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SATURN I FLIGHT TEST EVALUATION

SA-1/SA-10

by

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Saturn I Flight Test Evaluation

Just about two months ago I'm sure that many of you were thrilled by the realization that we have started into the next to the last lap of the race to the moon by successfully launching the first Apollo Saturn IB space vehicle. The very first flight of a new launch vehicle was a complete success in every respect. This, fortunately, has continued our string of successes which we achieved on the Saturn I flight test program - ten successful flights out of ten launches.

To most people once a space vehicle is launched the work is over. Not so however. Even from a successful flight, much can be learned. After each flight there is a very substantial effort expended to dissect the flight, looking into every aspect of it. This is the part of a space vehicle testing program that I would like to discuss this evening. I can touch in only a superficial way on some of the activities in a flight evaluation using some representative results arrived at in the Saturn I program for illustrations of the type of information obtained.

The techniques of flight evaluation used in the Saturn program cannot lay claim to being unique but must be considered as typical. The same general concepts are used, with different innovations and with varying depth of penetration, in any flight test program.

Before getting into the flight evaluation itself it would be well to first tell you a little about the Saturn I vehicle.

I. VEHICLE DESCRIPTION

The Saturn I program consisted of vehicles with, basically, three primary configurations as shown in Figure 1: the block I R&D, the block II R&D and the operational. This concept is indicative of an earlier philosophy of a gradual buildup in the vehicle complexity. This is in contrast to our present "all up" concept as represented by the first launch of the Saturn IB.

The block I vehicles, SA-1 through SA-4, consisted of a live first stage with dummy second and third stages and payload. The first stage was powered by eight Rocketdyne 165,000 pound sea level thrust H-1 engines. Control components to guide and steer the vehicle were carried in instrument cannisters on top of the S-I stage.

Block II vehicles consisted of two propulsive stages; both being active. The first stage was similar to that used on the Block I configuration but with the addition of stabilizing fins and using H-1 engines uprated to

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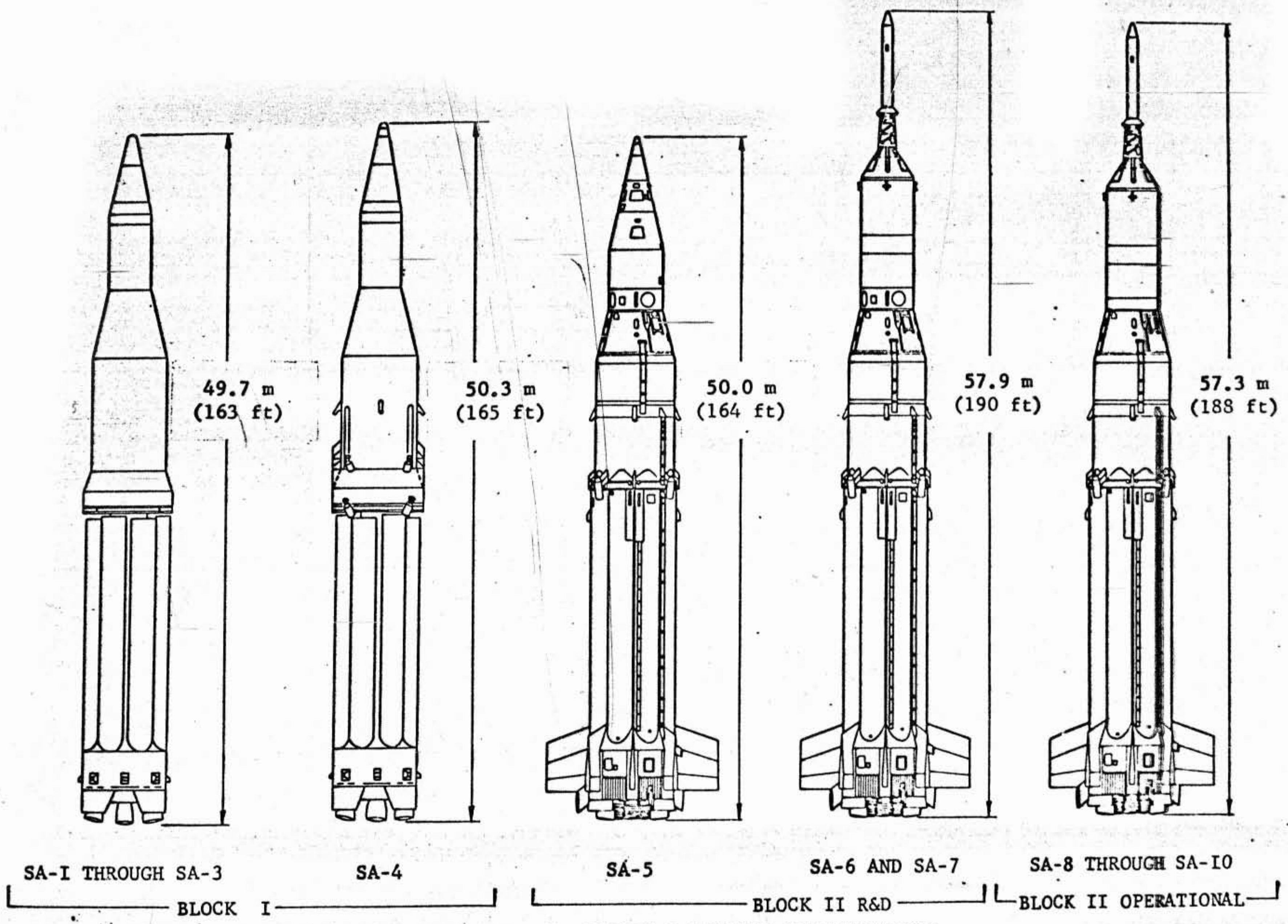

FIGURE 1 SATURN I VEHICLE CONFIGURATION

188,00 pounds thrust. The second stage, known as the S-IV and built by Douglas Aircraft, was powered by six Pratt & Whitney RL10-A3 liquid oxygen/liquid hydrogen engines. Each engine was rated at 15,000 pounds vacuum thrust.

The first block II vehicle, SA-5, carried a dummy Jupiter nose cone. SA-6 and SA-7 carried boilerplate Apollo command modules as payload.

The three operational vehicles (SA-8, SA-9 and SA-10), in addition to boilerplate Apollo command modules, carried the Pegasus meteoroid technology satellite. Guidance and control equipment for the block II vehicles was carried in an instrument unit mounted ahead of the S-IV stage. The R&D configuration consisted of a pressurized central tube and four cross tubes for the equipment mounting. The operational instrument unit was an unpressurized cylindrical section with components mounted around the periphery.

II. FLIGHT TEST PROGRAM INSTRUMENTATION

Obviously, to obtain data from a space vehicle flight for analysis requires instrumentation. With a vehicle the size and complexity of the Saturn and with the limited number of R&D flights, a very extensive instrumentation program had been incorporated. The philosophy was to instrument thoroughly enough so that if a malfunction occurred on a flight there would be enough data available to pin point the problem before flying the next vehicle. However, this still did not rule out some "trial and error" approaches to the smaller type problems.

The instrumentation may be divided into the following general classes:

1. telemetry
2. on-board tape recorders
3. on-board recoverable cameras
4. on-board television
5. ground engineering cameras
6. optical tracking
7. radio frequency tracking
8. ground acoustic measurements

9. meteorological measurements.

Telemetry information is one of the primary sources of information for the flight analyst. Measurements of parameters such as pressures, temperatures, engine deflections, control signals, event signals, electrical currents and voltages, and vibrations are sensed by transducers. A transducer generates an electrical signal proportional to the magnitude of the physical parameter being measured. The signal is then conditioned for the telemetry system and finally transmitted to the ground by the in-flight telemetry system. The signal is received at various receiving sites on the ground and recorded on magnetic tape for subsequent processing. Block I vehicles had from 566 to 618 measurements transmitted over 8 to 10 telemetry links. The number of telemetry measurements on Block II vehicles varied from 1154 to 2246. Eleven to 13 telemetry links were carried.

At stage separation, when retro and ullage rockets are fired, there is a blackout in RF communications to the ground for several seconds due to an ionization occurring over the telemetry antennas. This is also a critical period of the flight when data is expecially needed. To cover this interruption in communications, on-board tape recorders are utilized which record data during this period. The data is then played back later over the telemetry system at a time when good communications are again available and after the stage has achieved its objectives or during a period of low activity, for example, in orbital flight.

Each block II R&D vehicle carried eight encased movie cameras which were ejected from the vehicle after filming was complete. Two cameras utilized fiber optics to view the S-I/S-IV interstage area and two utilized fiber optics to view the interior of LOX tanks 03 and OC on the S-I stage. The remaining four cameras provided a direct viewing of the exterior of the vehicle. The arrangements of the cameras and associated euipment are shown in Figure 2.

The cameras were ejected 20 seconds after separation of the S-I from the S-IV stage. The capsules followed a ballistic path to an altitude of about 7,500 feet where paraballoons were deployed to decelerate the capsules and keep them afloat after impact until they were recovered. Impact occurred at ranges varying from about 420 to 490 nautical miles downrange.

