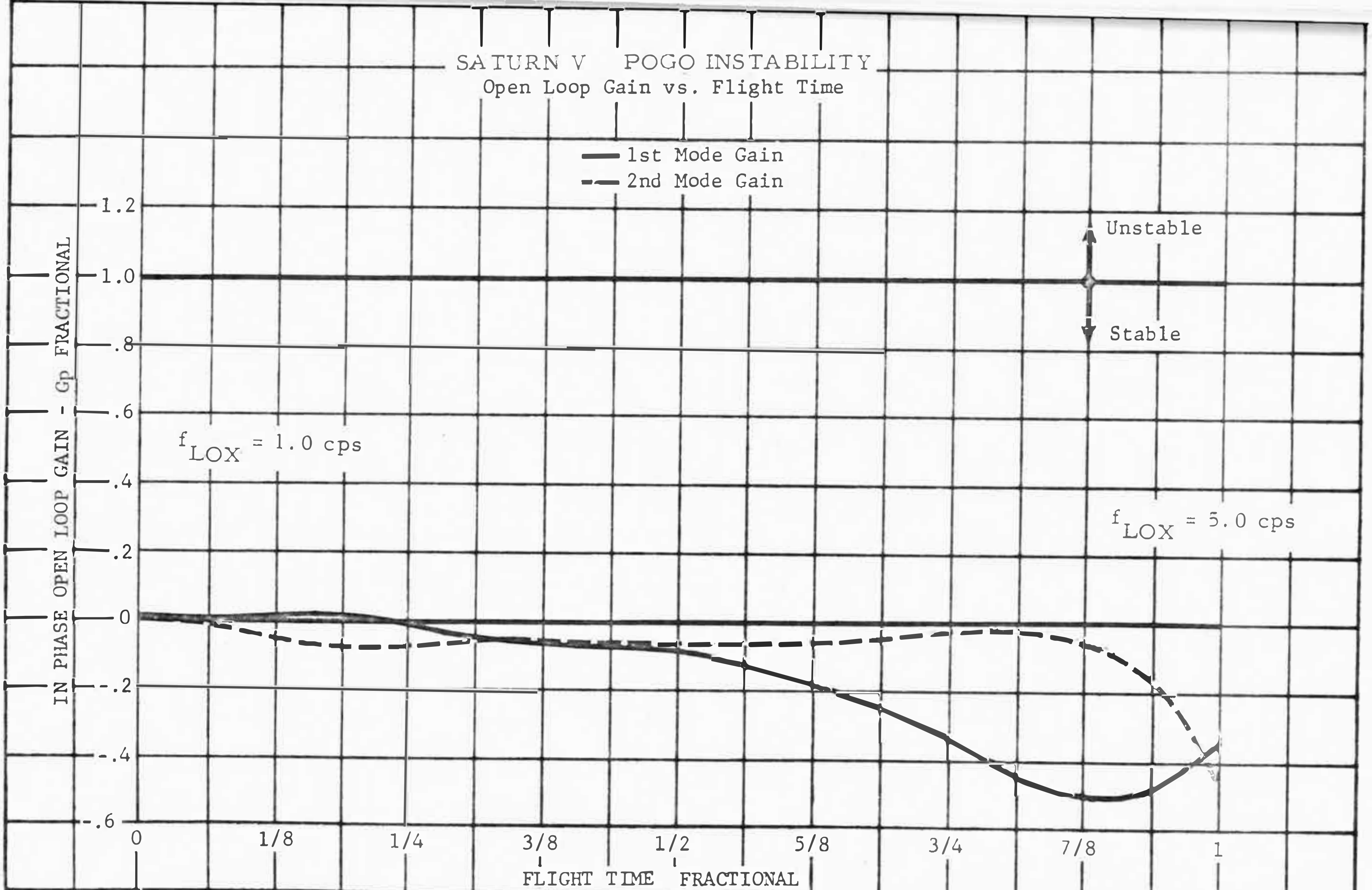
WHEN ΔT IS IN PHASE WITH T

CLOSED LOOP SYSTEM IS UNSTABLE WHEN $G_P > 1$

Figure 4

4-8-64
R-P&VE-SL


SATURN V ∽ POGO INSTABILITY
IN PHASE OPEN LOOP GAIN VERSUS FLIGHT TIME
= 5½ CPS LOX LINE FREQUENCY
= 6½ CPS LOX LINE FREQUENCY
IN PHASE OPEN LOOP GAIN
0
1
2
3
4
5
6
7
8
9
10
0
10
20
30
40
50
60
70
80
90
100
110
120
130
140
150
UNSTABLE
STABLE
FLIGHT TIME ∽ SECONDS
4-8-64
R-P&VE-SL
Fig 9



SATURN V POGO INSTABILITY
Open Loop Gain vs. Flight Time
1st Mode Gain
2nd Mode Gain
Unstable
Stable
f_{LOX} = 1.0 cps
f_{LOX} = 5.0 cps
IN PHASE OPEN LOOP GAIN - Gp FRACTIONAL
1.2
1.0
.8
.6
.4
.2
0
-.2
-.4
-.6
0
1/8
1/4
3/8
1/2
5/8
3/4
7/8
1
FLIGHT TIME FRACTIONAL
Fig 10

FIGURE 6

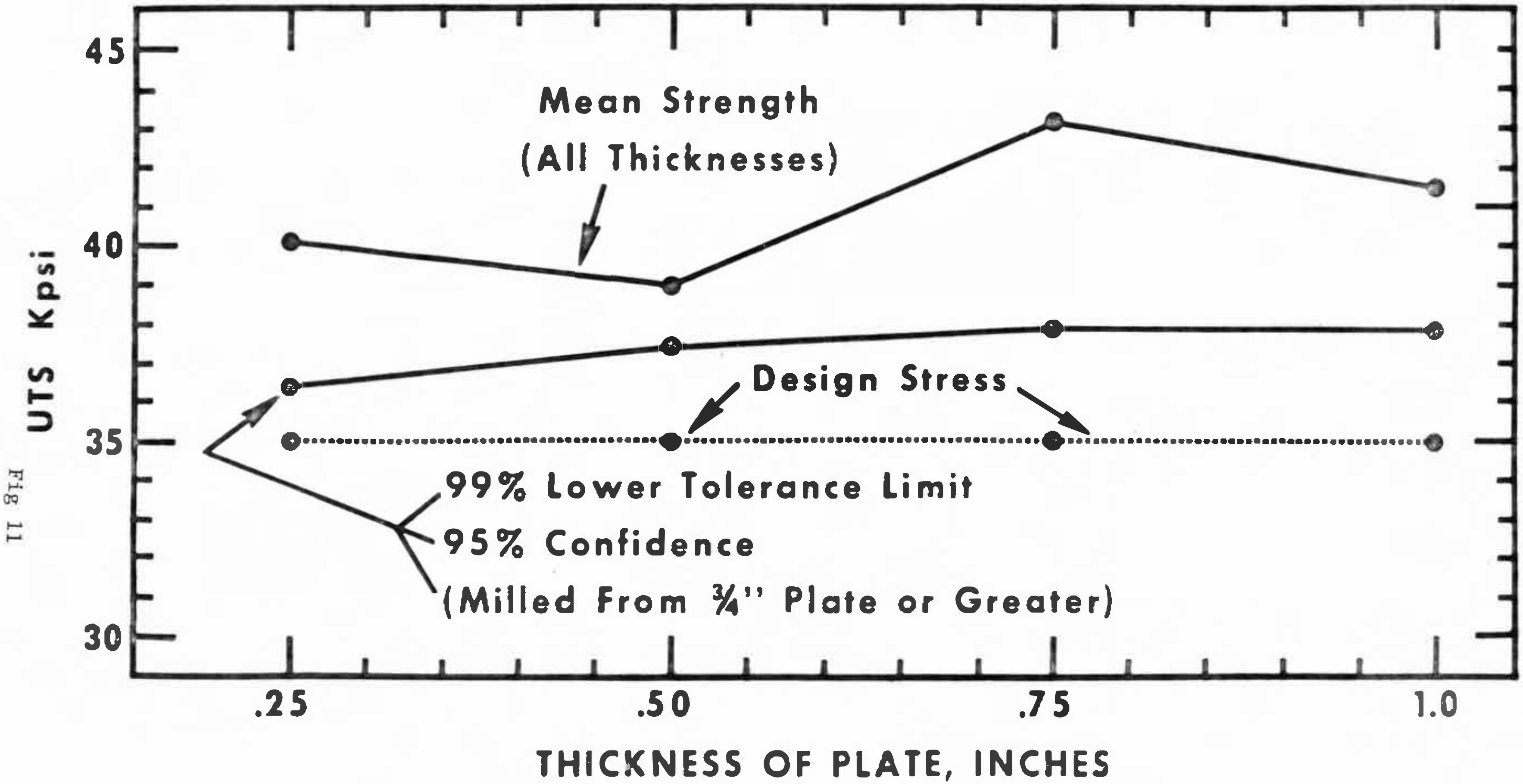


Fig 11

FAILURE STRESS OF 2219-T87 "AS WELDED"

Fig 12

Comparison of Analyses and Tests of Corrugated Shells

		Test	S-T	H	A	H	I_{ring}
1/8 scale model	1	100	162	99	83	67	1
	2	100	-	104	109	74	2
	3	100	-	117	141	98	4
proposed		-	100	80	76	87	">4"
prototype		-	-	100	89	121	~8

ASCENT TRAJECTORY CONSIDERATIONS

By Dr. E. D. Geissler

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The Lunar Orbit Rendezvous mode was selected as the best technique to land a man on the moon within the framework of the Apollo Program. There are three basic trajectory elements of the LOR mode from earth to the moon. These three are, of course, the trajectory from earth to orbit, the trajectory from earth orbit to lunar orbit, and finally the descent trajectory from lunar orbit to the lunar surface. My comments today are concerned with the first two of these trajectory elements which result in the achievement of a lunar parking orbit.

Perhaps it is interesting to note that some performance is given up (of the order of 3, 000 lbs. at lunar injection) by going through an earth orbit. But in order to take advantage of the higher payload, launch would have to take place at a single instant of time within a given month. This singular point occurs because injection into the lunar transit trajectory must take place on approximately the earth-moon line on the side of the earth opposite to the moon. This injection point cannot be reached in general without a coast time in earth orbit.

Earth orbits with the freedom of variable coast times and, therefore, easier access to the injection point in orbit increase the launch window to approximately four hours and they occur twice daily.

Earth orbit also provides an opportunity for adequate tracking of the vehicle which, in turn, makes possible more accurate injection into the lunar transit trajectory.

We have mentioned that the payload sacrifice to go through earth orbit is of the order of 3, 000 lbs. The exact amount of the sacrifice in payload is, to some extent, dependent upon the height chosen for the earth orbit.

One-hundred nautical miles has been chosen as the best compromise between performance, which increases with decreasing attitude, and the desire to get up higher because of aerodynamic heating; also, tracking and telemetry favor altitudes higher than 100 nautical miles.

The design of trajectories which connect Cape Kennedy on the earth with a selected landing site on the moon requires consideration of the overall geometry which is somewhat complicated by the rotation of the earth, the rotation of the moon, and the motion of the moon around the earth. Suffice it to say, however, that it is possible to design optimal trajectories for nominal flights where no powered plane changes are required in the portion of the trajectory from earth to lunar orbit. Small powered plane changes may be required in the descent trajectory to the moon.

During early launches of Apollo vehicles to the moon, it is desirable that an additional condition for free return trajectories be enforced. In this case, only landing sites on the moon near the lunar equator may be reached without powered plane changes while satisfying the free return constraint. Early lunar flights have landing sites near the lunar equator. Later flights to other sites must give up the free return constraint in order to avoid powered plane changes.

Within certain limitations, the trajectory from earth to earth orbit may be considered independently of the total mission. Of course, the inclination of the earth orbit and consequently the launch azimuth as a function of launch time must be obtained from a consideration of the total mission profile. For the following discussion we assume the launch azimuth to be known.

An obviously desirable goal of the boost trajectory into earth orbit is to obtain the maximum payload in earth orbit. For a circular orbit at a fixed altitude, the thrust direction time history required to deliver maximum payload is closely approximated by a linearly changing time function. This mathematically optimum thrust direction time history is not practical to fly because of other propelled flight trajectory shaping constraints.

For example, there is the practical necessity for a vertical launch and subsequent vertical flight during the first twenty seconds of flight in order to minimize liftoff dynamics problems.

Additionally, vehicle control and stability problems within the atmosphere along with structural load limits dictate that some restriction be placed upon angles of attack, engine swivel angles, and vehicle angle rates. These limitations must be enforced even in the presence of wind gusts and wind shear, possible engine out conditions, center of gravity offsets, and thrust misalignments.

Other constraints are placed on the trajectories because of even tighter restrictions on angles of attack and vehicular angular rates during the times of separation of the various stages. Collision of the separated stages must be prevented, and torques produced by the separation process must be held within reasonable bounds.

It is within this set of constraints that the trajectory shaping to achieve maximum payload in orbit must be carried out. A brief outline of this optimization procedure is given in the following paragraphs.

A completely nominal vehicle is used to determine the optimum trajectory. This means nominal weights, thrust, fuel consumption and aerodynamic parameters.

The vehicle lifts off the pad vertically and immediately performs a roll maneuver into the correct azimuth. The vehicle continues to fly vertically for approximately twenty seconds. At this time an angle of attack is introduced which causes the vehicle to pitch over into the proper launch azimuth. After the pitch over is initiated, the vehicle flies a zero lift trajectory to the end of first stage burn.

Beginning with the ignition of the second stage, the thrust direction is determined by the calculus of variations so that maximum payload is achieved for the particular zero lift trajectory of the first stage. Since there is a complete family of first stage trajectories with zero lift, it is necessary to select from this family the single first stage trajectory which provides the maximum payload to orbit. This selection is accomplished by repeating the process described above for different initial angles of attack to begin the pitch over from vertical flight.

When this nominal trajectory is completed, the control and stability problems, liftoff motion studies, sloshing, bending and separation studies are made to confirm a satisfactory trajectory.

As was pointed out earlier, this process does not produce the absolute maximum payload in orbit. Maximum payload is not achieved because of the restriction being placed that the first stage fly with zero angle of attack. At the present time we are studying the possibility of relaxing this constraint on the angle of attack after the maximum dynamic pressure region has been passed. Early indications are that the introduction of positive angles of attack after max q will add 3,000 lbs of payload in orbit for the Saturn V vehicle. Questions still remain whether this type trajectory may be permitted due to increased control, stability and structural problems.

After payload optimizations have been completed, another interesting phase of the boost trajectory to orbit is the calculation of propellant reserves which are carried by the S-IVB stage.

The propellant reserve which is provided in the S-IVB stage of the Saturn V vehicle is normally considered to consist of two separate reserves, the flight mechanical performance reserve (FPR) and the flight geometry reserve (FGR). The purpose of these reserves is to assure, up to some level of probability, that the vehicle will have the freedom to fly nominal LOR missions under any earth moon geometry and the capability to correct for errors in profile execution.

Before the flight geometry reserve can be determined, an LOR reference profile must be defined. We make the following definition:

Reference Profile

a. Launch occurs from KSC

b. Launch azimuth 72°

c. Parking orbit coast will be less than 4.5 hours

d. No powered plane change

e. Mission in 72 hour flight time to the moon at mean distance

Due to the motion of the moon and the requirement that launch windows be provided, both on the pad and in earth orbit, it is necessary to deliberately introduce variations away from the reference profile just defined. Additional perturbations will come about due to hardware errors and imperfect guidance laws and execution. A ground launch window is required for the Saturn V since the vehicle cannot be launched at a given instant which is specified a long time in advance. A launch window can be achieved if the launch azimuth is allowed to vary so that the plane defined at launch contains the moon at arrival time. The azimuth (measured East of North) range which is presently considered is from 72° to 105°. This range will produce two launch windows daily, each of which is approximately four hours in duration.

It would appear at first that it is necessary to have allowances for the ground launch window in a propellant reserve. However, the reference profile was designed for a 72° azimuth which has a greater required velocity increment than any other azimuth in the range considered. This implies that the ground launch window can be achieved without a propellant reserve. (An additional 500 lbs of payload can be injected into orbit for an azimuth of 90°).

It will be necessary to provide a launch window for the ignition in orbit. A launch window of 60 seconds requires a velocity increment of 5.5 m/sec. A 30 second launch window can be achieved for 1.5 m/sec.

The velocity increment required to correct a launch azimuth error not greater than .1° is less than 4 m/sec.

Even though there is a launch window available for orbit ignition, it may be impossible to ignite in the first launch window. It would then be necessary to go around the parking orbit one more revolution. Generally the parking orbit will not be coplanar with the moon travel plane, therefore, when the vehicle stays in orbit one more revolution and flies with the planned flight time, the moon would have moved away by the time the vehicle arrives. To correct this, the vehicle must either make a plane change if flight time is fixed or decrease the flight time to the moon by one parking orbit perior (fixed arrival time) and make no plane change. If the fixed flight time is used, the plane change and velocity increment is dependent on the inclination of the orbit plane to the moon travel plane. For a given launch azimuth in general, one launch is possible with a high inclination of the launch plane and the second one is achieved with a low one. The velocity increment for low inclination is 15 m/sec. and a velocity increment of greater than 100 m/sec. is required for the high inclination launch window. These values depend upon the relative inclination of the orbit plane and the moon travel plane. These relative inclinations vary not only during the daily launch window but also with the launch date. The values given here are for the years 1968-1969 when the moon travel plane reaches its maximum declination with respect to the earth equator. They are, therefore, for the most extreme worst conditions.

The conclusion that would be drawn from this is that only the low inclination launch window shoud be used. However, this window will occur at the wrong time of day during some periods of time, making the launch impractical because of lighting conditions.

Using the method of fixed arrival time previously mentioned, the velocity increment required for igniting one revolution late can be greatly reduced. It would be necessary to increase the ignition energy slightly to reduce the flight time by 1.5 hours. These energy changes imply velocity increment changes up to three meters per second.

The flight geometry reserve is summarized in Figure 1. This shows that the required velocity increment can be as low as 31 m/sec. or as high as 133 m/sec. depending upon the method chosen to correct the errors. (One meter per second is equivalent to 30 lbs. of payload).

Flight performance reserves have been calculated by considering the effect of 3σ variations in thrust levels, I_{sp}, weight of propellants, dry weights, winds and 3σ variations in the density of the atmosphere. These variations are summarized in Figure 2 which shows the effect on payload of the individual variation as well as the RSS Δ payload.