Seven of the eight cameras were recovered on SA-5, all on SA-6 and three were recovered on SA-7. The cameras from SA-7 could not be recovered immediately since hurricane Gladys was in the impact area at the time of flight. Two of these cameras were recovered 50 days after the flight and the third was found even later. The film was still good in the first two cameras but that from the third was bad.

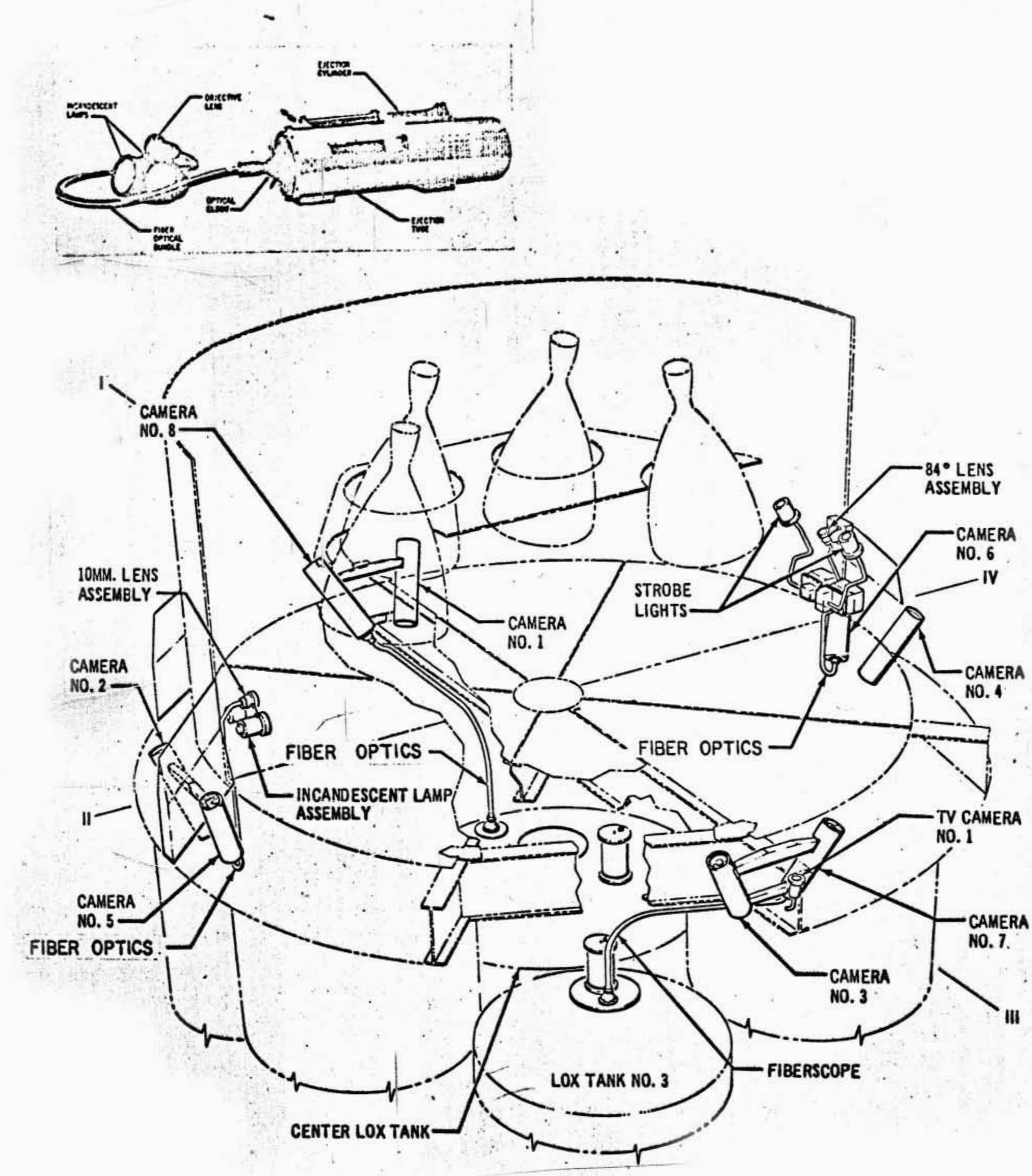

FIGURE 2 ONBOARD CAMERA LOCATIONS

Television cameras were also carried on all Block II vehicles except SA-10. The most dramatic usage of TV was in conjunction with the monitoring of the wing deployment of the Pegasus satellite. Excellent coverage was obtained and gave positive qualitative verification of the deployment.

Tracking information of the ascent trajectory was obtained from fixed cameras, theodolites, Azusa, UDOP, ODOP, Mistram, Glotrac and FPS-16 or TPQ-18 type radars. Orbital tracking of the Block II vehicles was provided by Radar beacon and skin tracking, minitrack and optical tracking.

III. SATURN I FLIGHT TEST RESULTS

Trajectory

Figure 3 summarizes the trajectory profiles of all Saturn I flights. As indicated in this chart the first four flights were ballistic flights. SA-2 and SA-3 were intentionally destructed at altitude to release the water ballast from the upper stages into the upper atmosphere as part of a scientific experiment, "Project Highwater." SA-6 was the first vehicle to be flown with adaptive guidance flown in closed loop during the S-IV powered flight phase. This flight provided an unexpected, but excellent, test of the capability of the guidance system. One of the engines on the first stage failed, cutting off prematurely, and a malfunction of the thrust controller also occurred on one of the second stage engines, causing a higher than expected thrust level. The guidance system corrected for these very significant changes in performance and steered the vehicle along an adapted and re-optimized trajectory to the proper altitude and velocity conditions for insertion into the desired orbit.

Vehicles SA-8, SA-9 and SA-10 were put into a higher orbit to obtain a longer lifetime for the Pegasus meteroid experiment. SA-9 and SA-8 were inserted into elliptical orbits with a perigee of about 270 n. mi. and an apogee of about 400 n. mi. SA-10 was inserted into a nearly circular orbit of about 286 n. mi.

Figure 4 shows a comparison of the velocity differences actually achieved with predicted at outboard engine cutoff of the S-I stage and S-IV stage cutoff. The deficit in velocity of SA-1 was due to a 1.6 second premature cutoff with a significant contribution to this caused by propellant sloshing activating the propellant level cutoff probe prematurely. The large deficit of 328 ft/sec on SA-6 at S-I stage cutoff was due to the premature engine failure. Most of the remaining vehicles indicate a general increase in level of performance over predicted. The excess velocity at S-IV cutoff