The purpose of the adaptive guidance scheme is to insure achievement of accurate orbit while utilizing fuel in the best possible way in the presence of disturbing factors such as winds, density variations, off nominal powerplant performance, changes in weight of structure and propellant, and even one engine out during either the first or second stage. Mathematically the optimum trajectory for each set of initial conditions is obtained by making use of the calculus of variations to determine the steering angles. However, the computational effort required by the calculus of variations exceeds that available on board the Saturn vehicle. Thus it becomes necessary to develop methods which approximate the results as defined by the calculus of variations but can be obtained by substantially simpler mathematical procedures suitable for programming in the on-board computer. There are presently at Marshall two conceptually different schemes of guidance which approximate the results obtained by the calculus of variations. Both of them are examples of adaptive guidance schemes which nearly optimize performance under a widely varying set of conditions.

In the minimax guidance system, the desired steering angle χ is produced in the form of a polynomial in the state variables, position, velocity, and acceleration as they are measured by the inertial sensors on the guidance platform. The polynomial is a curve fit of a family of disturbed trajectories around the nominal case including the most severe cases for which basic performance is still sufficient to complete the mission.

This, of course, is a series expansion of the trajectory in term of the state variables. Exact representation of the trajectories obtained from the calculus of variations would call for an infinite series. However, it is possible to design a finite series with a reasonable number of times which adequately represents the desired trajectories.

An example of such a finite representation of the steering function is presented on Figure 3.

This polynomial contains 18 terms and these have been adequate for Saturn I vehicle missions. The form of the polynomial was selected after a careful consideration of coordinate systems, the behavior of the partial derivatives of χ with respect to r, V, Θ, $\frac{F}{m}$, and a careful selection of the data points to be included in the curve fit.

The first guided Saturn flight, that of the recent SA-6 vehicle, flew with an early form of the minimax guidance scheme. The polynomial used in that flight was expressed in cartesian coordinates and had a total of 33 terms, a form which we believe to be inferior to the present concept.

In that flight, however, the guidance scheme had a rather severe test, since there were three independent adverse conditions during the first stage burn including one engine out. These adverse conditions resulted in a large velocity error, at second stage ignition.

In spite of this large perturbation, the second stage minimax guidance system guided the vehicle back to an accurate orbital injection and in doing so achieved near optimum performance, the payload loss compared to loss on the nominal trajectory being negligibly small. A comparison of the actual trajectory of the SA-6 vehicle with the pre-calculated nominal trajectory is shown in Figure 4.

Some typical results of the minimax guidance system into earth orbit are shown in Figure 5. These results were obtained with a simple velocity cutoff into earth orbit.

Figure 6 shows the results of application of the polynomial technique to steering from earth orbit to lunar transit injection. (Errors are shown at lunar arrival). Here, twelve coefficients are needed for the pitch steering function. Cut off computation is made on the basis of energy only, thus, $V^2 - \frac{2\mu}{r}$ = constant.

Trajectories for the Saturn V vehicle are more complicated than Saturn I trajectories and, therefore, a more difficult task for the guidance system. These additional complexitives are due to discontinuities in thrust during the guided portion of flight when staging occurs between the S-II stage and S-IVB stage. Another complexity arises from the desirability to have the guidance system perform without modification throughout the launch window where the launch azimuth is constantly changing. It is obviously desirable that the pitch steering equations be independent of the launch azimuth.

Present indications are that a single polynomial will be adequate for guidance during the entire 2-1/2 stages of guided flight and for all launch azimuths.

We have said that the task of a guidance scheme is to produce commands which steer the vehicle to desired terminal conditions with a minimum amount of fuel expended, even if severe perturbations are encountered. Since a closed form solution of the two-point boundary value problem for the calculus of variations does not exist for trajectories on a spheriodal earth, numerical evaluation of the calculus of variation is required. These procedures are too cumbersome for on-board calculations. We have seen how the minimax guidance mode insures accuracy and optimality based on a polynomial representation of calculus of variations steering programs.

On the other hand the iterative guidance mode is a closed form solution to a simplified calculus of variation problem. Under the assumption of a flat earth in vacuum, the optimum thrust direction for rocket flight to orbit is $\tan \chi = \frac{A + Bt}{1 + Ct}$. By elimination of any constraint on the range at which injection takes place into orbit, this expression can be further simplified to $\tan \chi = A + Bt$.

It has been demonstrated that by proper choice of the constants A and B in this expression that this representation is also very nearly optimal for a speriodal earth where guidance is active out of the sensible atmosphere. Two conditions are desired when injecting into a given orbit. These are the altitude and path angle. By integrating a simplified set of equations of motion to the end of expected burning time and using this linear steering law, one may by inverting this process solve for the values of the constants A and B which will satisfy the altitude and path angle constraints. The solution is updated at discreet time intervals during flight, hence, the name "iterative".

Since the flat earth assumptions used in this process implies a constant gravity magnitude and direction, it is easy to see that refinements must be superimposed over this simple guidance law to improve its performance when applied to the more complicated problem of steering a vehicle under realistic geophysical and environmental conditions. Since we know where we are and where we want to be, an approximation to the gravity magnitude and direction (perhaps an average value between the instantaneous and final state) can be assumed. Here it can be seen that as the steering program is updated, the accuracy improves as the end point is approached, because the assumptions become increasingly more valid.

The information which is necessary for the on-board computation of the iterative guidance mode comes partially from navigational measurements on board and partly from preset input. The vehicle knows its instantaneous position, velocity, and acceleration from on-board computations and measurements. Desired final conditions are simply input along with the nominal specific impulse of the engines.

The advantages of having such a limited amount of input can be summed up in the most dramatic characteristic of the iterative scheme: flexibility. A change in mission, say, a different orbit, implies a change in only three input values: the horizontal component of velocity at ignition, the normal component of velocity at injection, and the distance of the vehicle from the center of the earth at injection. The equations for the guidance laws remain unchanged.

If the specific impulse of an engine is uprated, this input change is necessary. If other vehicle parameters change, such as lift off weight, but the mission remains unchanged, no change of input is required. Later it will be shown how this inherent flexibility especially enhances the abort capability of the iterative guidance scheme.

Typical performance perturbations in Saturn V trajectories to orbit are shown in Figure 7, together with accuracy and payload loss data. Present plans call for the iterative scheme to be tested on Saturn flights SA-8 and SA-9.

The same iterative guidance equations used to guide a vehicle to orbit can also be used for guidance out of earth orbit to the moon. Different inputs are required into these equations, however. Injection into the earth orbit requires a fixed velocity, fixed altitude and fixed cut off path angle which are achieved at the end of the powered portion of flight.

The terminal boundary conditions of the powered flight into the lunar transit trajectory are not so straightforward. There are many ways to get to the moon while simultaneously satisfying certain conditions such as a specified time of flight, specified periselenum, and a specified inclination of the orbit with respect to the lunar equator. The problem is to find the optimum injection conditions which satisfy these conditions.

This optimization problem has been resolved by the development of a closed form set of equations which represent a "cut off" hypersurface. The hypersurface equations are used with the iterative guidance equations to obtain both the steering function and cut off conditions.

Development of the equations for the hypersurface starts with the calculation of an accurate trajectory to the moon by means of the calculus of variations and numerical integration. The conditions thus derived for position, velocity and path angle are inserted into the closed form equations for an ellipse. The resulting ellipse closely approximates the trajectory obtained from the calculus of variations but is not identical to it. One then selects a point on the ellipse in the vicinity of the moon and designates it as an aiming point.

The family of ellipses passing through this aiming point and having a specified perigee near the earth constitutes what we refer to as the cut off hypersurface. We then deliberately commit two errors which largely compensate each other. (This compensation is exact for a nominal vehicle). We guide the vehicle with a somewhat inaccurate set of equations for steering and cut off. The error thus made is then compensated by offsetting the aiming point from the desired target point. Actual trajectories fly through the desired target or terminal conditions. The geometry of the ellipse and actual trajectory is shown in Figure 8. Errors at lunar arrival calculated using the hypersurface in conjunction with the iterative guidance equations are shown in Figure 9.

To force every set of perturbations in waiting orbit to inject at points on the nominal transfer ellipse would be costly in payload; thus the nominal transfer ellipse should be rotated in space such that an optimum maneuver will be performed during the injection burn, once the desired orientation of the ellipse is determined.

The resulting error at periselenum due to the simplified hypersurface, as compared with the exact cutoff hypersurface is less than 15 km at periselenum, 15 sec. time of flight, and reflects a midcourse velocity requirement at the lunar sphere of influence of less than 3m/sec. When the iterative guidance mode is combined with the simplified hypersurface, the loss due to the iterative as compared with a fully optimized calculus of variation trajectory is less than 10 pounds.

By using the analytic equations to describe the hypersurface, terminal boundary conditions are available to the iterative scheme such that the same guidance equations can be used for going out of orbit as for going into orbit.

This feature is quite advantageous to the programming of the onboard digital computer. Full scale error analyses have been conducted over the span of azimuths from 72° to 105°, which represents a launch window of approximately five hours. There is virtually no error at lunar transit injection due to the iterative guidance equations; the small errors at the moon quoted previously are due to the cutoff hypersurface alone.

In the event of a malfunction of the launch vehicle it is still possible to accomplish various alternate missions with the Saturn V vehicle. The alternate mission selected would certainly depend upon many things, such as the time point in the lunar program, the particular failure encountered, the possibility of performing other worthwhile experiments, etc. In Figure 10 the capability of the Saturn V to inject the spacecraft into a 100 nautical mile circular orbit either with or without the use of the service module is presented. Also included is the capability of the vehicle to perform a circumlunar flight. For example, if an engine should fail on the S-IC stage at any time later than 5 seconds after liftoff it would be possible to inject the complete spacecraft into a 100 nautical mile circular orbit. From that point several possibilities would be open. The astronauts could remain in orbit until reaching a point from which they could return to a convenient recovery location on the earth; various orbital operations and experiments could be performed; or, as indicated in the last column, the S-IVB could be reignited and used in conjunction with the service module to propel the spacecraft into a circumlunar flight. In the case of failure of the S-IC prior to 5 seconds the possibility of using the service module does not help. This is due to the fact that the problem encountered is one of the dynamics of motion and not one of propellant availability. The same is true in the case of failure of two of the S-IC engines, as presented in the next line. However, it should be pointed out that the S-IC is held down on the pad until all S-IC engines have ignited and are functioning properly, so these early failures are very improbable.

As might be expected, greater capability to perform alternate missions exists during S-II flight than during S-IC flight. For example, if one of the S-II engines fails to ignite, the performance capability exists to accomplish any of the missions presented. Some problems are encountered at second plane separation with one of the S-IC engines out, however it is felt that these can be solved.

The lifetime of the S-IVB/CSM combination in a 100 n. mi. circular orbit is less than one day, so for long time operations, it is necessary to go to higher orbits. The maximum circular orbit into which a fully loaded CSM/LEM can be injected by Hohmann transfer from a 100 n. mi. circular orbit is shown in Figure 11. This Figure demonstrates that the payload capability to establish circular orbits well above the radiation belts exists even if one or two S-IC engines should fail at some time in flight. In developing the data for this chart, it was assumed that the apogee impulse was given by the S-IVB. If this should prove impossible due to the pressurization problem on the S-IVB, the apogee impulse could be delivered by the service module.

The discussions so far have considered alternate missions which are designed to obtain useful data from flights that have malfunctioned. Another possibility would be to abort the mission and return immediately to earth. First indicated, in Figure 12, is the capability of the service module to modify the earth return footprint, assuming that staging is made directly from the S-IC to the SM following S-IC burnout. This case might arise, for example, if the S-II fails to ignite. Figure 13 presents similar information for the S-II and the S-IVB with a parameter of the time of failure from liftoff of one of the S-IC engines. Noting the change in scale from the preceeding diagram, it may be seen that the S-II and S-IVB possess much more capability for this mission than the service module. Although the S-II and the S-IVB possess more velocity increment capability than the SM, the primary reason for the larger footprints is the higher thrust to weight ratio of the S-II and S-IVB.

Figure 14 indicates the footprint possible by staging from the S-II to the S-IVB if it is assumed that the S-II stage fails 154 sec. after S-II ignition. The downrange capability becomes infinite in this case, since the possibility of going into orbit exists.

Now, summarizing the preceeding diagrams with an overlay on the surface of the earth,in Figure 15, it may be seen that abort during the S-II with the S-IVB stage allows much more area coverage than is required to return to any specified recovery area within the range safety limits. Abort from S-IC burnout conditions with the S-IVB allows return to any crossrange location within the limits. Since the S-II capability exceeded that of the S-IVB, it will clearly be able to cover the entire range safety band also. Going directly to the service module could cause some trouble, however, as indicated by the relatively small area covered on the chart.

To demonstrate the capability of the iterative guidance mode to perform missions of the nature just discussed, failure of the S-II at various times in flight with staging directly to the S-IVB was considered. Assuming the mission to be continuation of the flight into a 100 n. mile parking orbit, the errors at orbital insertion were found to be negligible. A comparison of the performance of the guided cases with similar cases which were optimized with calculus of variations techniques revealed that the performance losses were also negligible.

In summary, the iterative guidance mode gives an explicit solution to a steering program, which adheres quite rigidly to an optimum steering program, and which requires a limited amount of input. The equations required for onboard computation, though more extensive than a polynomial, are easily implemented on the Saturn V computer, and there is no change in the program required for configuration changes or for mission changes.

SUMMARY OF VELOCITY INCREMENTS COMPOSING THE FLIGHT GEOMETRY RESERVE

VARIATION FROM PROFILE	VELOCITY INCREMENT (m/s)			
	FIXED FLIGHT TIME		FIXED ARRIVAL TIME	
	Low Inclination Window	High Inclination Window	Low Inclination Window	High Inclination Window
Ground Launch Window (≈ 4 hr)	0	0	0	0
Orbit Launch Window (80 sec)	10	10	10	10
Launch Azimuth Error (± .1 deg)	4	4	4	4
One Revolution Late Ignition	15	105	3	3
Distance to Moon (72 hr at Apogee)	14	14	14	14
Total	43	133	31	31

Fig. 1

THIS CHART EXCLUDED BECAUSE OF CLASSIFICATION.

Fig 2

THE MINIMAX GUIDANCE MODE

$$\chi = \sum_{0,0}^{6,0} A_{ij} F^{j} X_{i}, \quad \text{WHERE } X_0 = 1,\ X_1 = R,\ X_2 = V,\ X_3 = \theta,\ X_4 = RV,\ X_5 = R\theta,\ X_6 = V\theta$$

χ =

	I	R	V	θ	RV	$R\theta$	$V\theta$
I	A_{00}	A_{10}	A_{20}	A_{30}	A_{40}	A_{50}	A_{60}
F	A_{01}	A_{11}	A_{21}	A_{31}	A_{41}	A_{51}	A_{61}
F^2	A_{02}	A_{12}	A_{22}	A_{32}	0	0	0

$$\begin{aligned}\chi = {} & (A_{00} + A_{01}F + A_{02}F^2) + (A_{10} + A_{11}F + A_{12}F^2)R + (A_{20} + A_{21}F + A_{22}F^2)V \\ & + (A_{30} + A_{31}F + A_{32}F^2)\theta + (A_{40} + A_{41}F)RV + (A_{50} + A_{51}F)R\theta \\ & + (A_{60} + A_{61}F)V\theta\end{aligned}$$

Fig 3

Fig 4

THIS CHART EXCLUDED BECAUSE OF CLASSIFICATION.

Minimax Guidance Mode
3 1/2 Stages to Orbit with
5 sec Coast between S-II
and S-IVB

Nominal Injection Conditions
$h = 185.2$ km
$\vartheta = 0°$ (Horizontal)

PERTURBATION			ΔR (km)	$\Delta \vartheta$ (deg)	Δw (lb)
$S\text{-}II_1$	$S\text{-}II_2$	S-IVB			
STD	STD	STD	-.114	0	- 12
ΔF = -22,000 lb	STD	STD	.115	0	- 23
STD	ΔF = -18,200 lb	STD	-.253	.028	- 21
ΔF = -22,000 lb	ΔF = -18,200 lb	STD	.014	.031	- 10
STD	STD	ΔF = -8,000 lb	-.033	-.005	- 16
ΔF = -22,000 lb	STD	ΔF = -8,000 lb	.226	-.010	- 35
STD	ΔF = -18,200 lb	ΔF = -8,000 lb	-.174	.025	- 51
ΔF = -22,000 lb	ΔF = -18,200 lb	ΔF = -8,000 lb	.125	.022	- 54
STD	Δw = +2,500 lb	Δw = +1,000 lb	-.169	.016	
ΔF = -22,000 lb	Δw = +2,500 lb	Δw = +1,000 lb	.078	.014	
STD	Δw = -2,500 lb	Δw = -1,000 lb	-.045	-.019	
ΔF = -22,000 lb	Δw = -2,500 lb	Δw = -1,000 lb	.162	-.016	

Fig 5

MINIMAX ERRORS AT LUNAR ARRIVAL

A_z = 72°, 2nd ORBIT

NOMINAL: RCA = 1,946 km

PERTURBATION	ΔRCA (km)	Δt (sec)
NOMINAL	0	0
ΔF = +8,000 lb	38	30
ΔF = −8,000 lb	-50	-61
ΔI_{sp} = +8.62 sec	-24	-47
ΔI_{sp} = −8.62 sec	27	50
Δw = +2,500 lb	-10	-10
Δw = −2,500 lb	10	9
ΔV = +5 m/s	13	26
ΔV = −5 m/s	-11	-26
ΔV = +15 m/s	36	75
ΔV = −15 m/s	-36	-76
Δt_{ig} = +15 sec	-26	45
Δt_{ig} = −15 sec	-20	-103
RMS	28	54

Fig 6

Fig 7

Iterative Guidance Mode
3 1/2 Stages to Orbit with
5 sec Coast between S-II
and S-IVB

Nominal Injection Conditions
R = 6,559,440.0 m
V = 7,795.3747 m/s
ϑ = 0° (Horizontal)

PERTURBATION	ΔR (km)	ΔV (m/s)	$\Delta \vartheta$ (deg)	Δw (lb)
NOMINAL	-.003	.007	0	-34
ΔI_{sp} = +5.05 sec (S-II_1)	-.003	.014	.0006	-85
ΔI_{sp} = -5.05 sec (S-II_2)	-.003	.005	.0003	-75
Δw = +2,500 lb (S-II_1)	-.003	.006	.0010	-36
ΔF = +18,200 lb (S-II_2)	-.003	.006	0	-54
ΔF = -18,200 lb (S-II_2)	-.003	.008	.0005	-57
ΔI_{sp} = +5.14 sec (S-II_2)	-.003	.005	.0005	-44
ΔI_{sp} = -5.14 sec (S-II_2)	-.003	.005	.0005	-47
ΔF = +8,000 lb (S-IVB)	-.004	.005	.0005	-34
ΔF = -8,000 lb (S-IVB)	-.003	.005	-.0019	-51
ΔI_{sp} = +8.62 sec (S-IVB)	-.003	.012	-.0003	-33
ΔI_{sp} = -8.62 sec (S-IVB)	-.003	.002	-.0002	-33
Δw = +2,000 lb (S-IVB)	-.004	.005	-.0011	-40
Δw = -2,000 lb (S-IVB)	-.003	.006	-.0002	-26
RMS	-.003	.008	.0008	-51

SCHEMATIC DIAGRAM OF PERTURBED EARTH-MOON TRAJECTORIES

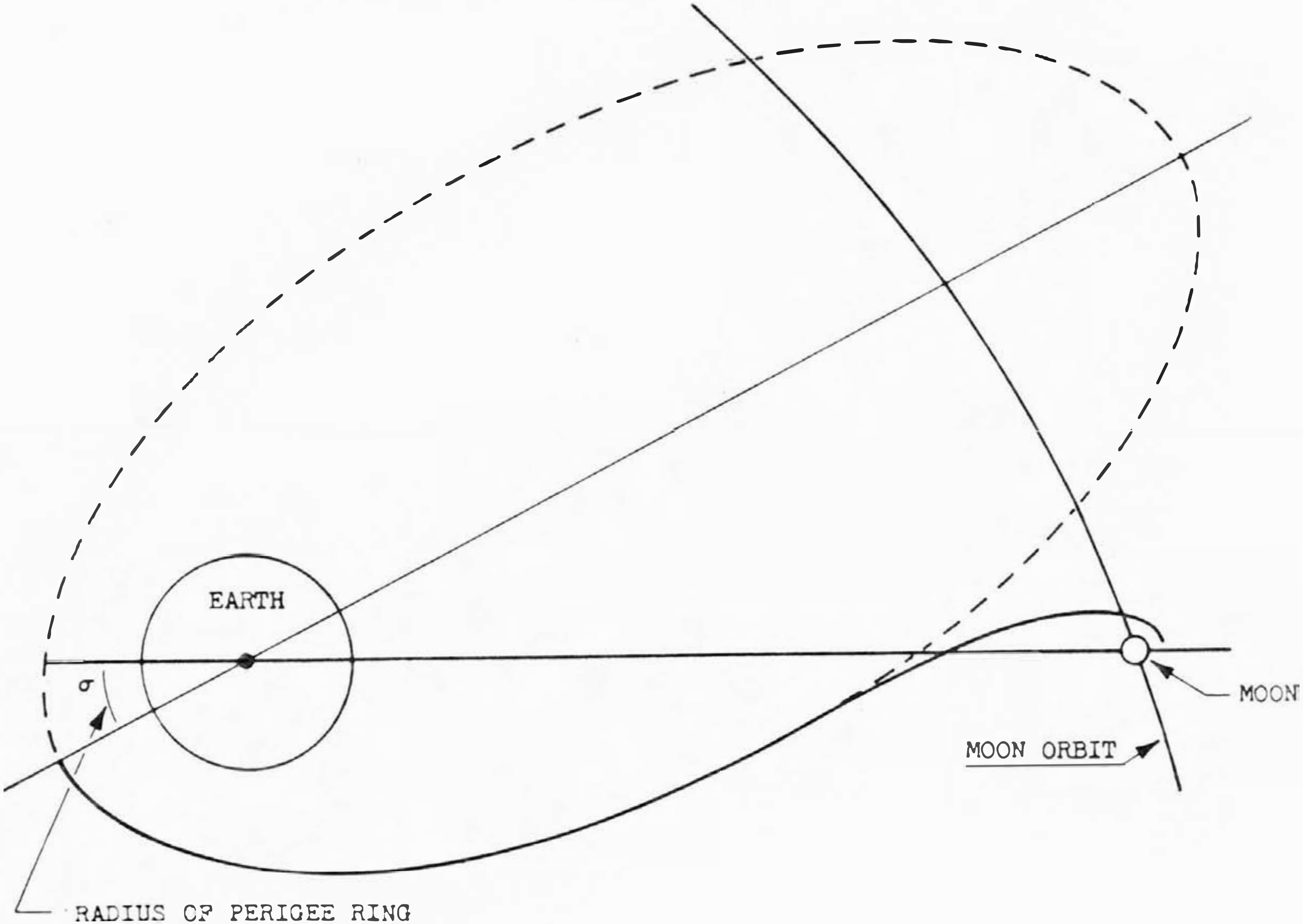


Fig 8

ERRORS AT LUNAR ARRIVAL

$A_z = 72°$, 2nd ORBIT

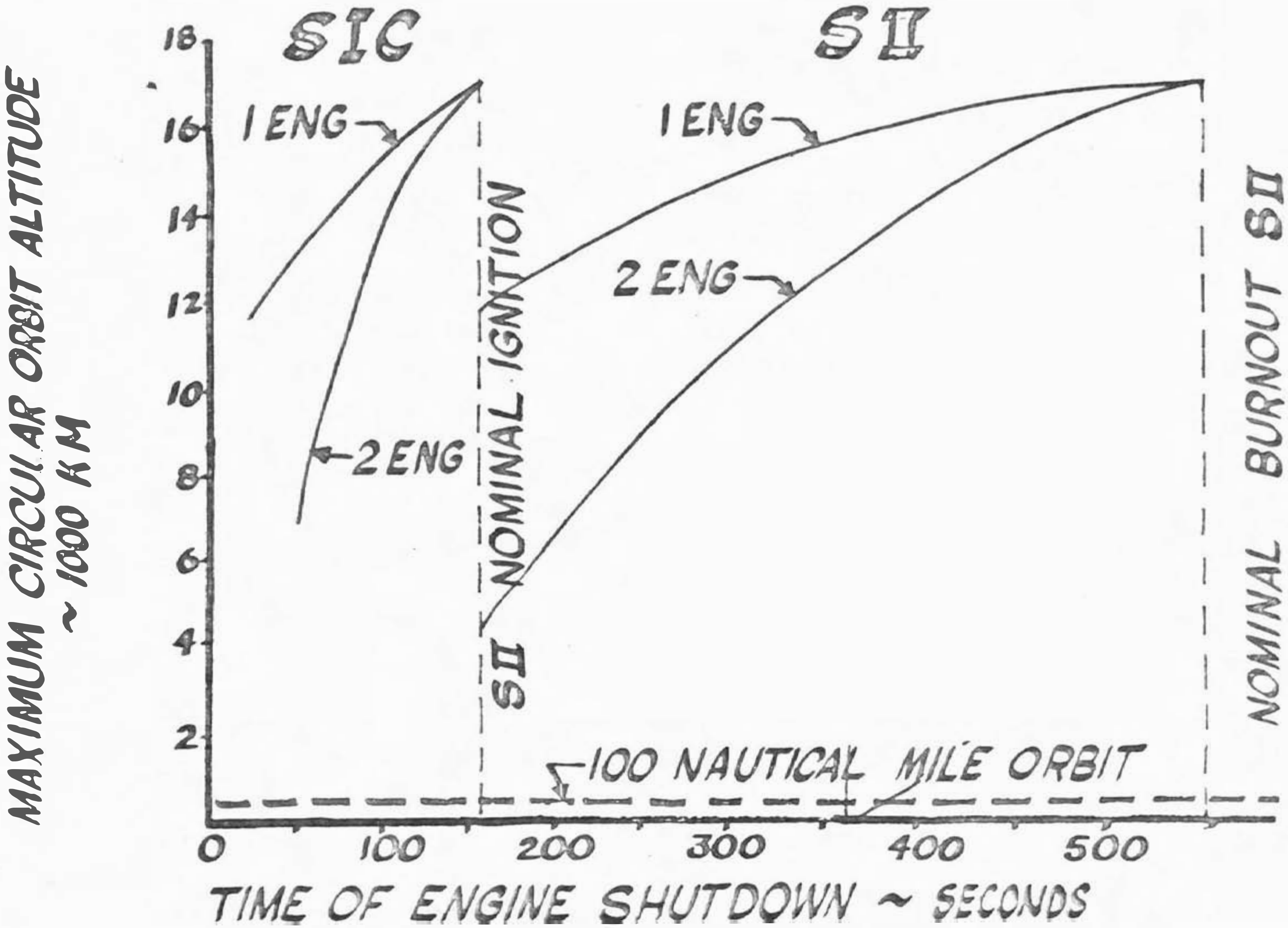
NOMINAL: RCA = 1,918.9 km; Inc = 1.101 deg; t = 20 Oct '66, 9 h 45' 8.6"

PERTURBATION	Δ RCA (km)	Δ Inc (deg)	Δt (sec)
NOMINAL	0	0	0
$\Delta F = +8000$ lb	- 4.8	.002	-17.2
$\Delta F = -8000$ lb	-24.7	.012	9.3
$\Delta I_{sp} = +8.62$ sec	-11.9	.003	9.8
$\Delta I_{sp} = -8.62$ sec	9.8	0	-10.1
$\Delta w = +2500$ lb	- 5.7	.002	4.9
$\Delta w = -2500$ lb	5.4	0	-4.8
$\Delta z = +10$ km	-11.8	.936	-6.9
$\Delta z = -10$ km	14.6	.623	5.4
$\Delta \dot{z} = +10$ m/s	- 2.7	.150	-3.0
$\Delta \dot{z} = -10$ m/s	4.7	.021	0.2
$\Delta R = +10$ km	2.8	.013	-2.4
$\Delta R = -10$ km	.8	.012	2.4
$\Delta V = +5$ m/s	1.5	.012	-2.1
$\Delta V = -5$ m/s	0.5	.013	-1.0
$\Delta R = +30$ km	16.4	.010	-11.1
$\Delta R = -30$ km	- 14.0	.019	7.3
$\Delta V = +15$ m/s	2.2	.012	-1.4
$\Delta V = -15$ m/s	- 0.7	.014	0.4
RMS	10.0	.268	7.1

Fig 9

FAILURE DESCRIPTION	COMPLETE SPACECRAFT CAPABILITY IN ORBIT FOLLOWING NORMAL MISSION SEQUENCE	SPACECRAFT CAPABILITY IN ORBIT USING THE SERVICE MODULE	SPACECRAFT CAPABILITY FOR CIRCUMLUNAR FLIGHT USING THE SERVICE MODULE
	EARLIEST ACCEPTABLE FAIL TIME AFTER LIFT OFF	EARLIEST ACCEPTABLE FAIL TIME AFTER LIFT OFF	EARLIEST ACCEPTABLE FAIL TIME AFTER LIFT OFF
S-IC ONE ENGINE SHUTDOWN	5	5	5
S-IC TWO ENGINES SHUTDOWN	29	29	29
S-II ONE ENGINE SHUTDOWN	154.4	154.4	154.4
S-II TWO ENGINES SHUTDOWN	208	208	208
S-IVB IGNITION FAILURE	MISSION CAN NOT BE COMPLETED	543.5	MISSION CAN NOT BE COMPLETED

Fig 10



ORBITAL ABORT MODE CAPABILITY

Fig 11

ABORT FROM S1C STAGE BURNOUT

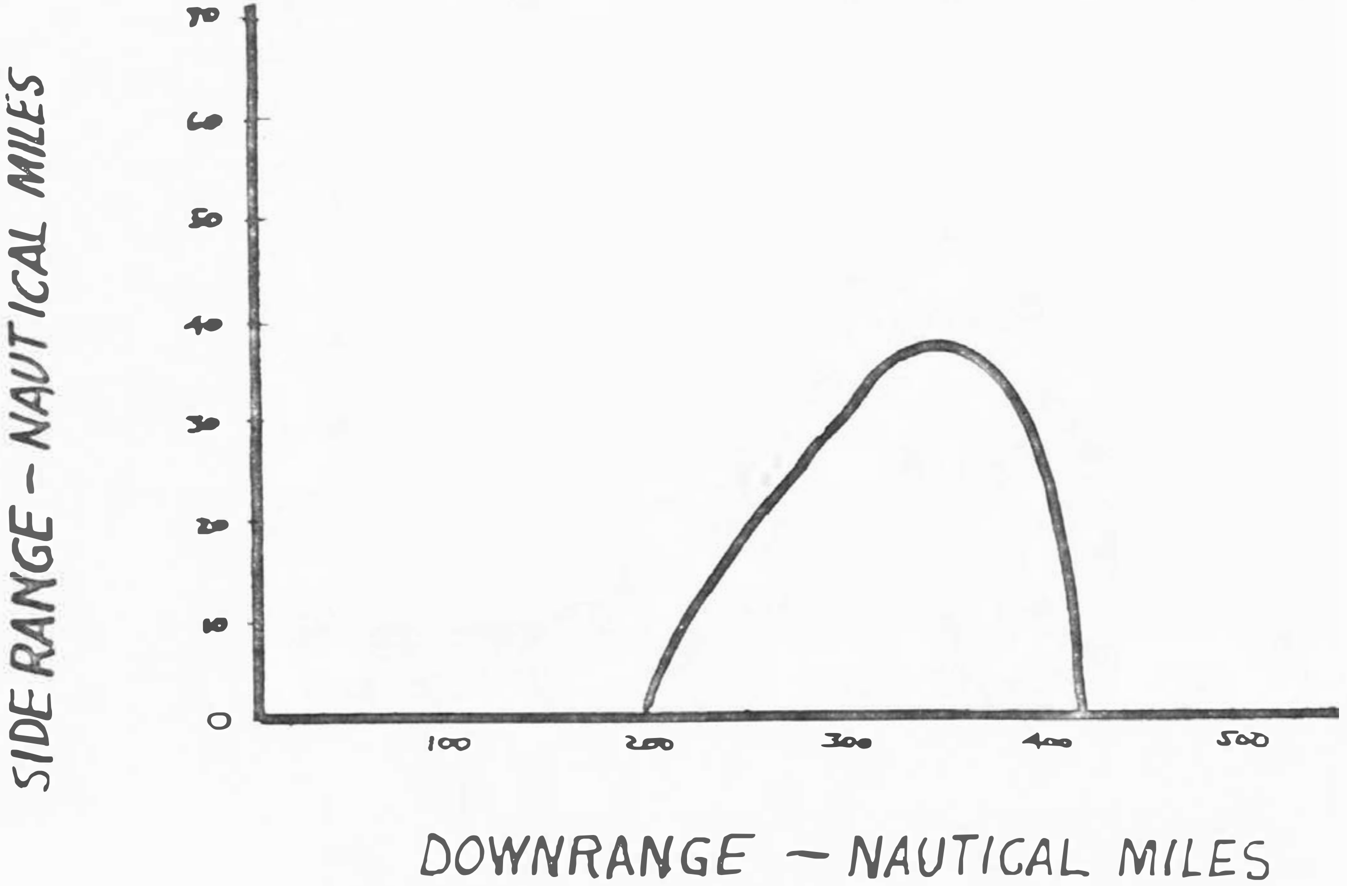
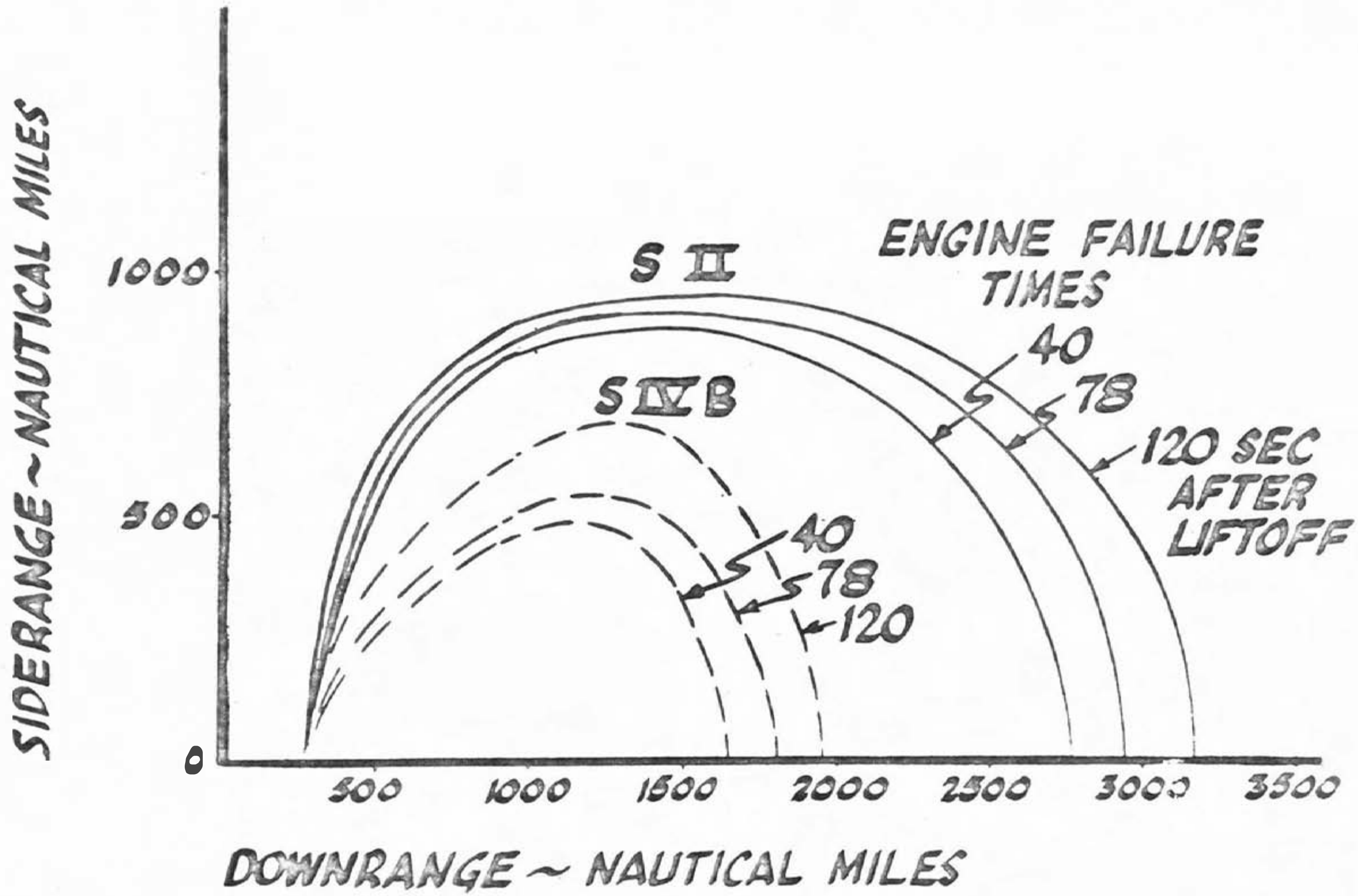