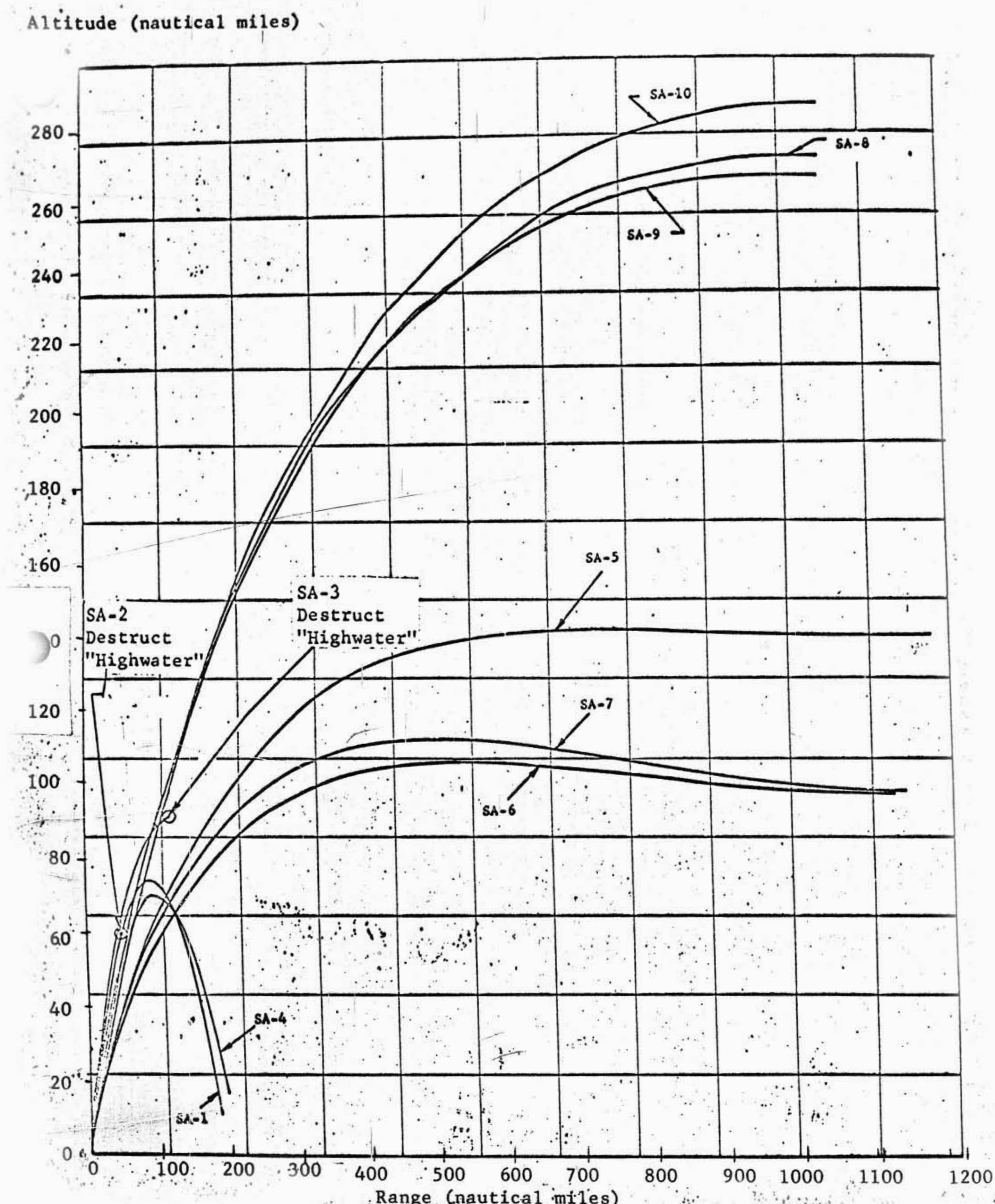

FIGURE 3 TRAJECTORY PROFILES

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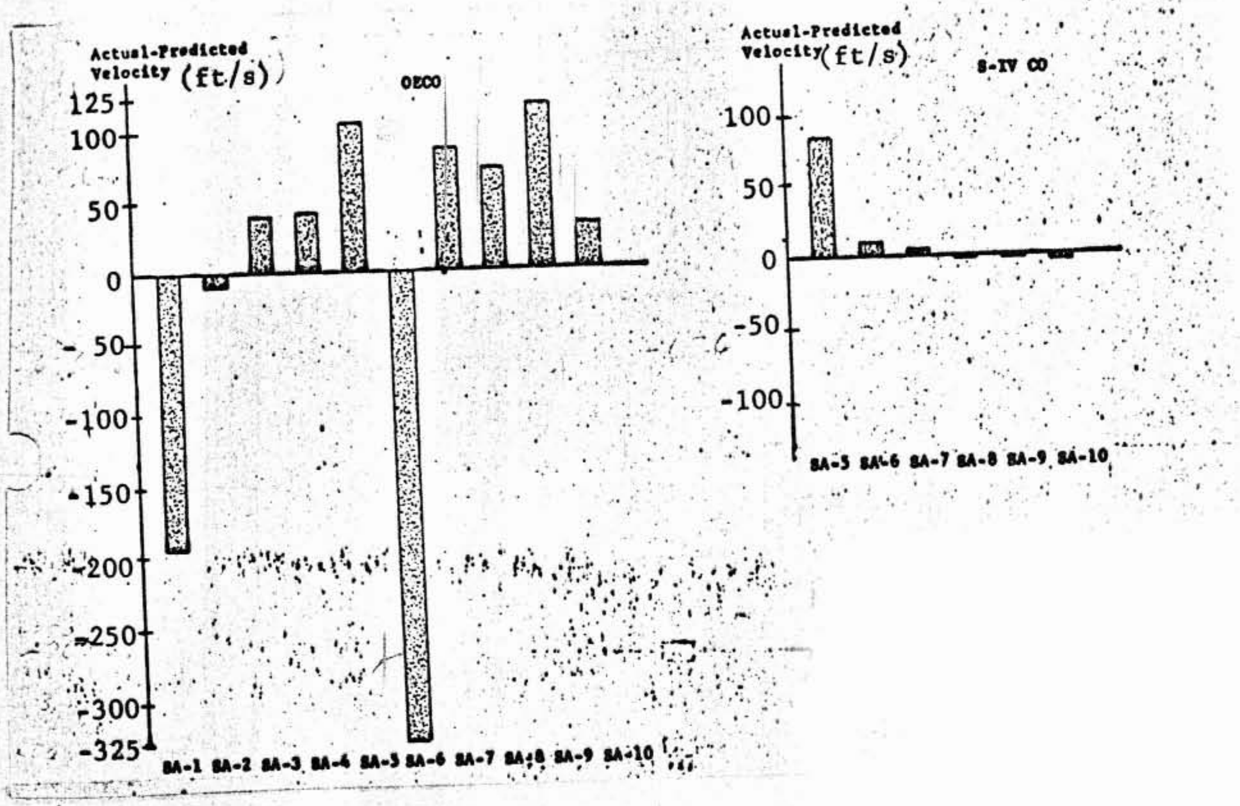


FIGURE 4 VELOCITY DEVIATIONS

shown for SA-5 was because this vehicle did not have a guidance cutoff capability and the S-IV stage burned to a depletion cutoff.

Propulsion

The propulsion performance of the S-I stage was satisfactory on all flights except one, SA-6.

The S-I stage had been designed to have an engine out capability. To actually test this capability, an engine was intentionally shut down on SA-4 at approximately 100 seconds of flight by a preset timer signal. This shutdown was accomplished smoothly and the remaining engines burned several seconds longer to offset the loss in performance. The propellant interchange systems functioned properly to feed the remaining seven engines.

While SA-4 was a planned test of the engine out capability, an actual failure of an engine on SA-6 also proved the capability of the clustered engine concept. On this flight, the engine number 8 turbopump assembly failed causing the engine to be shutdown at 117.3 seconds of flight. The engine shut down in 1.7 seconds compared to a nominal 2.4 seconds for a normal engine cutoff. The failure caused a sudden increase in the telemetered temperatures of several specific high speed pinion bearings and of a bearing on the turbine shaft in the propellant turbopump assembly. Analysis of the telemetered information led to the conclusion that the teeth were stripped from one of the gears in the turbopump causing the abrupt stop of the engine. Previous ground testing of the turbopump had indicated a possible marginal design to that a change had already been planned to increase the width of the gear teeth. Coincidentally, this was to be effective on the next vehicle, SA-7. No further significant problems were experienced with the H-1 engines.

One important aspect of vehicle performance is the efficiency of propellant utilization. This is especially noticeable for the first stage for which maximum propellant usage is desired for every flight. Propellant usage for the S-I stage was determined indirectly from engine analysis to obtain flow rates and directly by measurements within the propellant tanks. Direct measurements consisted of discrete level probes and continuous level probes in each tank and propellant level cutoff probes in two outer LOX tanks and two fuel tanks.

Figure 5 shows the propellant tank interchange systems used to feed the engines on the Block I and Block II configurations. Because of the difficulty of predicting the flow behavior from ground tests and theoretical calculations it was necessary to utilize flight experience to improve predictions of propellant usage.

Figure 6 shows a factor, referred to as the Propellant Utilization Number, to indicate the efficiency of propellant usage. This shows the steady increase throughout the flight program. A noticeable increase is noted for the block II vehicles with the improved propellant feed system. The relatively poor efficiency for SA-1 was due to the sloshing problem and that for SA-6 was due to the premature engine shutdown.

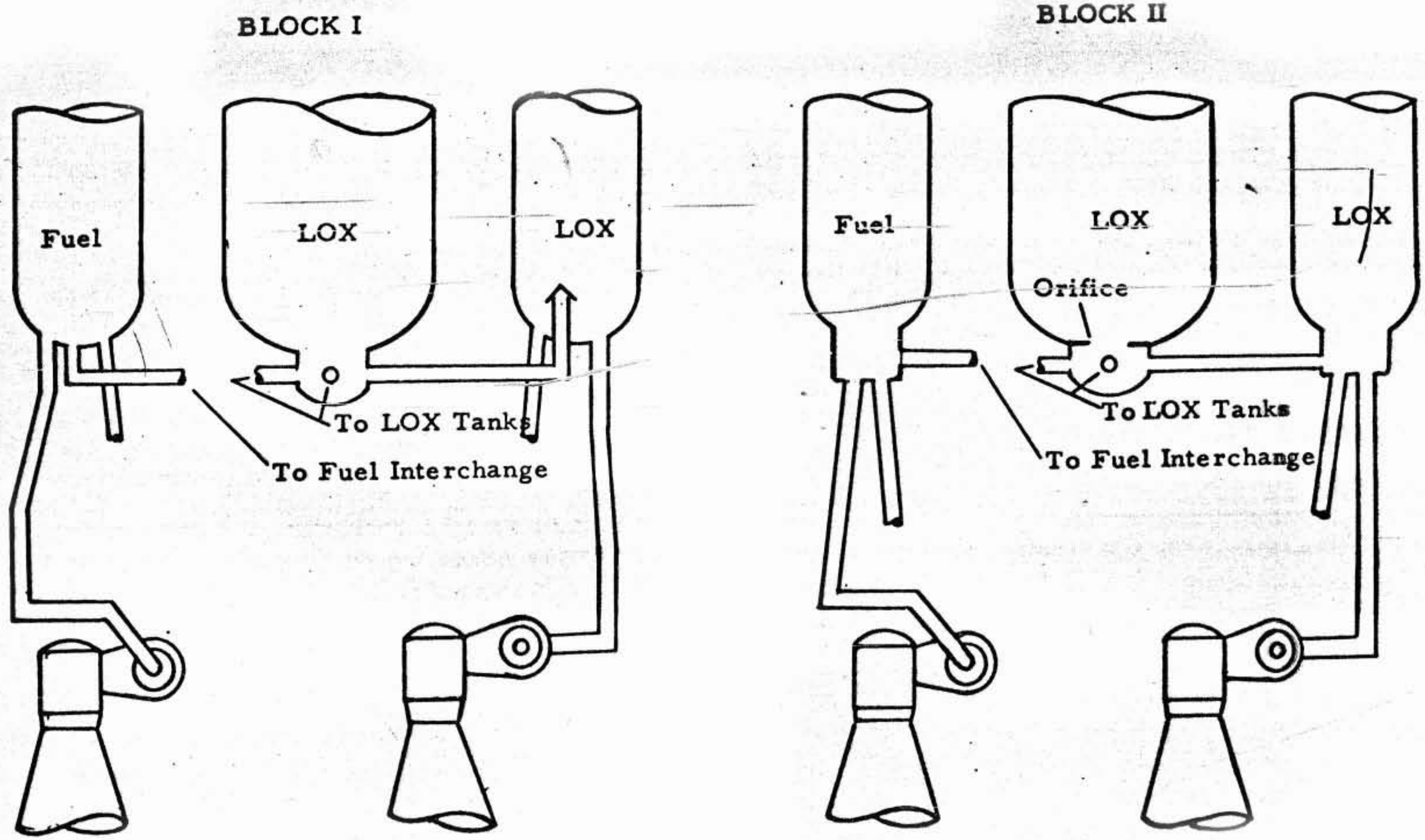
Performance of the S-IV stage propulsion system in flight was normal except for a malfunction of the thrust controller on one of the engines on SA-6. This malfunction caused the thrust level on this engine to be unregulated, causing it to run 9.5% above predicted. No detrimental effects were noted from this high level of operation.

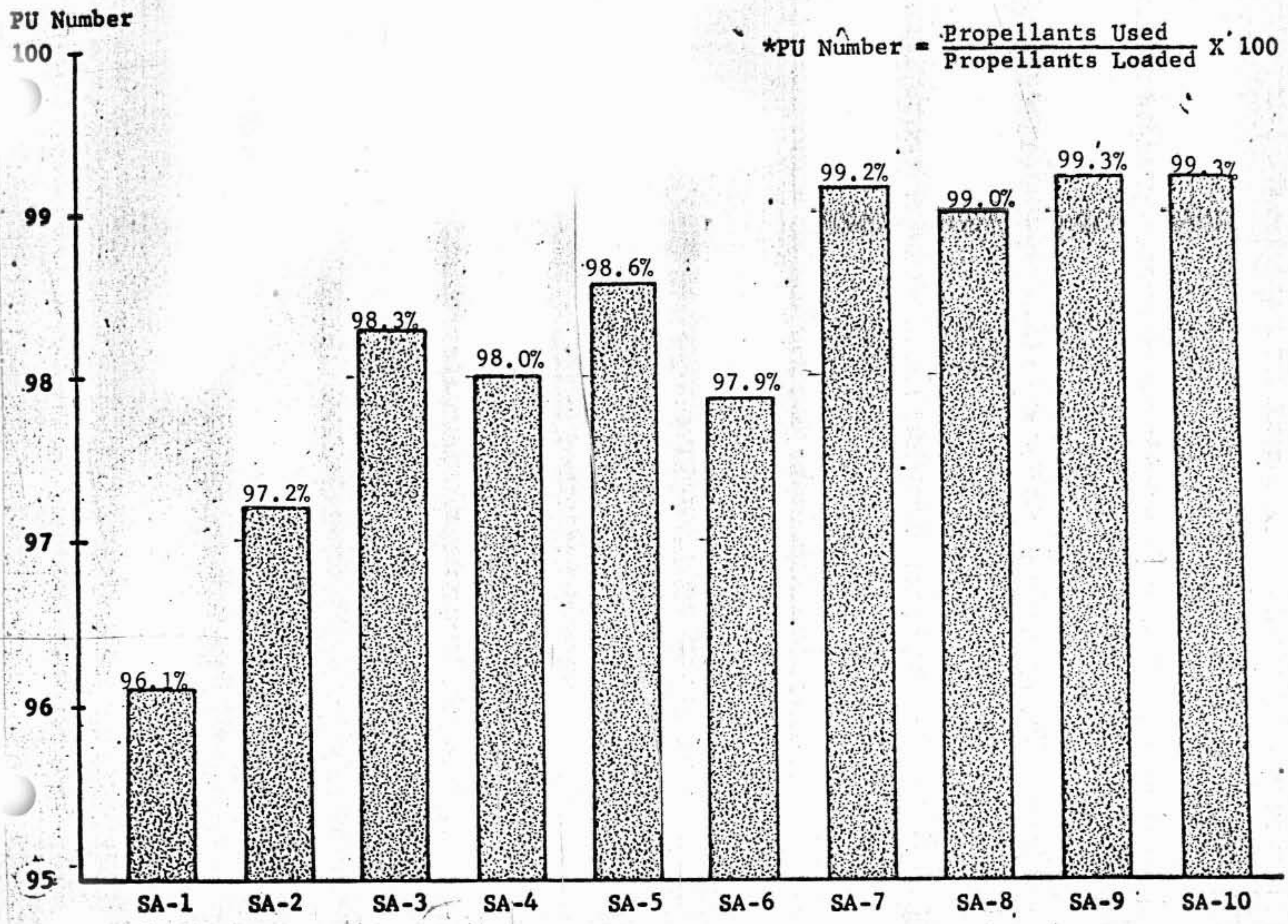
One consistent and unexpected phenomena came out of the analysis of the prupulsion performance of the S-IV stage. There are two basically different ways that a vehicle propulsion system performance may be analyzed. One, is based on analysis of telemetered measurements of engine parameters such as combustion chamber pressure, turbine rpm's, propellant flow rates, etc. The other approach is a trajectory simulation technique. In this latter approach it is assummed that the shape of the thrust and flow rate curves with respect to time are known from the telemetered engine measurements but that the average level is unknown. Adjustments to these levels are made so that a simulated trajectory time history would match within very close tolerances the trajectory as determined from tracking data. Specific impulse, especially, can very precisely be determined from this approach. (Specific impulse is the amount of thrust per unit of propellant flow rate - lb/lb/sec).

Specific impulses of the S-IV stage obtained from the flight analysis are shown in Figure 7. The first three flights all indicated the specific impulse of the stage was consistently low; being between 0.6 to 1.0 percent below predicted. After the flight of SA-7, the evidence was considered sufficiently conclusive to make a change in the performance predictions. For the last three flights the predicted specific impulse was decreased by 0.7%. Post-flight analysis verified that this adjustment was essentially correct.

It has been theorized that the decrease in specific impulse observed from the flights was not attributed to the individual engine operation but was a consequence of the clustering of the engines. Some model tests were conducted subsequent to the flights in an attempt to identify any effects associated with interaction of the external flow from the engines. These tests gave no further evidence towards an explanation and at the present time the cause is still a mystery. The problem still has some importance because of its implications for future stages with clustered engine configurations.

FIGURE 5 LOX & FUEL INTERCHANGE





S-I STAGE PROPELLANT UTILIZATION NUMBER

PROPELLANT RESIDUAL DIFFERENCES (lb)

Act-Pred:	SA-1	SA-2	SA-3	SA-4	SA-5	SA-6	SA-7	SA-8	SA-9	SA-10
Fuel	582	2789	3003	1404	-1470	-2425	-769	1742	-342	-485
LOX	13625	4447	1314	-1217	154	7266	421	-357	-291	- 35
Total	14207	7236	4317	187	-1316	4841	-348	1385	-633	-520
Type of Out-board CO	Out-board Timer CO	Out-board Timer CO	Out-board Deple-tion CO	Out-board Deple-tion CO	Out-board Timer CO	Out-board Timer CO	Out-board Timer CO	Out-board Deple-tion CO	Out-board Deple-tion CO	Out-board Timer CO

FIGURE 6 S-I STAGE PROPELLANT UTILIZATION

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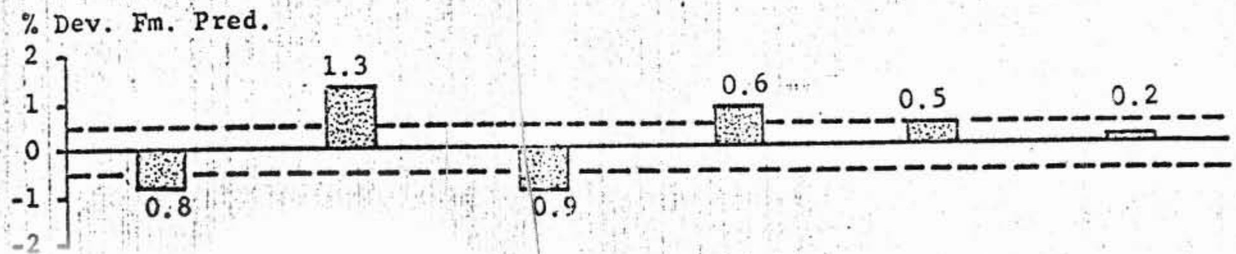
THRUST:

Average Long't Thrust (Vehicle) Vacuum Conditions (lb)

	SA-5	SA-6	SA-7	SA-8	SA-9	SA-10
Actual	88,909	90,412	89,004	89,987	89,360	89,053
Pred	88,981	89,232	89,806	89,460*	88,903*	88,897*
Act-Pred	-72	1,180	-802	527	457	156

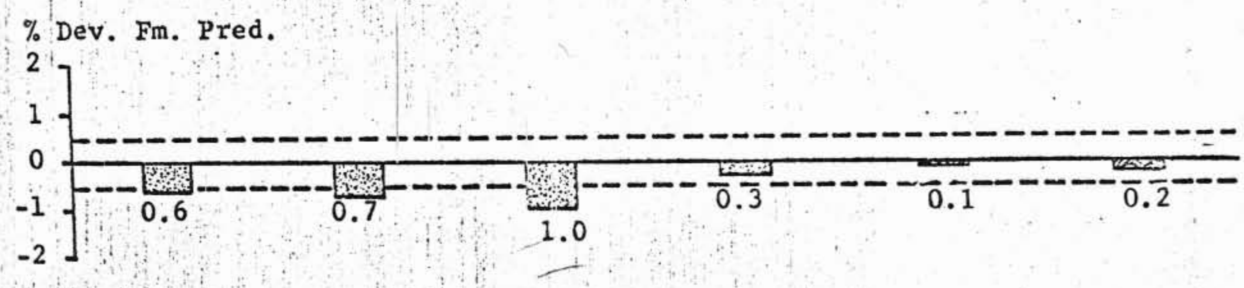


——— Pred. 2σ Perturbations (±0.5%)

SPECIFIC IMPULSE:

Average Long't I_{sp} (Vehicle) Vacuum Conditions (sec)

	SA-5	SA-6	SA-7	SA-8	SA-9	SA-10
Actual	424.9	426.6	425.3	424.1	425.9	424.9
Pred	427.4	429.5	429.5	425.2*	426.5*	425.8*
Act-Pred	-2.5	-2.9	-4.2	-1.1	-0.6	-0.9



*Biased 0.7% lower than engine results to account for deviations shown on the first three flight tests (cluster effect and base pressure effect).

FIGURE 7 S-IV PROPULSION PERFORMANCE SUMMARY

Control

A space vehicle control system is designed to provide control forces to steer the vehicle along the desired flight path. This path may be defined by a pre-set fixed program (as the tilt program in the pitch plane during S-I burn) or an attitude command based on signals generated by the guidance system to steer to desired end conditions. The other function of the control system is to provide required vehicle stability in the presence of such forcing functions as winds, torques due to vehicle assymmetries, vehicle bending, and propellant sloshing.

Control sensors used during the Saturn I program are summarized in Figure 8. The table indicates the systematic and gradual buildup followed in proving flight qualified hardware by flying it as a passenger prior to actually being incorporated in closed loop control. This is reflected in the use of the ST-90 stabilized platform and local angle of attack meters which were adapted from the military Jupiter missile program. The ST-90 had a fixed azimuth only capability. Later Saturn vehicles required a variable flight azimuth capability since the launch complex for a vehicle as large as the Saturn could only be built for a fixed launch azimuth. Therefore, a new stabilized platform, the ST-124, was developed to provide this variable flight azimuth capability. As an interim measure, the ST-90 was modified to the ST-90S to provide the capability to roll the vehicle after liftoff from the launch azimuth to the desired flight azimuth. All block II vehicles, except SA-10, were rolled from a 90 degree launch azimuth to a 105 degree flight azimuth. The flight azimuth of SA-10 was 95.2 degrees.

Lateral load relief was accomplished by first using the local angle of attack sensors. It was originally planned to phase over from the local angle of attack sensors to a Q-ball type angle of attack sensor. This sensed differential pressure over a sphere in an instrument mounted on the top of the launch escape system motor atop the spacecraft. Originally, there was some concern over incorporating accelerometers in closed loop control because of the sensitivity of its location on the vehicle for proper stabilization of bending and also because of localized vibrations. However, after flight data was investigated with respect to body bending characteristics and localized effects on the first few flights it was decided to put the accelerometers in closed loop control on SA-4. Accelerometers were on all subsequent Saturn I flights.

One of the most significant factors that influences a space vehicle flight is the wind environment during atmospheric flight. The magnitude and

Function	Type Sensor	S-I Flight Stage Utilization										S-IV Flight Stage Utilization					
	SA-	1	2	3	4	5	6	7	8	9	10	5	6	7	8	9	10
Attitude Reference:	ST-90 Platform	X	X	X	X												
	ST-90S Platform					X	X					X	P				
	ST-124P Platform			P	P												
	ST-124 Platform					P	P	X	X	X	X	P	X	X	X	X	X
Angular Rates:	Electrical Differentiation (In Control Computer)	X	X	X	X	X	X					X					
	Control Rate Gyros	P	P	P	P	P	P	X	X	X	X	P	X	X	X	X	X
Lateral Loads:	Local Angle-of-Attack meters	X	X	X	P			P		P	P						
	Q-ball Angle-of-Attack Meters (Control type)				P	P	P	P	P	P	P						
	Prototype Q-ball	P	P	P													
	Control Accelerometers	P	P	P	X	X	X	X	X	X	X						

X = Closed Loop Active Sensor
P = Passenger

FIGURE 8 CONTROL SENSORS

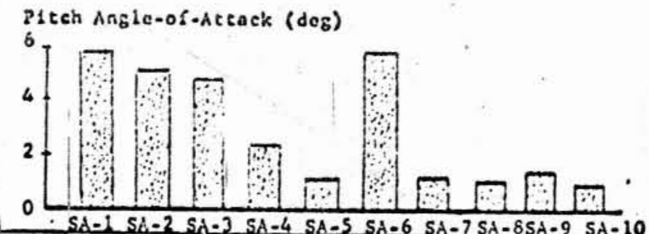
characteristics of the wide (shears, direction changes) influence both the amount of control capability required and the structural loads.

In order to minimize control requirements in case an engine failed, both SA-1 and SA-2 tilt programs were based on seven engines operating. This was done to minimize the control requirements in case an engine did fail. The SA-3 tilt program was based on eight engines operating until 20 seconds of flight and seven engines for the remainder of flight. By SA-4 sufficient confidence had been obtained in the engine reliability that a decision was made to base the tilt program on that flight and all subsequent on eight engines operating. This provided optimum payload capability for eight engines operating during the entire required time.

Tilt programs for the flights of SA-4, SA-5 and SA-9 were all biased to fly a zero angle of attack with the mean wind profile for the month of launch. This was done to increase the probability of launch under maximum wind conditions. SA-6, launched in May when very light winds prevail, utilized a tilt program intentionally biased to produce a larger angle of attack. This was done to obtain improved information on vehicle aerodynamic static stability parameters and structural loads. Head winds of 20 m/s during the maximum dynamic pressure region of flight of SA-6 yielded a maximum angle of attack of 5.5 degrees. Figure 9 shows the maximum pitch component of angle of attack for all ten flights and the maximum wind encountered during the Block I and Block II flights. Also shown for comparison is the design wind profile.

One of the more significant problems in the Saturn I program with respect to control and stability occurred on the SA-1 flight. A sloshing instability was discovered near the end of the powered flight. Although it did not approach the point of endangering the vehicle, it did very likely contribute to a slightly premature cutoff.

The sloshing was indicated indirectly by many telemetered measurements such as angular velocities, engine deflections and normal accelerations. The most direct evidence was obtained from differential pressure measurements obtained from probes installed in three of the propellant tanks to monitor sloshing of the propellants. Probes were installed in the 105 inch diameter center LOX tank, in one 70 inch LOX tank and one 70 inch fuel tank. These sensed a differential pressure directly and the measurements had to be multiplied by a theoretically determined conversion factor to determine the actual propellant wave height. This factor was a function of liquid level, longitudinal acceleration and the frequency of oscillations. A typical time history plot of the slosh heights deduced from the differential pressure measurements is shown in Figure 10. Corresponding vehicle responses are shown in Figure 11.



	SA-1	SA-2	SA-3	SA-4	SA-5	SA-6	SA-7	SA-8	SA-9	SA-10
*	7	7	7/8	8	8	8	8	8	8	8
Wind Bias	No	No	No	Yes	Yes	**	No	No	Yes	No

*Indicates the number of engines operating upon which the tilt program is based.