Fig 12

SERVICE MODULE FOOTPRINT

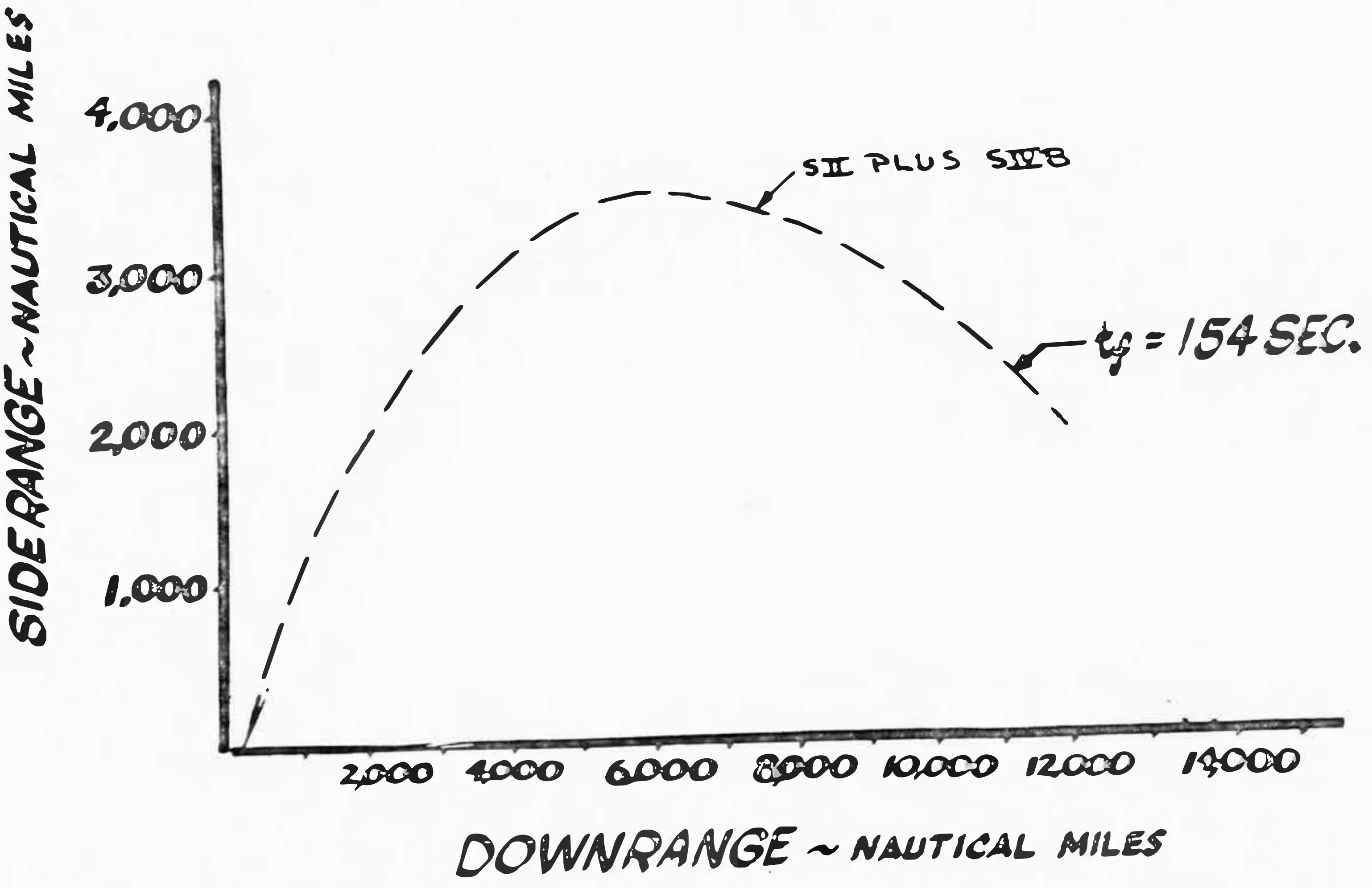
ABORT FROM S1C STAGE BURNOUT



EARTH DIRECT MODE FOOTPRINTS

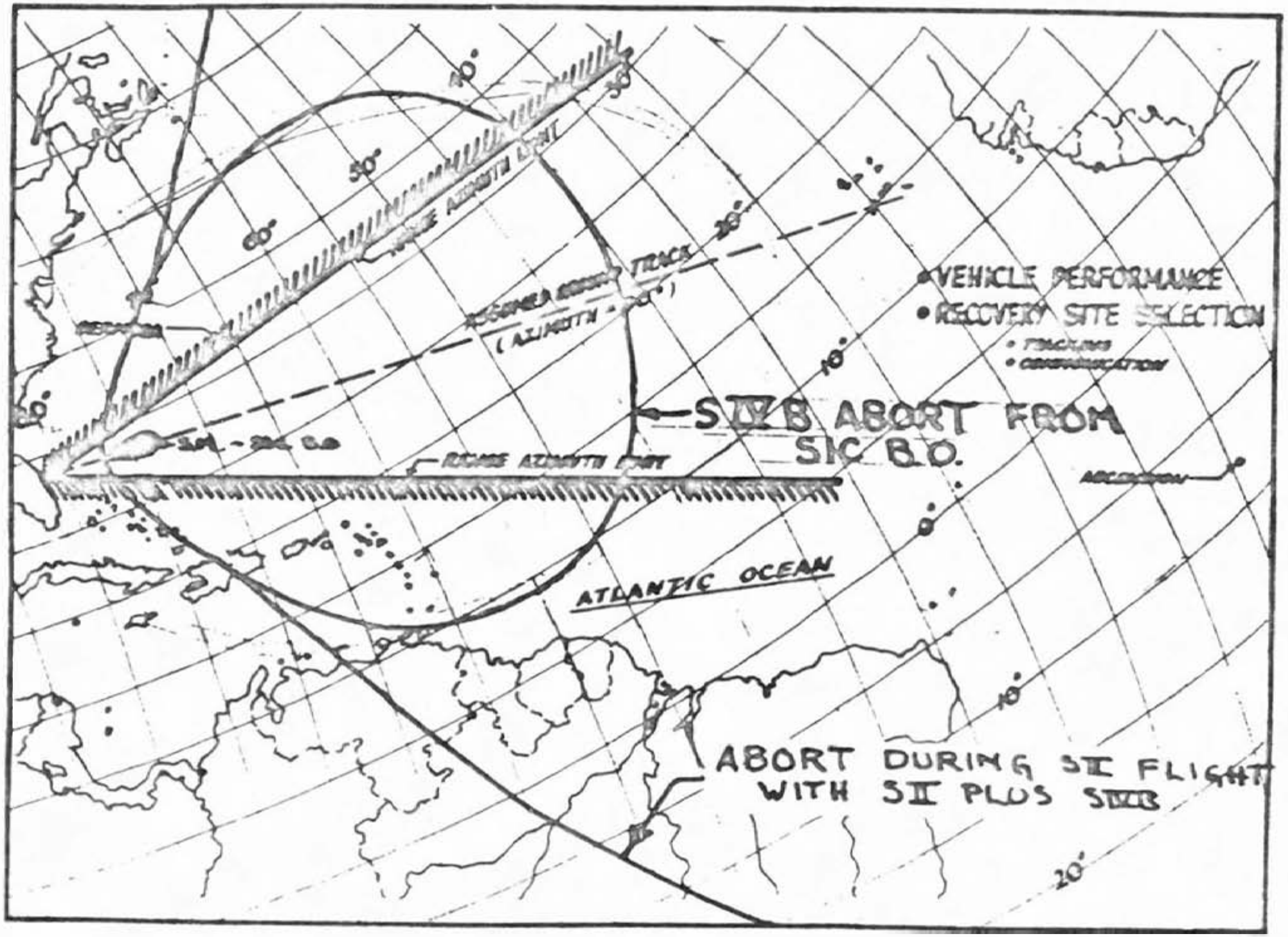
Fig 13

ABORT DURING SII BURNING



EARTH DIRECT MODE FOOTPRINTS

Fig 14



DIRECT MODE RANGE REQUIREMENTS

Fig 15

GUIDANCE, CONTROL AND INSTRUMENTATION

By Mr. Ludie Richard

GUIDANCE, CONTROL, AND INSTRUMENTATION

by Mr. Ludie Richard

Before discussing the specialized functions of the electronic equipment in the Saturn vehicles, it is necessary to examine the overall concept on which the layout of this equipment is based. Figure 1 shows the layout of the astrionics systems in the Saturn vehicles wherein we have defined the astrionic system as the overall integrated system made up of the guidance and control, electrical, measuring and telemetry, and other electronic subsystems.

The diagram in Figure 1 shows that the launch vehicle consists of a number of propulsion stages stacked together with an Instrument Unit section to complete the desired overall configuration. Each one of the propulsion stages contains its own power supplies, measuring and telemetry systems, actuation mechanisms for thrust vector control, Emergency Detection System (EDS) sensors and the sequencing and commanding circuitry to operate the engines that provide the stages propulsion and to operate other subsystems within the stage itself. The Instrument Unit contains the centralized equipment required to perform and initiate functions associated with guidance and control, vehicle sequencing , inflight testing, and receipt of externally-generated commands. The Instrument Unit houses its own power and distribution system and also has a separate measuring and telemetry system. Contained within the Instrument Unit are tracking aids for the vehicle in the form of various transponders and the radar altimeter. (It will be noted that, where possible, tracking beacons have been placed in the lower stages where their functions are primarily associated with the operation of that stage.)

The EDS also receives information from various components within the Instrument Unit. In the EDS distributor the critical launch vehicle conditions are summed and the resulting information is forwarded to the spacecraft to aid in or cause an abort decision. The Instrument Unit also accepts commands from the spacecraft which will allow it to be steered or positioned by the spacecraft system during a portion of the flight. The Instrument Unit receives certain mode commands which are generated by the spacecraft system and command the Instrument Unit into the various modes of operation. Launch vehicle summary information is sent to the spacecraft from the Instrument Unit for display purposes.

It will now be possible to identify the function of the various components within the overall astrionic system as they pertain to guidance and control, vehicle sequencing, inflight tests, etc. The guidance system of Saturn is all-inertial and derives its information from an Inertial Measuring Unit. This stabilized platform system provides measurements of the vehicle attitude and the vehicle motion in three orthogonal directions. It is designed to be a three or four gimbal system depending on the requirement of the flight and the necessary freedom of motion involved. The information from the stabilized platform passes to the data adapter, which is essentially the input-output equipment for the digital computer. In the data adapter the information is converted or summed and passed to the digital computer where it is operated on and necessary steering commands derived. (The computations involved in deriving these commands have been discussed previously by Dr. Geissler.) This operation results in a command from the computer to the data adapter which is then conditioned and transmitted to the vehicle control system.

The control system accepts information from the stabilized platform and computer system and combines it with the vehicle turning rate information that it receives from the rate gyros which are located in either the first stage or the Instrument Unit depending on the flight phase. (Body bending requires the use of rate sensors in the first stage during its operation.) Control accelerometers may also be required to provide information to aid in lessening the aerodynamic loading on the vehicle during its flight through the atmosphere. After the control computer has filtered and summed these various inputs, it sends signals to the actuators of the operating stage.

Figure 2 shows the operation of the actuation system which controls the space vehicle attitude. As the illustration shows, the first and second stages are provided with five engines wherein four of them are gimballed to change the direction of the composite thrust vector. Four hydraulically-driven actuators are used to produce yaw motions and four actuators produce pitch motions. A command to all eight actuators produces a roll torque. Each actuator employs a hydraulic system which is driven by the particular engine associated with that actuator. In the case of the third stage the single engine is driven by two actuators, one producing yaw motion and the other producing pitch motions. To produce roll during the thrusting period, the auxiliary propulsion system of the S-IVB is utilized.

The auxiliary propulsion system is provided on the orbital stage to generate the necessary control torques to stabilize the vehicle while it is in free-fall. The actuation arrangement is shown on the drawing and consists of jets located at two positions on the periphery of the vehicle. These jet nozzles are so arranged as to provide roll, pitch, and yaw torques on command from the control computer.

Figure 3 demonstrates some of the problems associated with development of the control system for a vehicle the size of the Saturn. During the time the first stage is operating, some of the characteristics of the vehicle can be seen from the graph. The bending modes of the vehicle which have been previously discussed are shown as well as the sloshing frequencies of the liquids in the tanks of the vehicle. The control system receives exitation from these sources and must distinguish the overall movement of the vehicle from these local effects. The engine reaction zero frequency is representative of the condition known as "Tail Wags Dog." This frequency represents the sinusoidal response where the inertial effects of the accelerated engine induce forces at the gimbal equal and opposite to the side force obtained by swiveling the thrust vector. The net effect is that the vehicle response to the engine reaction at this frequency is zero. The thrust vector control frequency represents a local effect in the actuation system caused by compliance problems in the structure that the actuator presses against. The pitch-yaw control frequency is the designed characteristic frequency of the control servo system. It is selected to be a reasonable distance from the first bending mode and yet high enough in frequency to provide adequate vehicle response and to minimize rigid body structural loads and dispersions due to thrust vector misalignments.

A perfect filter arrangement to eliminate all of these effects on the control system is impossible to generate so the techniques utilized have been to phase-shift the effects of the first and second bending modes and attenuate all of the higher frequencies by filtering and shaping the system inputs. The sloshing problem is reduced by providing anti-slosh baffles in the propellant tanks. The interplay between the physical characteristics of the vehicle and the electrical characteristics of the overall control system is a continuing process in the Saturn program, which proceeds right up to the time prior to final shipment when all of the characteristics are known.

The sequencing of the on-off functions of the various stages is controlled by the digital computer and the data adapter working through a special device called the switch selector. A switch selector is located in each stage and in the Instrument Unit and can be addressed independently by the digital computer-data adapter combination. Through this system any switching function can be commanded at anytime by the digital computer-data adapter combination.

In the measuring and telemetry system, measurements are taken during all phases of vehicle flight. Appropriate transducers are provided for measuring such parameters as temperature, strain, and vibration. Measurement of signals in the inertial platform, digital computer, and other pertinent equipment is made to monitor vehicle operations.

Each measurement, after proper signal conditioning, is transmitted to ground stations via a telemetry link for analysis in real time or storage for subsequent analysis.

During orbital operation, an onboard analysis of the S-IVB/IU is performed prior to injection into lunar trajectory utilizing data from the measurement system. The measurement and telemetry system of the S-IVB is interconnected with the measurement and telemetry system of the Instrument Unit. These systems are then connected to the data adapter and digital computer in a manner that allows the computer to assess critical data from systems of the S-IVB and Instrument Unit combination to make an onboard "Go-No Go" decision. The system also carries an optional recorder to record data when telemetry reception is not possible and then to transmit the data when transmission is possible. One period of such recorder operation is during the separation sequences involved in staging. The ordnance equipment causes a telemetry blackout and detailed information on the separation functions cannot be transmitted in real time and therefore the recorder records this operation and transmits it after staging has been completed.

Another very important function of the measuring and telemetry system on Saturn is that function associated with prelaunch checkout of the overall vehicle. The system has been configured in such a manner that it provides vehicle data to the prelaunch checkout system in a form directly utilized by the automatic prelaunch equipment. This configuration of the measuring telemetry equipment is called Digital Data Acquisition System (DDAS) and is common to all stages of Saturn. The Remote Automatic Calibration System (RACS) contained in the measuring and telemetry system provides "hi-lo" stimulation for calibration of the various inputs to the system.

The command receiver located in the Instrument Unit is utilized with ground stations to provide updating in the guidance system based on tracking data and to assist the operations personnel in the checkout and assessment of the launch vehicle system while it is in orbit. Commands through this system can be used to sequence the vehicle or establish a new mode of operation, if required. The system provides a "secure" transmission capability and can transmit data to the onboard computer on a routine or priority basis.

The purpose of the tracking systems is (1) to determine vehicle trajectory for mission control, (2) for postflight evaluation of the vehicle performance, and (3) for range safety. Several tracking systems are used to determine the trajectory during powered ascent and orbital flight. Consolidation of tracking data from the several tracking systems provides the best possible trajectory information. Not all tracking transponders shown may be included on every mission. Due to the long burning time of the multi-stage vehicle, the powered flight cannot be "seen" completely from land-based tracking stations. The use of tracking ships is necessary to track insertion into orbit.

The C-Band Radar Transponder used as a tracking aid, increases the range and accuracy of C-Band Radar ground stations equipped with AN/FPS-16 and AN/FPQ-6 radar system. C-Band Radar stations at Cape Kennedy, along the Atlantic Missile Range, and at many other locations around the world, provide global tracking capability.

The MISTRAM System (Missile Trajectory Measuring System) is the latest tracking system installed at the Atlantic Missile Range. The MISTRAM ground stations are able to track the powered flight of the Saturn from launch to approximately cutoff of the S-II stage. The MISTRAM system provides data in real time.

The Offset Frequency Doppler System (ODOP) is a frequency modification of the UHF doppler system. It is a multi-station doppler tracking system with one station complex located at Cape Kennedy. Tracking from ODOP stations is limited to the first portion of powered flight (S-IC Stage). ODOP tracking data is provided immediately following vehicle lift-off. Other tracking systems cannot "see" the vehicle immediately following lift-off or their accuracy is limited by multipath propagation.

The ODOP tracking system determines vehicle position by using doppler frequency measurements. The signal, transmitted from the transponder, is received simultaneously at several ground stations located around Cape Kennedy. A minimum of three receiving stations is necessary to determine position; additional ground stations are employed to receive redundant data.

AZUSA is an interferometer tracking system with tracking stations located at Cape Kennedy and on Grand Bahama Island. These two tracking stations cover only a part of the vehicle flight path. The AZUSA system provides tracking data in real time.

The altimeter transmits RF pulses and measures the pulse travel time from the vehicle to the ocean surface and return. The measured time interval is digitally encoded and the data is telemetered to a ground station, or recorded on tape, for playback as the vehicle passes over a ground station.

Now that the overall systems layout has been discussed, it is possible to go back and re-examine the resulting systems from the standpoint of mission flexibility and overall operational reliability.

As can be seen from the way structure and mission affect the astrionics system, and due to the way developments occur, flexibility with short reaction time is a must. The layout just described results in the fact that items which change from mission to mission and vehicle to vehicle are all located in the Instrument Unit. The control computer is effectively programmed by plug-in module changes to handle the problems associated with changing stage and structural characteristics. The digital computer stored program is modified to meet the varying requirements placed on it by changes in guidance objectives and vehicle sequence operations. The programs of the digital computer can be modified to make commands from the ground take on different meanings as well as those from the spacecraft. In effect, the overall Instrument Unit is a mission-oriented segment that determines the operation of the launch vehicle on any particular flight. Modification to the stages in the name of changing missions have effectively been eliminated and with them the large management, cost, and interface problems which would normally result in a vehicle of this complexity.

In the area of overall system reliability a real advantage is gained in having concentrated the intelligence functions that would have normally been duplicated up and down the vehicle in the Instrument Unit. By having single items fulfill similar functions for all stages, a resulting part count decrease is obtained, which, in turn, improves expected performance. However, the main advantage in this concentration of high part-count components afford an even greater possibility which is to make these items redundant in one form or another to better meet the reliability requirements associated with this program. The previous discussions on the mission sequence have shown the launch vehicle will be called upon to operate unattended, both during prelaunch and flight modes, for long periods of time, and therefore normal design practices would not have yielded a system with a sufficiently high probability of mission success. Therefore, redundancy in various forms has been applied to the critical components or subsystems to better ensure mission fulfillment. Some of the forms of redundancy employed in the astrionics system are shown in Figure 4. The forms of triple modular redundancy shown apply to such items as the digital computer, data adapter, and the control system. The majority vote system shown in the upper left-hand corner of the illustration operates on the principle that each voter examines the output of each module and sends forward to the next module the majority condition. For instance, a failure in module A_1 would not be apparent to module A_2 since the voter would have sent the majority information forward, which would have been a good signal from B_1 and C_1. Likewise, a failed voter may have caused a bad signal to go forward, but this too would have been corrected by the next voting level. This majority vote-type redundancy is utilized mostly in the digital circuitry of the computer and data adapter. The other form of triple modular redundancy is normally utilized with analog or proportional-type signals wherein a voter is used to observe the outputs of two modules, in this case, module B and module C. Module C output is being used as the output of the overall configuration. If B or C should fail, the voter would switch the system to cause the output of A to be utilized. Again, this system requires two well coordinated failures to cause an overall failure in operation.

The quadruple or series parallel arrangement shown in lower left-hand corner is utilized where other forms of redundancy are not possible. This type of redundancy is employed in some of the power electronics and in the valving associated with operating the auxiliary propulsion system.

Another inherent reliability factor in Saturn is the characteristic of the multi-engine booster. Shown symbolically are the four control engines on either the first or second stage. A failure of one of the actuators or both represents a torque on the vehicle that can be sensed and immediately compensated for by a slight motion of the other three engines. It is an inherent characteristic that such a system can survive the loss of position control on a single engine. By isolating the electronics associated with each actuation system, it has been possible to allow electronics components failures associated with driving the actuator to be compensated for by the same effect.

The problem associated with the switch selector subsystem is twofold. Not only must its inherent reliability be high since it occurs four times, but transmissions to it must not be in error, in case of open or short circuit failures in the hundreds of feet of wire up and down the vehicle. Redundancy has improved the unit itself, and to overcome the transmission error, every command to the switch selector from the data adapter is returned on another bus to be verified before it is executed. In the case of a lack of comparison, a complimentary command is transmitted which takes into account the transmission failure and results in the same desired action out of the switch selector.

A potential "backup" for the inertial platform during the latter stages of the flight is provided by switching the spacecraft inertial measuring system into the launch vehicle control system. The engineering problems associated with using a primary sensor system located on the very top of a large elastic booster are severe and have yet to be solved, but this approach does offer additional backup possibility. Switching criteria and an exact hardware solution which does not overcomplicate or add excessive weight to the spacecraft are two problems yet to be completely solved.

The EDS system has been made fully redundant, not only to better ensure its reliability in acting when it should, but also to avoid its causing the loss of a good vehicle.

Other possibilities of single mode failures still remains in the overall system and some will be too costly or add too much weight to eliminate. A good example of this is the actuation system on the S-IVB stage. It is driven by a single hydraulic system powered by the engine. There are single actuators in pitch and yaw which can fail and curtail the mission. And yet, duplication of this hardware would lead to severe payload penalties.

As in any vehicle, there can be no substitute for adequate and complete ground testing and quality assurance. The designs to cause performance improvement mentioned previously only apply to making good items better. Redundancy offers very little improvement in areas where simplex reliability is low.

The entire design is throughly environmentally qualified as components and as systems, where size doesn't become prohibitive.

Another problem area is the one that comes with any system made as flexible as this overall system. Flexibility is a must to handle changing configurations and missions, but it brings with it a capability to make more mistakes. To test and control vehicle mission design and programming, extensive testing and progressively increasing simulation is done before every flight.

For example, all of the parameters affecting the control system design are verified on full scale flight articles and then reproduced in extensive ground simulation set-ups. As flight hardware, such as computers and sensors, becomes available, it is added to the simulator set-up and replaces the simulation which previously had been there for the flight article. The final simulation consists of the actual hardware system, operating with its total flight program through a total mission, wherein it must demonstrate its total characteristics and capability, both statically and dynamically in a real time, closed-loop form.

This rigorous ground testing is difficult and time-consuming but allows a degree of confidence that leads to an eventual need of less flight articles to reach the final level of confidence in the overall system.

SATURN IB/V ASTRIONICS SYSTEM

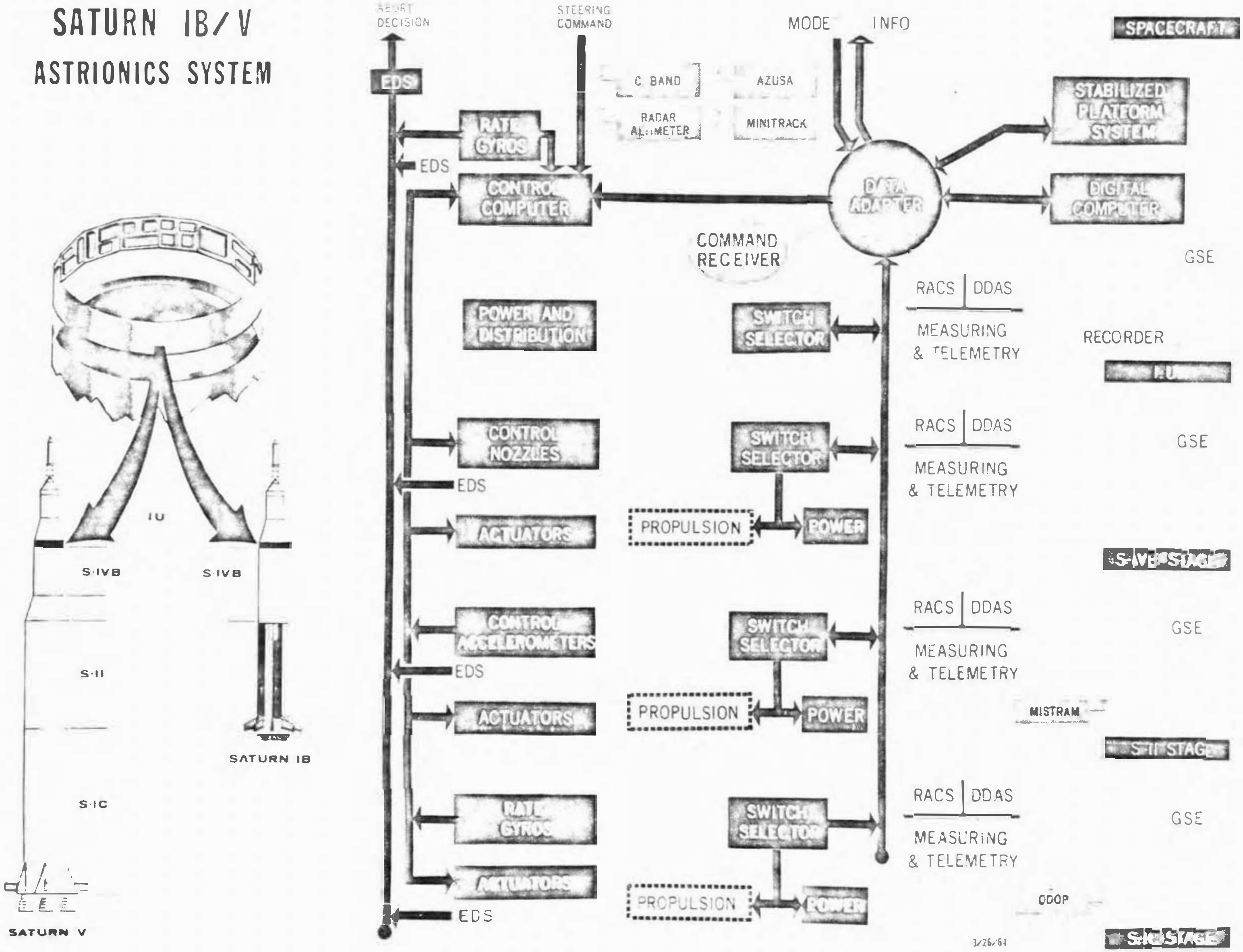
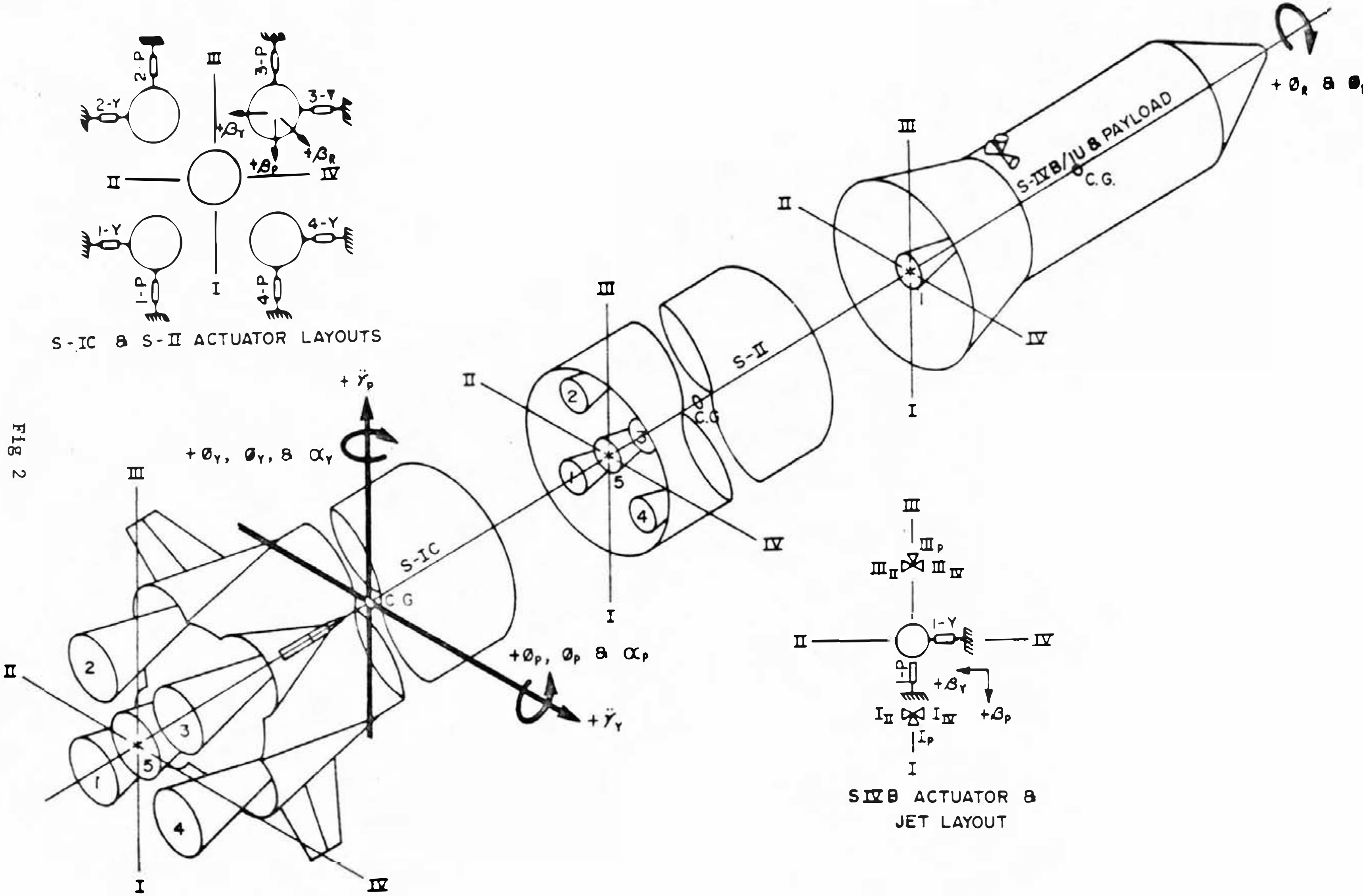


Fig 1



S-IC & S-II ACTUATOR LAYOUTS
2-P
2-Y
3-P
3-Y
1-Y
1-P
4-Y
4-P
$+\beta_Y$
$+\beta_P$
$+\beta_R$
S-IVB/IU & PAYLOAD
C.G.
$+\emptyset_R$ & $\dot{\emptyset}_R$
S-II
C.G
S-IC
C.G
$+\ddot{\gamma}_P$
$+\emptyset_Y$, $\dot{\emptyset}_Y$, & α_Y
$+\emptyset_P$, $\dot{\emptyset}_P$ & α_P
$+\ddot{\gamma}_Y$
SIVB ACTUATOR & JET LAYOUT
1-Y
1-P
$+\beta_Y$
$+\beta_P$
III_P
III_{II}
III_{IV}
I_{II}
I_{IV}
I_P

Fig 2

S-IC STAGE - SLOSHING & BENDING MODE FREQUENCIES

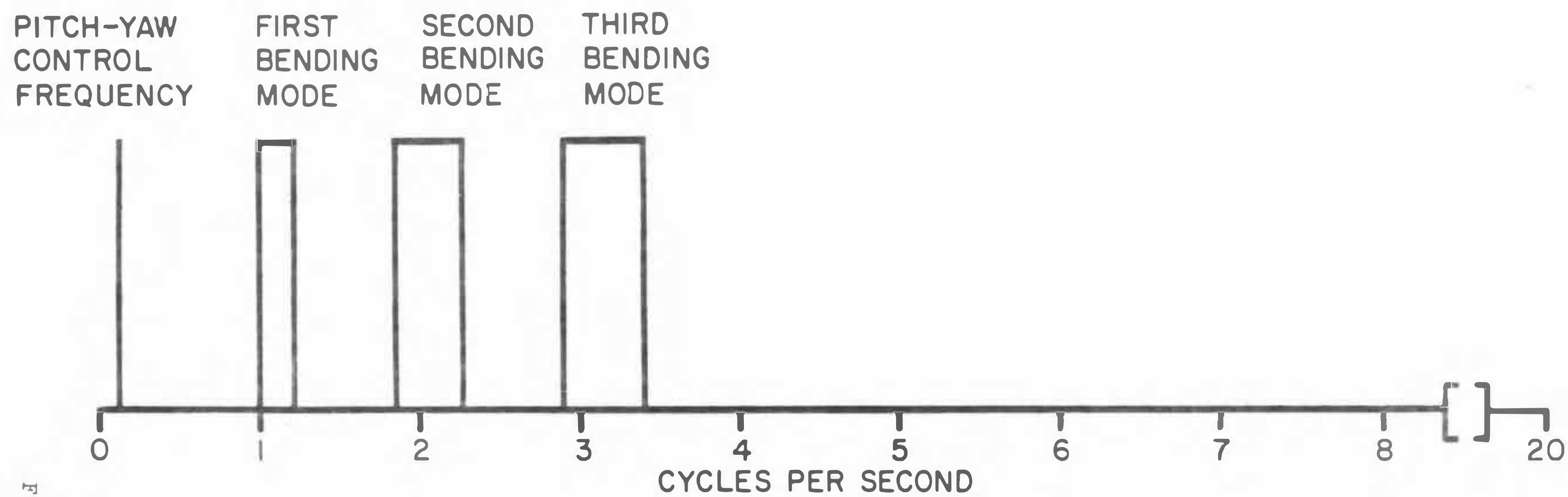
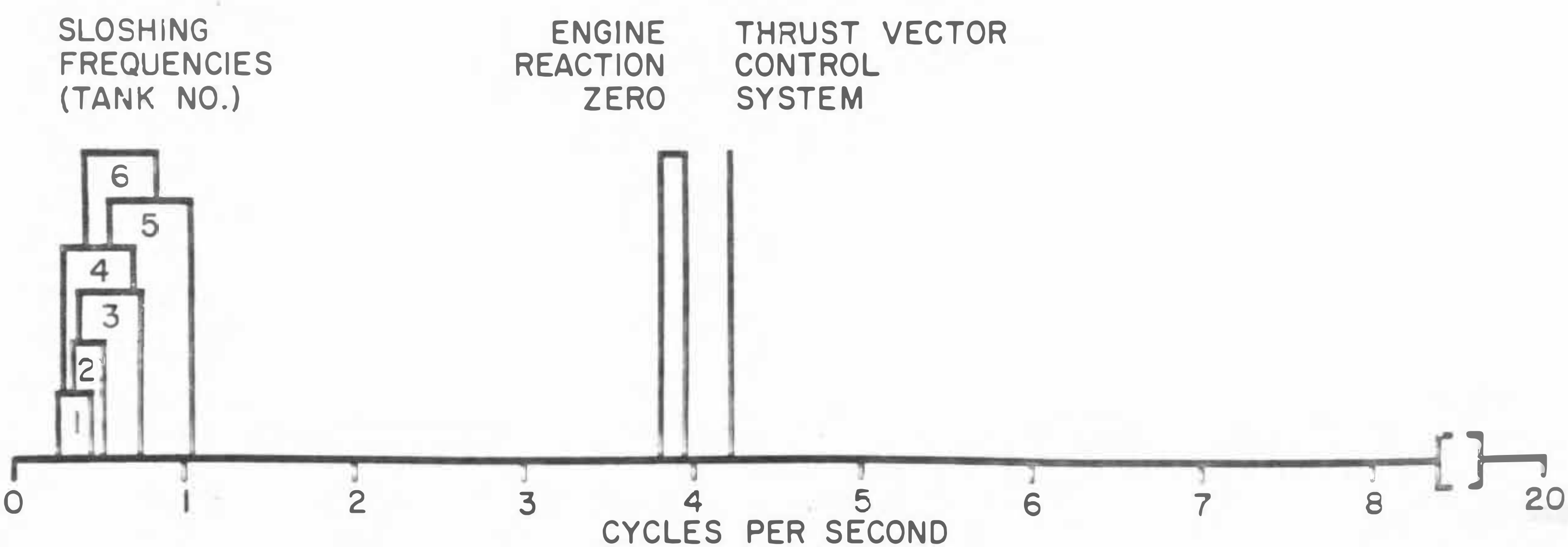
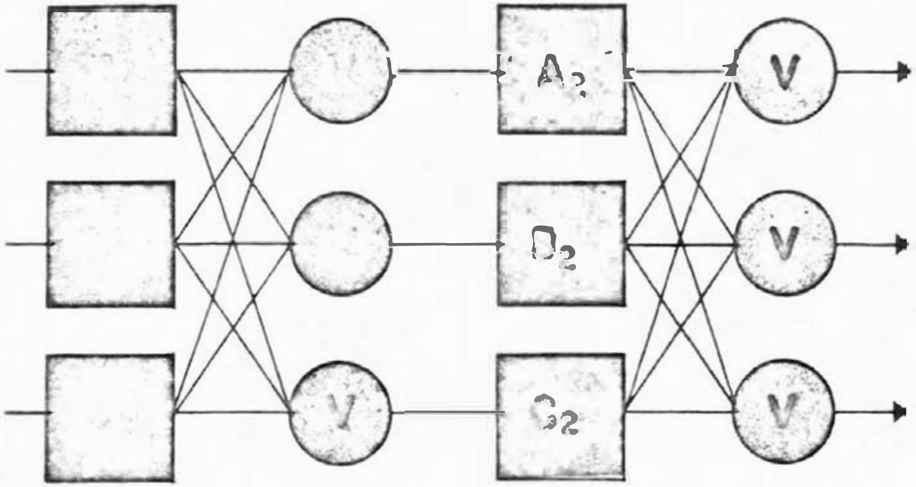