**Programmed for 4 or more degree

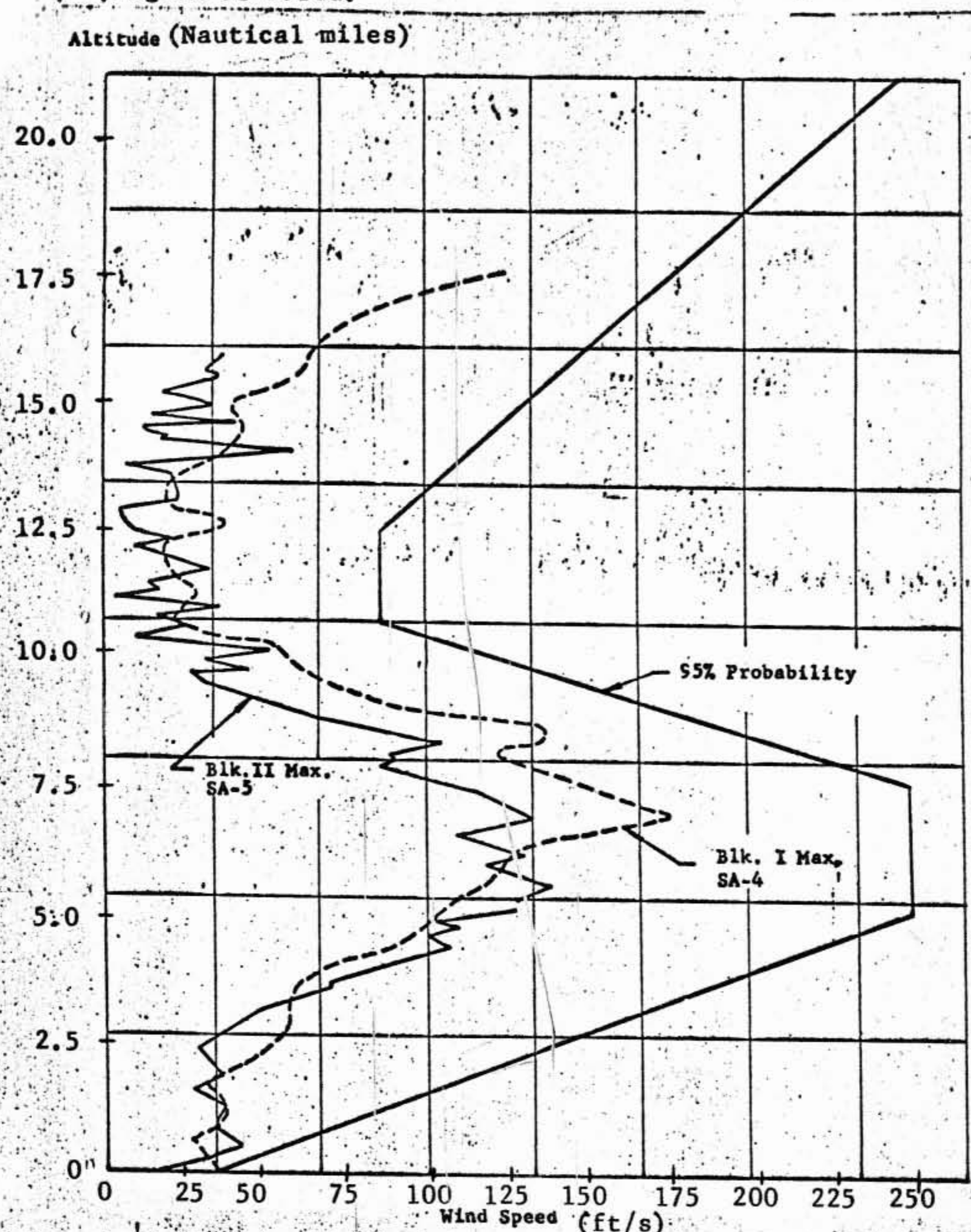


FIGURE 9 WINDS AND ANGLE-OF-ATTACK (α) MAXIMUM (PITCH PLANE)