Fig 3



EXAMPLES OF REDUNDANCY

TRIPLE MODULAR REDUNDANCY



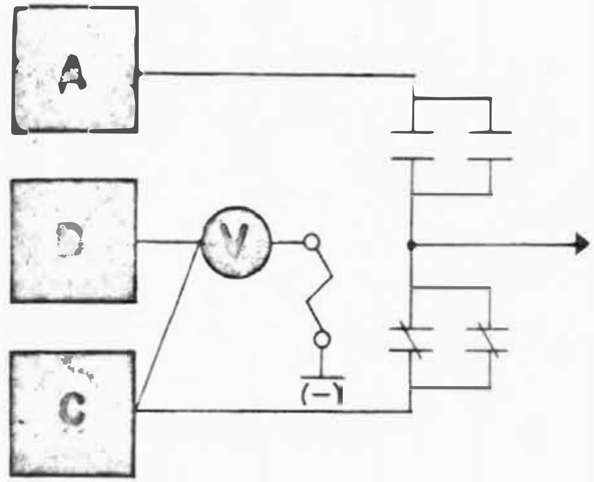
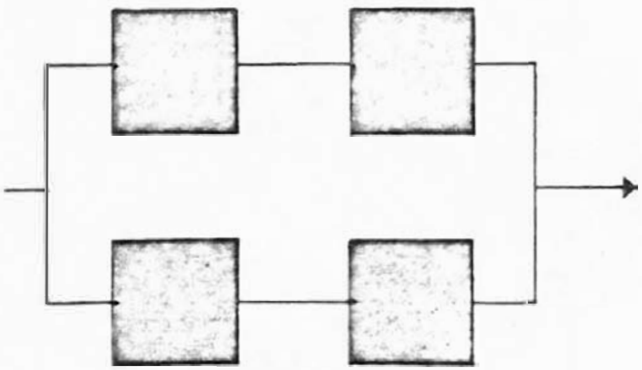
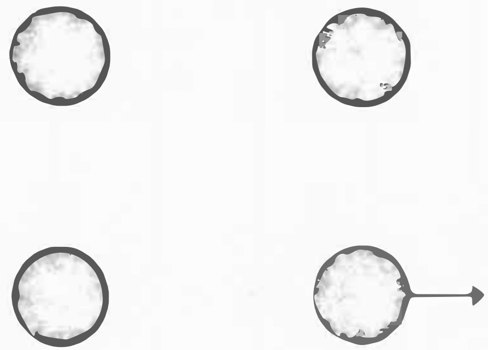


Fig 4

QUAD



"ENGINE CONTROL OUT"



S-IVB STAGE

By Mr. Roy E. Godfrey

S-IVB STAGE

By Mr. Roy E. Godfrey

As discussed earlier, the S-IVB is the third and final propulsive stage of the Saturn V launch vehicle and will supply the thrust necessary to inject the spacecraft into a translunar trajectory. The S-IVB stage is the end product of a sophisticated development program. It is a minimum risk program designed to provide high reliability for manned space transportation after very few development/demonstration flights.

The following presentation will trace the S-IVB Stage History - As a typical stage of the Saturn V Launch Vehicle - from the initial requirement through design definition, development, fabrication, testing, and launch operations, Figures 1 and 2.

The present S-IVB Stage configuration is based on the Saturn V mission requirement. Briefly stated, the Saturn V mission requirement calls for a launch vehicle capable of putting a spacecraft in a translunar trajectory which will permit a soft landing of a manned payload on the lunar surface and return.

From this overall launch vehicle mission requirement the specific stage requirements are developed. As the mission profile is expanded in detail, the stage configuration matures by joint MSFC - CONTRACTOR studies of mission requirements and stage design requirements, Figures 3, 4 and 5.

The basic function of the S-IVB Stage is to provide the last velocity increment to boost the Apollo payload into its translunar trajectory. This is done in two stages. First, the S-IVB injects the Apollo spacecraft into earth orbit, and at the proper moment, restarts and injects it into the lunar trajectory. The payload consists of the three man command module, the service module, and the lunar excursion module, which will be fully described by others at this meeting.

Approximately 80,000 pounds of propellants are expended during the first burn into earth orbit. An auxiliary propulsion system provides roll control during powered flight and full attitude control during coast. At the termination of the first-burn, the S-IVB and the

Apollo payload go into a 100-mile earth orbit for up to 4-1/2 hours. This provides three full orbits with which to pick the proper launch window for the lunar transfer orbit. During orbit the continuous vent system, previously described, is operated to provide the small amount of thrust necessary for ullage orientation and to maintain proper pressure in the propellant tank. Additional ullage thrust is provided for the chilldown cycle of the J-2 engine prior to restart. The second burn of the S-IVB injects the Apollo payload into the translunar trajectory by expending the remaining 150,000 pounds of propellant. After S-IVB burnout and verification that lunar transfer conditions have been achieved, the nose to nose rendezvous maneuver between the command module and the lunar excursion module (LEM) is performed. The lunar excursion module remains attached to the S-IVB stage which provides the attitude control for the LEM during this maneuver. After the maneuver is complete, the spent S-IVB is backed away from the LEM, and its mission is complete.

The stage which evolved from the mission requirements is 22 feet (260") in diameter, some 60 feet long, carries approximately 230,000 pounds of usable propellants and is powered by a 200,000-pound thrust Rocketdyne J-2 engine. The propellants are liquid oxygen and liquid hydrogen burned at a mass mixture ratio of 5 to 1. It is of all-welded construction with a common bulkhead separating the oxydizer from the fuel. The end bulkheads are hemispherical, the common bulkhead is a segment of a sphere welded to the lower bulkhead to form a lenticular shaped liquid oxygen tank, Figure 6.

After the basic configuration and design requirements are defined, the design and development effort follows. In this area will fall the engineering effort necessary to define the various stage systems and specific hardware and to perform the test and evaluation of this detail design.

During the system studies phase, the basic design requirements are refined and expanded upon to determine the various sub-systems which will be required, and we arrive at a point where all major subsystems are identified.

Concurrent with and following the system studies is the effort of component and system development. Here many factors must be weighed and compared and the optimum approach determined.

This operation of transforming the mission requirements to basic design requirements to the actual sub-systems and components required is a crucial period in a stage program for here much time, money, and performance can be gained or lost later in the program. In trying to present this effort in a logical manner it may appear that the process is a clean "step, check, and proceed to the next" step type of operation. This is not necessarily the case though, because many factors which guide the program, including mission requirements, do change based on design problems, material restrictions, technological advances and new experimental test data. Thus, the management channels of communication and control between all the government developing agencies and the associated contractors must be responsive to these deviations from a previously established approach. To assist in this close communication and control, MSFC utilizes technical area working groups and panels with membership composed of MSFC, other NASA Centers, and contractor personnel as appropriate. These groups have regular and called meetings to review in detail any problem areas, determine corrective action, and in general, to discuss the program progress and status.

As the development of the stage continues it becomes apparent that not all problems are restricted to stage system development. The magnitude and complexity of the systems along with the related high reliability dictate equally large and complex manufacturing efforts, Figure 7.

The total effort must be reviewed and based on this, general manufacturing plans are formulated. This manufacturing planning information forms the basis for all future manufacturing efforts and includes the following:

A detailed breakdown of the anticipated manufacturing operations.

Prediction of anticipated problem areas.

Estimation of facilities and special tooling that will be required.

Developmental and experimental work are necessary to establish and prove out new and unique methods and processes. During the formulation of these plans, much information is gained from the past experience of both MSFC and DAC in determining the best approaches to take in such areas as fabrication of large diameter monocoque propellant tank structures, fabrication and installation of the cryogenic insulation systems, welding techniques and specialized materials and assembly handling techniques.

In the area of facilities and tooling, many special areas had to be considered. A special facility was created at Huntington Beach, California, for development and production of S-IVB Stage. The major parts of this facility are an engineering building, fabrication and assembly buildings, a structural test facility and an engineering development systems integration laboratory, Figure 8. Examples of special tooling set-ups required are as follows:

a. The structural elements of the S-IVB consist of a main cylindrical section, 20 feet long. This cylinder consists of seven segments joined by longitudinal welds. Each segment is machined from a 10' x 20' x 3/4" 2014T-6 aluminum alloy plate. A waffle pattern is machine milled on a vacuum chuck horizontal milling machine to provide the 9" square waffle pattern for shell stiffness. The webs in the pockets of the waffles are left at .134 inch thickness. By making the joining welds in thicker material (9/32 in.) we are able to leave them in the as-welded condition. The flat segments are then formed to their cylindrical contour on a Verson brake with plastic cookies inserted in the waffle tops to prevent deformation. Figures 9 and 10 show the panel in the brake forming machine and after forming.

b. Another area requiring special tooling and set-up is that of the propellant tank end domes. The manufacture of the spherical end domes is accomplished by forming nine orange peel segments on the stretch press. These orange peel segments are joined in the dome welder and the entire fixture rotates under a stationary welding head, Figure 11. This head is automatically positioned by a servo controlled sensing element. The shielding of the weld arc and cooling of the metal adjacent to the weld is done primarily by means of high inert gas flow. The chilling process is augmented on the underside by pneumatically clamped titanium bars.

c. Many areas required methods and process development on a relatively large scale. One very significant example of this effort is in the fabrication process for the common bulkhead. The common bulkhead is a segment of a hemisphere of the same 130 inch radius as the end domes, and it consists of two separate aluminum diaphrams also welded from nine orange peel segments. This assembly welding is done on the same bulkhead welding tool. The lower bulkhead is filled with a 1-1/2" temperature resistant fiberglass honeycomb. By means of special fitting techniques similar to the bluing-in process in gun manufacturing, the top bulkhead is exactly fitted to the contour of the honeycomb. This fitting technique is accomplished as follows: With the lower bulkhead in place on the bonding fixture, approximately 1,000 gage blocks are placed on the lower bulkhead, each with a small glob of putty on top, prior to the installation of the honeycomb, Figure 12. The top bulkhead is then lowered onto these gage blocks, vacuum applied to assure a proper fit, then released and the upper bulkhead removed. The putty is thus mashed down to indicate the exact distance between these two bulkheads at each particular gage point. This provides the means for making an absolute contour map for the top surface of the honeycomb. The honeycomb, when it is applied, has the flexibility to conform to any irregularities in the lower bulkhead surface, Figure 13. The honeycomb used is approximately 1/8" thicker than it normally needs to be. This is bonded under heat and pressure in an autoclave with an epoxy filled phenolic resin. After making this first bond, the bulkhead is removed from the autoclave. The top surface of the honeycomb is spot-faced to a depth in accord with the contour map at each gage point; thus giving some 1,000 points on the top surface of the honeycomb that can be faired by machine and hand sanding to give an almost perfect fit. When this operation is complete, the top bulkhead is bonded to the fitted honeycomb surface. To complete the bulkhead, the edges of two peripheral tees are now welded together to provide a seal for the bulkhead core and to tie the top and bottom bulkheads together structurally.

In a low risk development program such as the Saturn V launch vehicle, testing is inevitably a major portion of the effort, Figure 14. In the S-IVB program, the testing effort accounts for at least 50% of the total effort in terms of manhours and physical resources required. The test effort logically falls into three major phases: 1) ground test, 2) the static firing program, and 3) the development or demonstration flight test.

During the initial stages of design, the only information available is that which we have obtained from previous programs plus our system and detail analyses. Since most analyses require assumptions, there is usually an amount of uncertainty in the analytical information. During the design and manufacturing stages of the development, research testing and development testing go on concurrently in an iterated process; that is, the research and development test results that are needed to verify the design analyses are fed back in, prior to release of final production drawings. As soon as components become available from the manufacturing operations and procurement systems, additional development and the start of qualification testing begins. As the process proceeds downstream sub-system testing and complete system testing is initiated. In parallel, the static firing program is started as soon as possible in order to give us a look at the components, subsystems, and systems as a total vehicle system. It is interesting to note that many components which pass laboratory environmental tests supposedly to the limit of the vehicle environment fail to operate satisfactorily when put into a total system on a static firing. The effectiveness of this type of dual approach is demonstrated by the successful first six flights of the MSFC S-I Stage and the successful S-IV Stage performance on the 5th and 6th flights.

The total S-IVB ground test program, Figure 15, requires approximately 935 tests of components and/or sub-systems. Each of these 935 tests actually consist of several different environmental tests. The total number of "component-environments" approach 30,000. This program encompasses research, development, formal qualification, and reliability testing. Much of the testing is multi environmental testing; that is, the components or sub-systems are subjected to a synthesized flight environment. For example, a telemetry rack may be subjected to its vibration spectrum while in the expected temperature environment of either heat or cold. All through the testing, the equipment would be electrically operated and radiating to a test receiving station. A second example would be the cold helium storage spheres. These helium spheres are mounted internally in the hydrogen tank and thus would be tested in the vibration environment while submerged in liquid hydrogen at a temperature of -423°F.

A second portion of the ground test program is the dynamics testing which will be carried out by Marshall Space Flight Center at Huntsville. An entire Saturn V launch vehicle will be assembled in a special tower provided for this purpose and tested to determine primary and secondary bending modes and other dynamic characteristics. Similar tests will be conducted with the first stage removed to simulate dynamic conditions after first stage separation and finally a set of dynamic tests will be run with only the S-IVB stage and the Apollo spacecraft mounted above. For these tests Douglas will supply a complete flight weight structure with all installations and components in place, though in some cases they may be dynamic simulations rather than live hardware. Very closely related to the dynamics testing will be the extensive static loads testing of the structural test stage by the DAC. Information gathered during this testing effort will be utilized in conjunction with data from other tests on vehicle mechanical sections to assure that the stage meets the structural design requirements.

Another less dramatic but equally important portion is the facility checkout test that will be conducted on Launch Complex 39 at the Atlantic Missile Range. For this, Douglas will supply another set of flight type tankage with all equipment necessary to interface with the launch ground support equipment and facilities. It will not have a live engine installed. With this and similar stages from the other Saturn V launch contractors, complete launch facility launch vehicle mating will be run, including propellant loading tests.

The first portion of the static firing program is what is termed a "battleship" program, Figure 16. It takes its name from the fact that it uses heavy weight tanks made of stainless steel plate geometrically similar to the flight tankage. This has two major advantages. It provides us with a complete propulsion system test that is not keyed to the schedule of the flight weight structure and lets us start propulsion system testing some nine months to one year earlier than would otherwise be possible. Secondly, this steel tankage has a much greater strength factor and since some of the stage components will not have been qualified prior to the time of the first firing, this provides us with extra margin of safety. The battleship program presently envisioned runs for some six months, utilizing two engines and will require some 4,000 seconds total firing time. The battleship stage will be primarily controlled by manual ground support equipment similar to that used in the S-IV program. Some automatic checkout items will be phased-in during the battleship program in order to gain operational experience prior to their use with flight stages.

The second half of the static firing program is what we call the "all systems" program, because it is executed with flight-type hardware. The start date of this portion of the program trails the battleship program by nine months. The vehicle used will be identical to a flight vehicle with the exception that it will be much more heavily instrumented. Throughout the program the vehicle will be controlled and checked out by the fully automatic computerized ground support equipment. About thirty-five separate static firings are envisioned. Some of them are full duration (approximately 8 minutes) designed to test every element of the S-IVB system and explore the limits of the flight environment. All static firings are made at the Douglas Static Test Center in Sacramento, California.

Many milestones must be achieved before the Saturn V launch vehicle undertakes its ultimate mission of propelling the first lunar landing party on its way, Figure 17. One of the more dramatic of these milestones will be the flight test program. In this area the S-IVB stage has and will continue to benefit from the previous flight tests of the Saturn I program. Of prime importance has been the opportunity to observe and analyze the performance of the S-IV stage, which formed the foundation upon which the S-IVB detailed design was built. The S-IV stage performance to date indicates a well founded design concept as well as proving out specific design approaches.

The additional Saturn I flights with the S-IV stage and the first flight tests of the S-IVB stage atop the Saturn IB vehicle, scheduled for late in 1965, will provide additional data for refinements to the S-IVB stage and associated support systems required for the Saturn V Flight Program. The final phase of the S-IVB Flight Test Program will be initiated by the first unmanned flights of the Saturn V launch vehicle which will follow by approximately one year the first Saturn IB flights. The first manned flights have not yet been identified and will depend to a large extent on how well and successfully the preliminary parts of the test program are carried out.

The area of logistics, like all other areas, is magnified by the magnitude and complexity of the program. One portion of logistics which typifies this is the physical transportation of the assembled stage, Figure 18. This illustration shows the total Saturn transportation picture, and you can see that the S-IVB Stage transportation encompasses the West Coast, Central America and the East Coast. Not shown are the modes of movement involved. Included are the most modern forms of transportation except air and rail, and air is not completely ruled out although at present it appears that the cost of a specially developed aircraft may be prohibitive. Also, to reduce damage and system degradation during the approximately five weeks of transit, special handling tools are developed and provisions made for a protective environment for the entire stage. Another item to be considered from the logistics area is the supply of propellants and the establishment of maintenance procedures and spare parts requirements. Because of the tremendous amounts of propellants and gases used at widely scattered locations, considerable planning must be done to assure that the demand can be supplied as needed. Although the S-IVB maintenance and spare parts programs are not of great magnitude compared to a military missile system, they can be quite important to a complicated and time critical launch vehicle countdown.