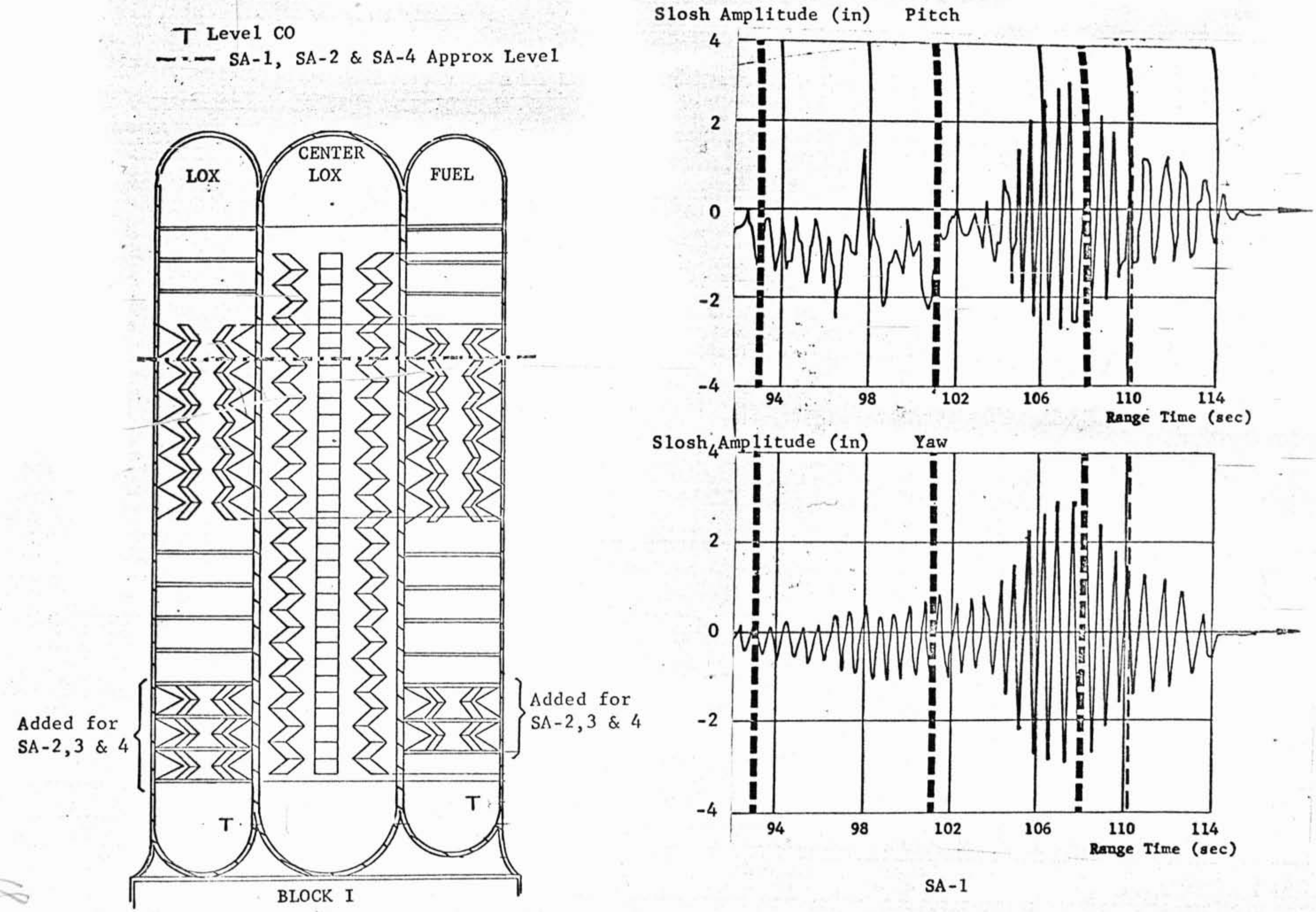


FIGURE 10 SLOSH BAFFLE CONFIGURATION AND SLOSHING AMPLITUDES

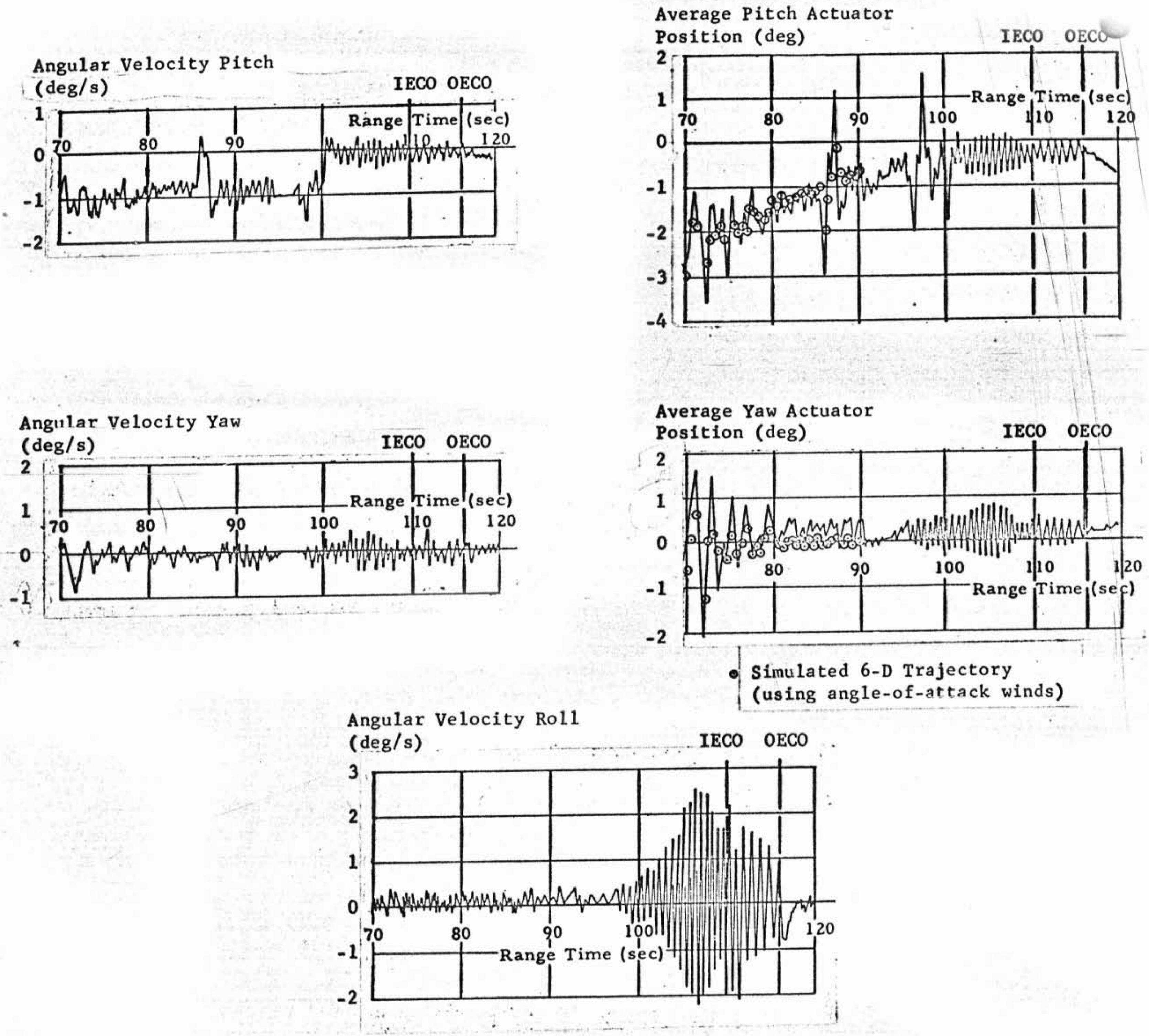


FIGURE 11 RESPONSE OF SA-1 TO PROPELLANT SLOSHING

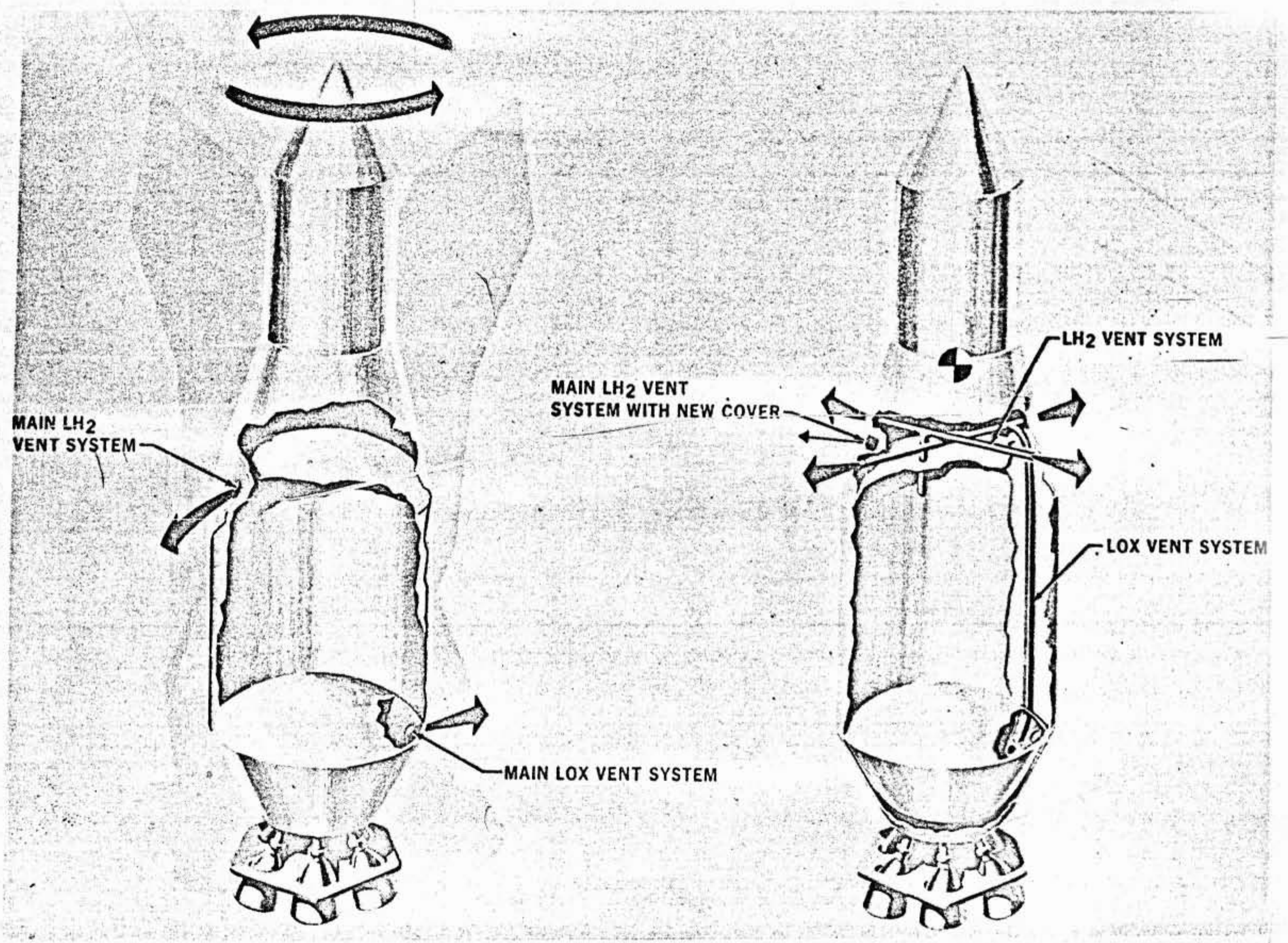
While a slight instability of sloshing showed up in the pitch and yaw planes, that in the roll plane was most serious. A theoretical analysis after the flight indicated an instability beginning around 60 seconds of flight for the case if sloshing in the outer propellant tanks was considered to be oscillating in phase in the roll mode. Maximum sloshing forces probably occurred around 107 seconds as indicated by the peak response in roll occurring at this time.

Actual IECO was given by a level sensor probe in one of the fuel tanks. A sloshing amplitude of 5 inches at the tank wall would give an 0.8 second early cutoff. The remaining time difference (1.6 sec actual early cutoff) could have been due to differences in engine performance or due to additional slosh height. Exact relative contributions were essentially impossible to determine.

Orbital Attitude

An interesting example of how flight test evaluation influenced a vehicle sub-system design is illustrated by the experiences with the S-IV stage propellant tanks venting in orbit. Venting of the propellant residuals was necessary to prevent an excessive pressure buildup in the LOX and LH_2 tanks during orbital flight. Venting of the LH_2 tank took place over approximately a 1.5 hour period and the LOX tank required about 24 hours. In the original design no special concern was given to the effect of venting on vehicle angular rates in orbit. The particular venting geometry used on SA-5 and SA-6 (Figure 12) produced excessive angular rates after S-IV stage cutoff. The Pegasus experiments scheduled for the operational vehicles had a design requirement not to exceed a 6 deg/sec roll rate. The actual rates experienced on the Block II R&D flights are shown in Figure 13.

A non-propulsive vent system was installed on SA-7 as shown in Figure 12. The non-propulsive vent system performed very well on SA-7 in reducing the roll rate. Considerably more propellants were left on-board SA-7 compared to SA-6, (2626 lbs, compared to 723 lbs.). Even with the larger residuals, the roll rate was reduced from 28 deg/sec on SA-6 to less than 6 deg/sec on SA-7 because of the changed vent configuration. Whereas the tumble rates on SA-5 and SA-6 had been negligible, that on SA-7 was approximately 6 deg/sec. This was also considered too high for the Pegasus. Therefore, an auxiliary "blow-down" non-propulsive vent assembly was added to SA-9, the first Pegasus flight.