To provide the detail information and assistance necessary to the successful performance of the Apollo mission, support in form of both personnel and equipment is supplied by MSFC and DAC, Figure 19. This support actually begins long before the scheduled arrival of the stage at the launch site. In the earlier stages it takes the form of detail coordination and planning to assure that programs are defined which are compatible with both the stage and launch vehicle system operations and the over-all mission operations. As the program progresses, the support equipment takes on more significance and especially so in the case of the Saturn V Program because of the introduction of the computerized automated checkout concept. This concept has been introduced because with the tremendously complicated three-stage Saturn V launch vehicle, an equally complicated spacecraft, and the breadth and depth of checkout required of each element in order to assure the desired confidence for launch, nothing short of completely automatic checkout would be acceptable. The automatic checkout equipment for the S-IVB stage will be used by the DAC for system checkout and testing at their manufacturing location and static test site and similar checkout programs will be used at the KSC launch site to provide program continuity and the best possible assurance to the checkout operations.

To assist in the over-all launch operations DAC and MSFC personnel familiar with the detail operations of the S-IVB stage system and its operational interfaces, will be assigned to the launch area and work with the launch and mission operations personnel during pre-launch, launch, and orbital operations.

Based on the mission operations, the impression could be taken for the S-IVB stage that "it is all over" after completion of the flight mission. However, this could not be more wrong, especially for the first demonstration flight tests. At this point data collected during the flight must be analyzed both for feedback of results into the S-IVB Program and for its possible importance to the scientific and technical community.

The analysis of flight results, Figure 20, becomes of major importance in the case of apparent or obvious malfunctions. Causes must be isolated and defined and evaluations made to determine possible modifications and improvements. In this respect the S-IVB stage telemetry system is designed to provide the channel capacity, frequency response, and accuracy necessary for the instrumentation of the stage during its development flights and for operations use during the lunar mission. The stage contractor and MSFC work closely in the analysis of this data, with the contractor making a detailed review and analysis of the data peculiar to the S-IVB stage and its operation. Results of the analysis and any appropriate recommendation are then passed on to MSFC for inclusion in the overall vehicle evaluation.

S-IVB STAGE DEVELOPMENT SEQUENCE

INTRODUCTION

- S-IVB Stage is third and final stage of the Saturn V Launch Vehicle
- Mission - To inject the Apollo Spacecraft into orbit and then into a translunar trajectory
- Stage is the end product of a minimum risk development program designed to produce high reliability space transportation after very few development flights
- Development Sequence
  - Initial Requirement
  - Design Definition
  - Development
  - Fabrication
  - Testing
  - Launching

Fig 1

PROJECT HISTORY AND SCOPE

- S-IVB Stage configuration is based on the Saturn V LOR mission requirements

- The structural design elements, hydrogen technology and manufacturing techniques were directly adapted and scaled from the S-IV Stage.

- S-IVB Contract History

  - Saturn V Stage Design Study Contract - April 1962
  - Definitive Contract for Saturn V Flight Stages - August 1962
  - Saturn IB Stage Design Study Contract - September 1962
  - Definitive Contract for Saturn IB Flight Stages - March 1963
  - Conversion to incentive type contract estimated by - August 1965
  - Saturn V Contract Value (6 stages) $141.1 Million
  - Estimated Runout Costs (10 stages) $670.3 Million

Fig 2

PHILOSOPHY OF PROGRAM MANAGEMENT

- Concurrency is required in the S-IVB Project to permit the stage configuration to mature and respond to changes as the mission profile expands in detail and test results indicate required changes:

  - EXAMPLE - Refinement in upper altitude wind shear criteria increases vehicle bending loads.

    IMPACT - Structure must be reinforced, affecting structural test activities, and first flight vehicle hardware in production

  - EXAMPLE - Engine pump suction head requirements must be increased due to problems encountered in engine testing

    IMPACT - Tank pressures must be increased in first flight stages with a 600 lb payload penalty

Fig. 3

GOVERNMENT - CONTRACTOR INTERFACES

- Direct technical contact is maintained in early phases by working groups and panels representing all major technical disciplines and providing the necessary inter-disciplinary membership.

- As the project proceeds and the design requirements and contract end items can be adequately defined, a configuration baseline is established which is controlled and monitored by a configuration change board utilizing formal procedures.

- All direction to the contractor is provided by the stage project office acting through the contracting officer and a resident office in the contractors plant.

Fig. 4

TRANSFORMING MISSION REQUIREMENTS INTO STAGE DEVELOPMENT PROGRAM

- STAGE DESIGN CRITERIA AND DEVELOPMENT TEST PROGRAM BASED ON

  - Launch Vehicle interface requirements stage to stage - Stage to Launch Facility - Stage to Ground Support Equipment

  - Environments induced during launch vehicle powered flight of first and second stages

  - Stage mission requirements during first engine burn, orbital coast and maneuvers, engine restart and injection into translunar trajectory and translunar coast maneuvers

  - Launch crew and astronaut safety during launch preparation and flight mission

  - Pre-flight and orbital checkout capability

  - Servicing and maintenance

  - Cost effectiveness

Fig. 5

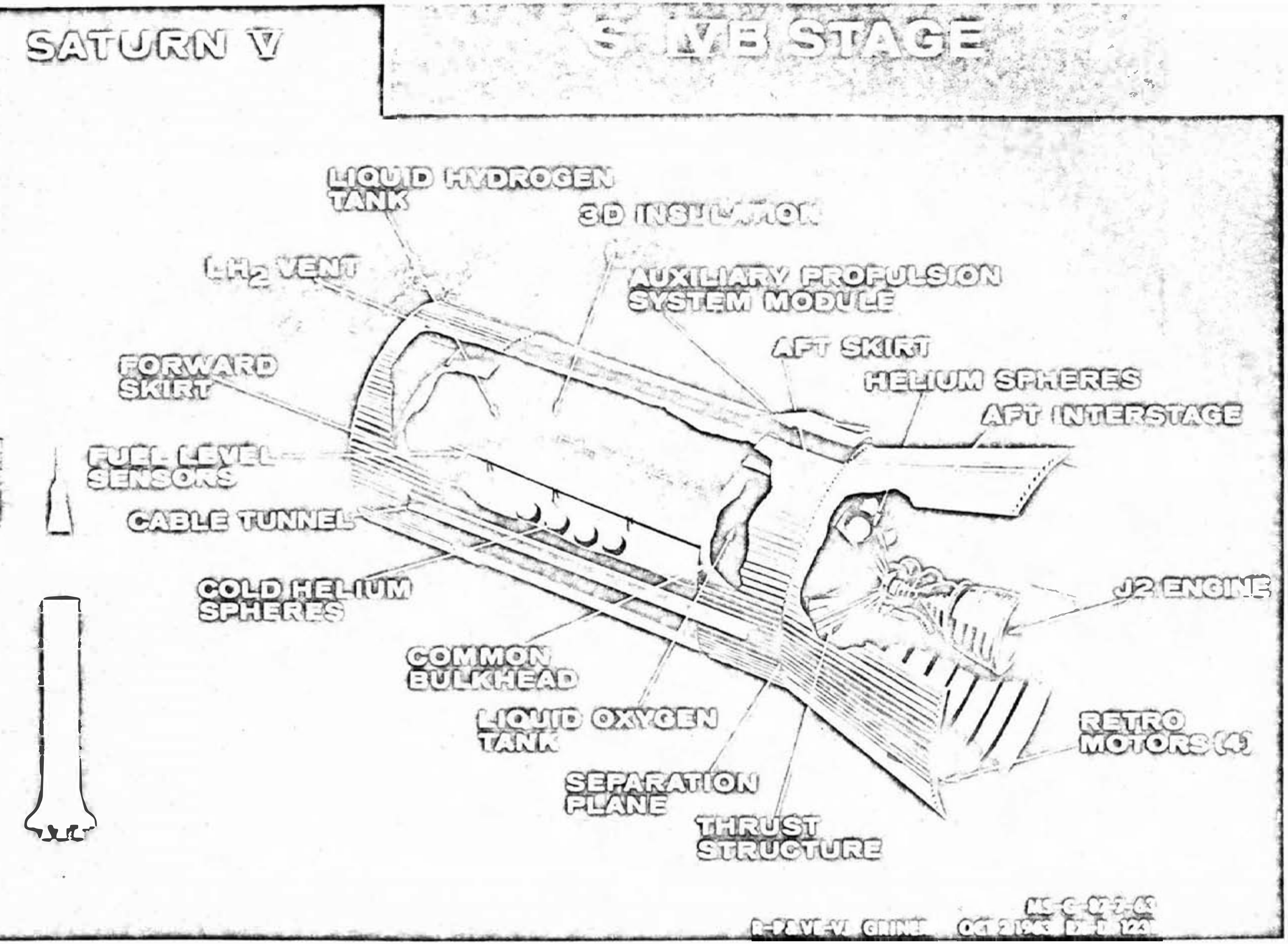

SATURN V
S-IVB STAGE
LIQUID HYDROGEN TANK
3D INSULATION
LH2 VENT
AUXILIARY PROPULSION SYSTEM MODULE
AFT SKIRT
HELIUM SPHERES
AFT INTERSTAGE
FORWARD SKIRT
FUEL LEVEL SENSORS
CABLE TUNNEL
COLD HELIUM SPHERES
J2 ENGINE
COMMON BULKHEAD
LIQUID OXYGEN TANK
RETRO MOTORS (4)
SEPARATION PLANE
THRUST STRUCTURE
OCT 2 1963

Fig. 6

MANUFACTURING

- BASIC TECHNIQUES OF PROVIDING THE S-IVB STAGE HAVE BEEN UNDER DEVELOPMENT SINCE THE S-IV PROJECT WAS INITIATED IN 1960.
- STRUCTURAL REQUIREMENTS ARE FOR HIGH DESIGN EFFICIENCY (over 0.9 mass ratio) AND ADEQUATE DESIGN SAFETY MARGIN (1.4 over ultimate) FOR A MAN RATED VEHICLE.
- TO MEET THE DESIGN GOALS, WITHIN REASONABLE COSTS, MANUFACTURING PLANNING EFFORTS MADE FULL USE OF EXISTING TOOLING AND OPTIMIZED THE PRODUCTION TECHNIQUES AROUND A TWENTY VEHICLE PROGRAM AT A MAX. RATE OF ONE PER MONTH.

Fig. 7

Fig. 8

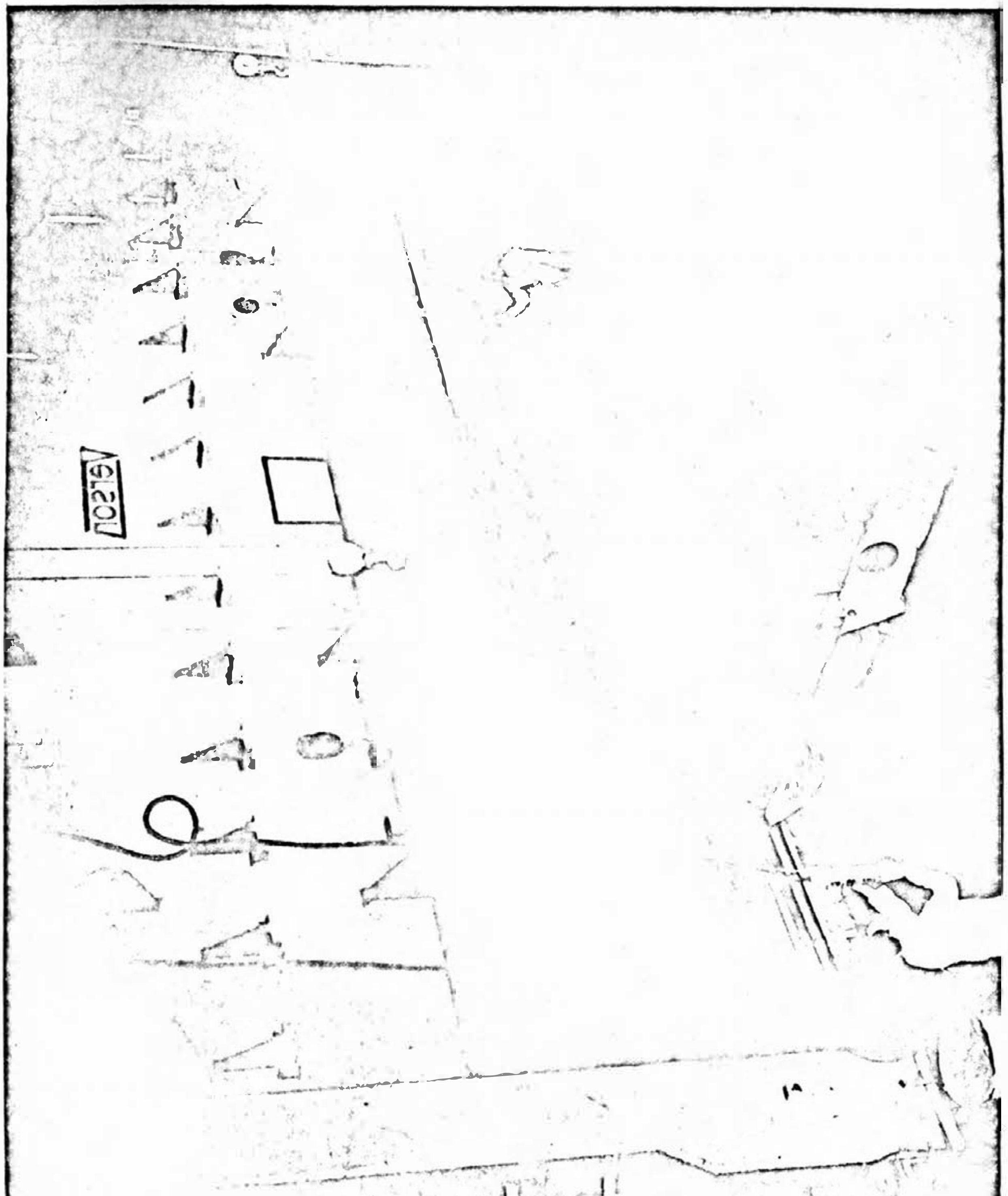


Fig. 9

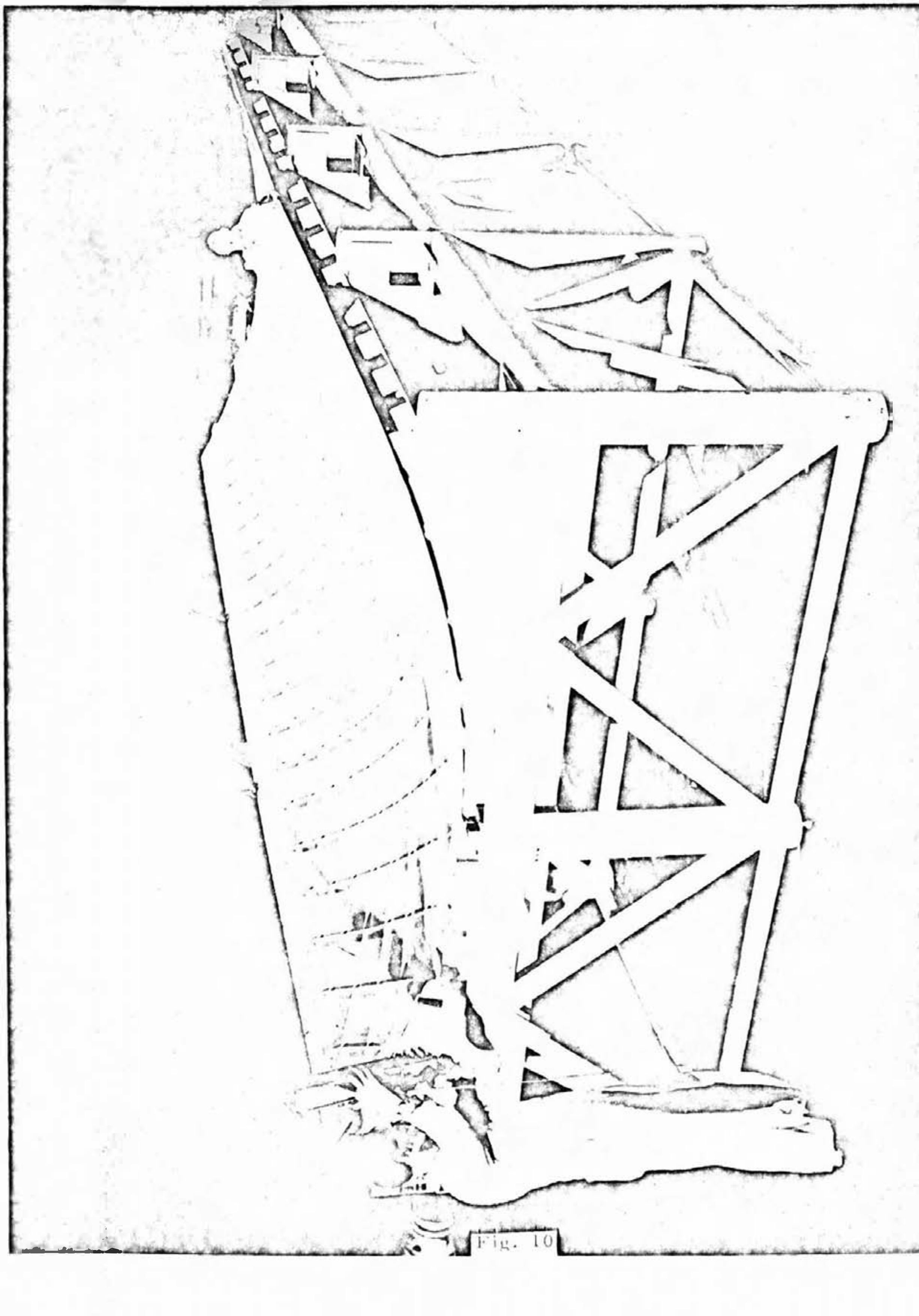
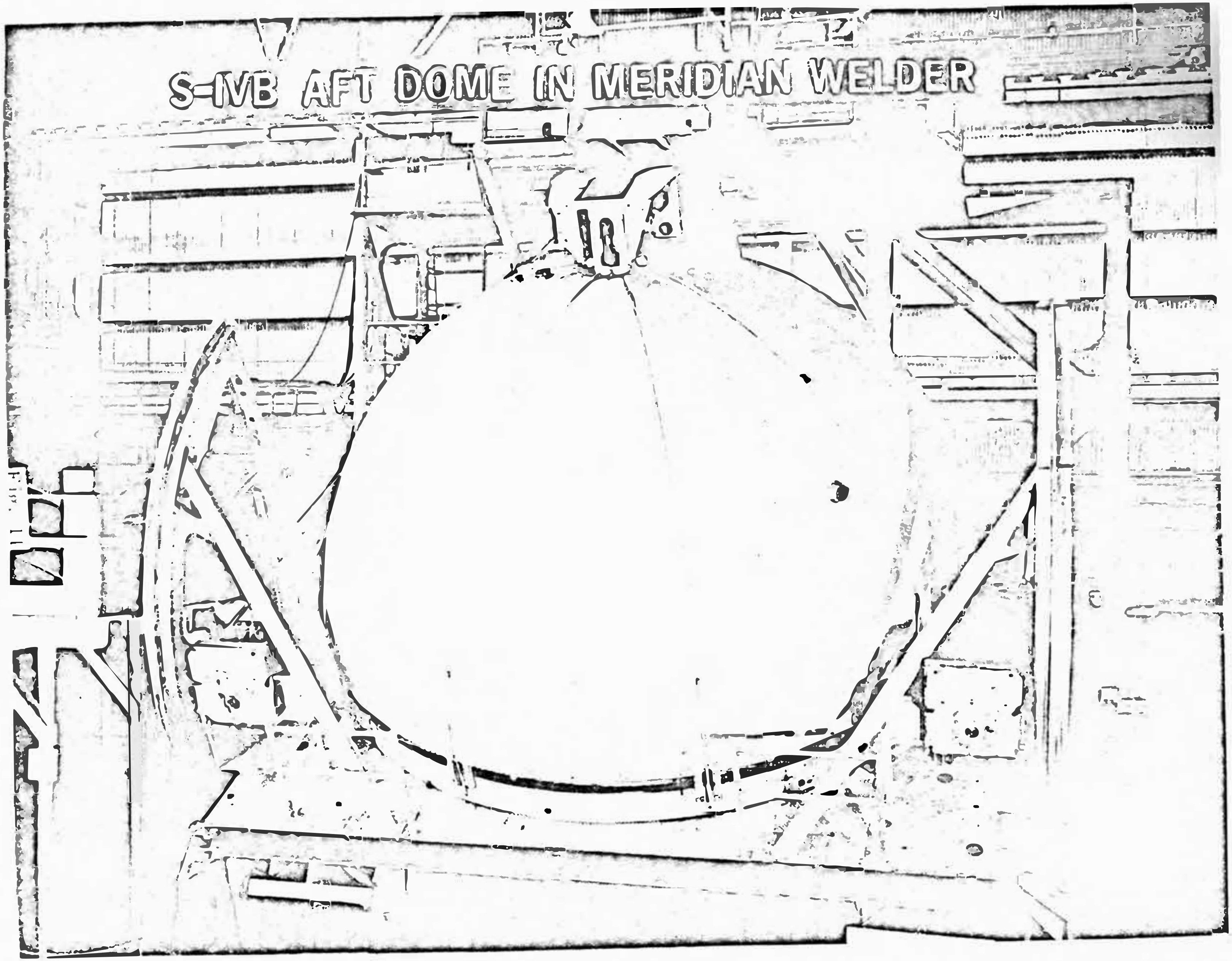
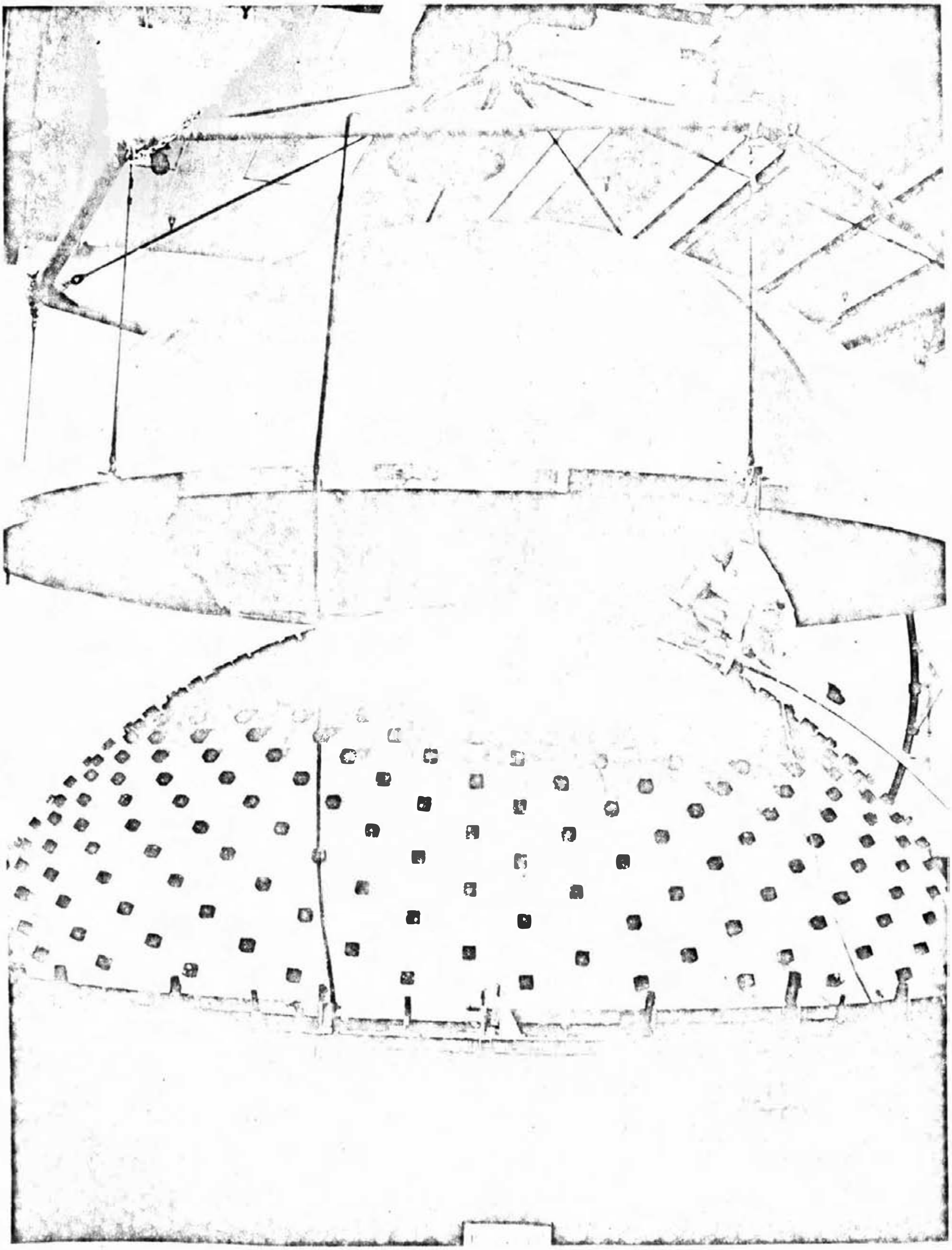
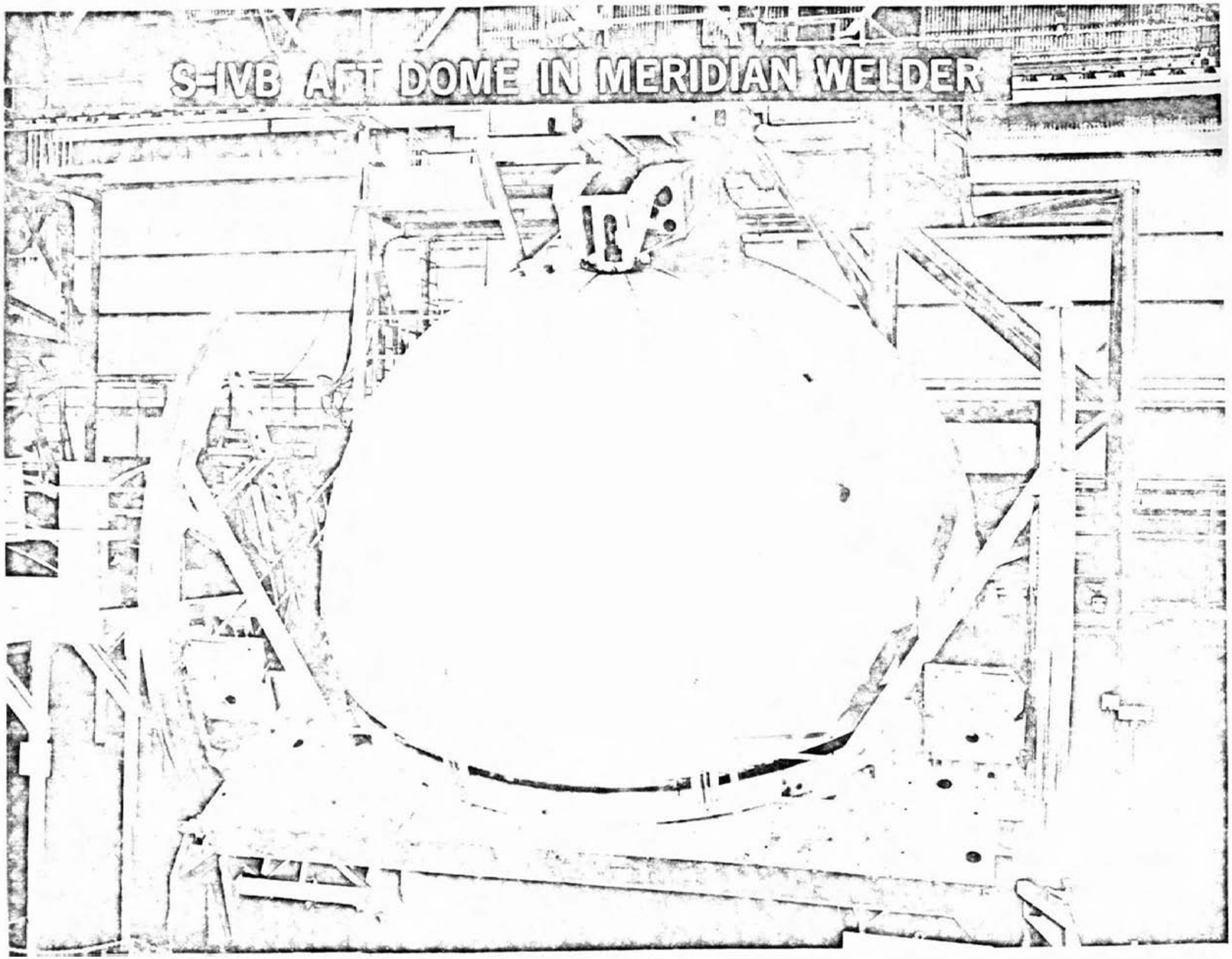


Fig. 10


S-IVB AFT DOME IN MERIDIAN WELDER




S-IVB AFT DOME IN MERIDIAN WELDER

TESTING PROGRAM

- Accounts for 50% of total effort
- Three Major Phases
  - Ground Test
  - Static Firing
  - Development Flight Test
- Progressive Program
  - Research Testing
  - Development Testing
  - Component Qualification Testing
  - Sub-System Testing
  - System Testing
  - Vehicle Testing

Fig 14

GROUND TEST PROGRAM

- COMPONENT QUALIFICATION TESTS
  - Approximately 1000 component and subsystem tests - 30,000 component environments.
  - Multi-environment approach is used to fully simulate flight conditions.
- VEHICLE DYNAMICS TESTING
  - Performed at Marshall Space Flight Center using flight configured vehicle with components mass simulated.
  - Determines vehicle bending modes and other dynamic characteristics.
- STRUCTURAL ELEMENTS STATIC LOAD TESTED TO DESIGN SAFETY MARGIN WITH FULL INSTRUMENTATION.
- FACILITIES VEHICLE - PROPELLANT LOADING TEST OF FLIGHT WEIGHT VEHICLE AT CAPE KENNEDY WITH FLIGHT CONFIGURATION PROPELLANT LOADING SYSTEMS.

Fig 15

STATIC FIRING PROGRAM

- BATTLESHIP TESTING
  - Utilizes heavy weight non flight tanks with flight engine and propellant system components.
  - Provides increased margin of safety.
  - Allows several months lead in hot testing program.
  - Planned test duration, 4000 seconds.
- ALL SYSTEMS TESTING
  - Utilizes Flight configuration stage.
  - Checkout and control by automatic GSE, with additional non flight instrumentation.
  - Full duration static firings planned with limits of flight environment explored.
- STAGE ACCEPTANCE FIRINGS
  - Full duration static test of each flight stage utilizing flight telemetry plus hardware instrumentation.
  - Post firing checkout with automatic GSE prior to shipment to Cape Kennedy.

Fig 16

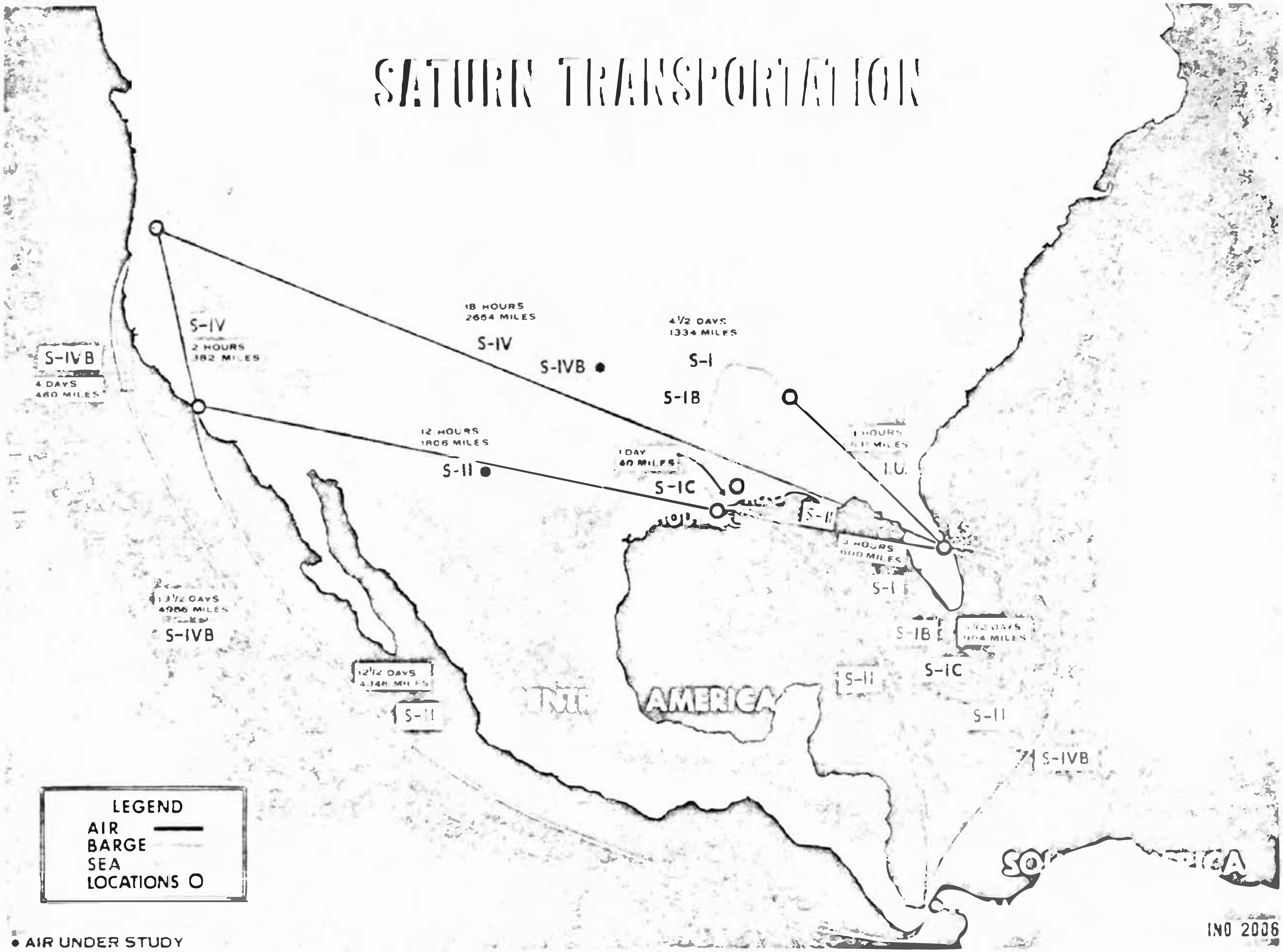
FLIGHT TEST PROGRAM

- S-IV Stage, Saturn I Program will contribute directly.

- Saturn IB Flight Tests provide advance data on:
  - Common S-IVB Stage Systems
  - Associated Support Systems

- Saturn V First Unmanned Flights
  - Heavily instrumented with additional R&D instrumentation
  - Example - Liquid separator performance during orbital coast to be verified

Fig 17


SATURN TRANSPORTATION
18 HOURS
2684 MILES
S-IV
S-IVB
4 1/2 DAYS
1334 MILES
S-I
S-IB
S-IV
2 HOURS
382 MILES
S-IVB
4 DAYS
480 MILES
12 HOURS
1806 MILES
S-II
1 DAY
S-IC
3 HOURS
600 MILES
13 1/2 DAYS
4986 MILES
S-IVB
12 1/2 DAYS
S-II
S-IB
S-IC
S-II
S-II
S-IVB
AMERICA
LEGEND
AIR
BARGE
SEA
LOCATIONS
• AIR UNDER STUDY
INO 2006

SUPPORT TO MISSION OPERATIONS

- PRIMARY GOAL IS TO TRANSFORM EXPERIENCE GAINED IN STAGE DEVELOPMENT INTO LAUNCH OPERATIONS.
  - Vehicle checkout operations at Cape Kennedy will utilize automatic checkout routines proven in factory and test site operations.
  - Contractor provides support in stage operations at launch site and technical support at the launch control center.

Fig 19

ANALYSIS OF FLIGHT RESULTS

- Quick reaction to malfunctions detected in first flights is necessary - next flight stage will be in or have completed acceptance firing

- Stage telemetry system and instrumentation

  - Five sub systems consisting of:

    (3) PAM/FM/FM Links
    (1) SS/FM Link
    (1) PCM/FM(DDAS) Link

  - Present planning calls for approximately 430 measurements telemetered in early developments flights

  - Provides about 15% increase in measurements depending on frequency response required

- Processing of data collected

  - Detailed reduction and analysis of S-IVB peculiar data by contractor

  - Overall vehicle and stage performance evaluation by MSFC

Fig 20