NORMAL S-IV VENTING
INDUCES PITCH, ROLL, YAW MOTION

NON-PROPULSIVE VENT SYSTEM
COUNTER-BALANCES FORCES FOR STABILITY

FIGURE 12 VENTING GEOMETRY FOR SA-5, SA-6, AND SA-7

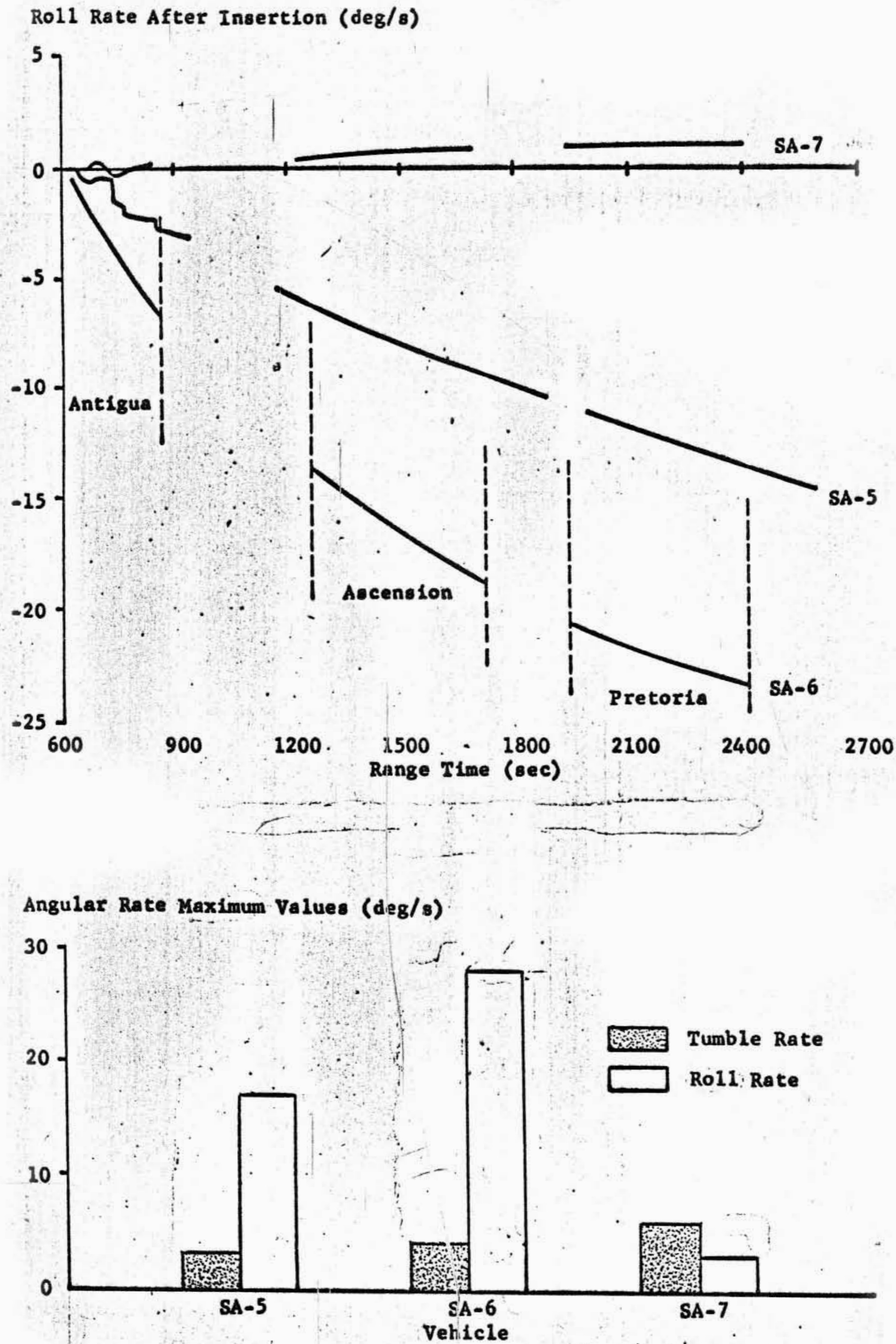


FIGURE 13. ANGULAR RATES DURING ORBIT (BLOCK II R&D VEHICLES)

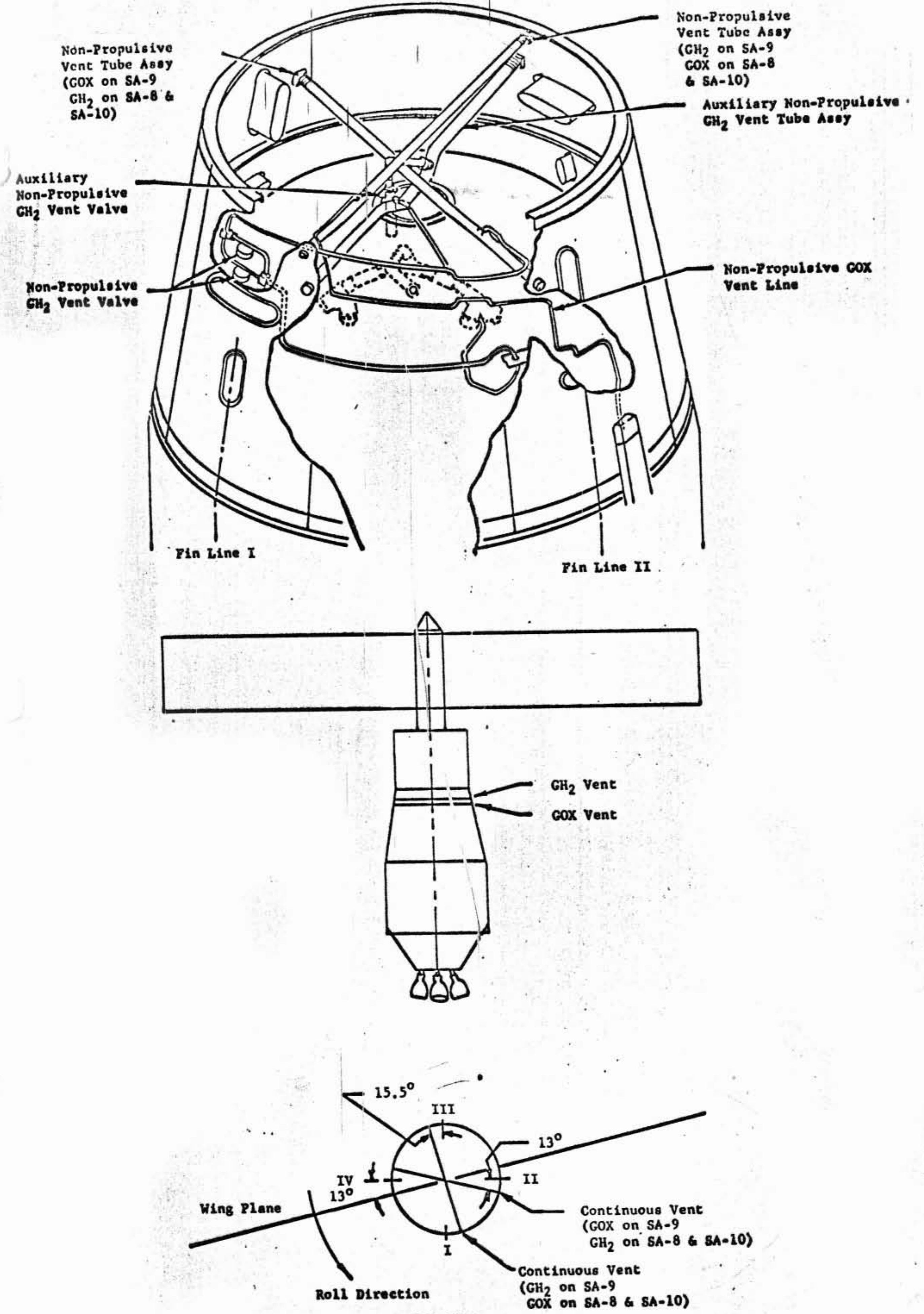


FIGURE 14 PEGASUS VENT CONFIGURATION

On the flight of SA-9 the roll rate increased to a maximum of 9.8 deg/sec. This rate had no apparent effect on the structure of the Pegasus satellite, however, it was considered to be too marginal for the remaining two flights. The geometry for the operational Pegasus vehicles is shown in Figure 14. Examination of the geometrical relationships indicated the possibility that the roll torque was due to the expanding GOX impinging on the Pegasus wings in an unsymmetrical fashion causing the roll.

There wasn't sufficient time prior to the next flight to make a major change, therefore a "short cut" solution was desired. Analysis of the SA-9 results indicated that 39,834 lbs.-sec. of GH_2 and 38,576 lbs.-sec. of GOX total impulse remained after cutoff to be vented. However, approximately 30% of the GH_2 total impulse was expended by the blowdown system before the Pegasus wings were extended. Therefore, the GOX and GH_2 non-propulsive vents were interchanged for SA-8 and SA-10, hoping to get a 30% reduction in the roll rate. The flights of SA-8 and SA-10 proved the concept, with the maximum roll rates being kept to acceptable levels of just slightly over 6 deg/sec.

Thermal Environment

A significant amount of instrumentation was carried on the Saturn I vehicles to measure pressure, acoustic and thermal environments on many areas of the vehicle. Pressure data was required to verify air loads analyses and structural design. Thermal information was required to verify such things as heat protection material used, determine heat input into the propellants during flight, and to determine heating effects caused by protuberances, etc.

One area of intensive analysis was the heat input to the S-I stage base due to the heating from the engine exhausts and the complex flow expected in this region. Thermal instrumentation consisted of total and radiation calorimeters and gas temperature probes. Total calorimeters were, in general, the slug type with a high value of thermal conductivity. The slug had a blackened surface finish with an emissivity factor close to one and was isolated from the surrounding structure to minimize heat conduction losses. A thermocouple was mounted to the rear of the slug to measure its temperature. Heating rates were determined by taking the time rate of change of temperature and correcting for heat losses from conduction and re-radiation. The radiation calorimeters consisted of a copper slug with a blackened surface which was enclosed behind a sapphire window to isolate the convective flow from the slug surface. A purge of nitrogen gas around the window prevented clouding over by the carbon particles circulating in the base region.

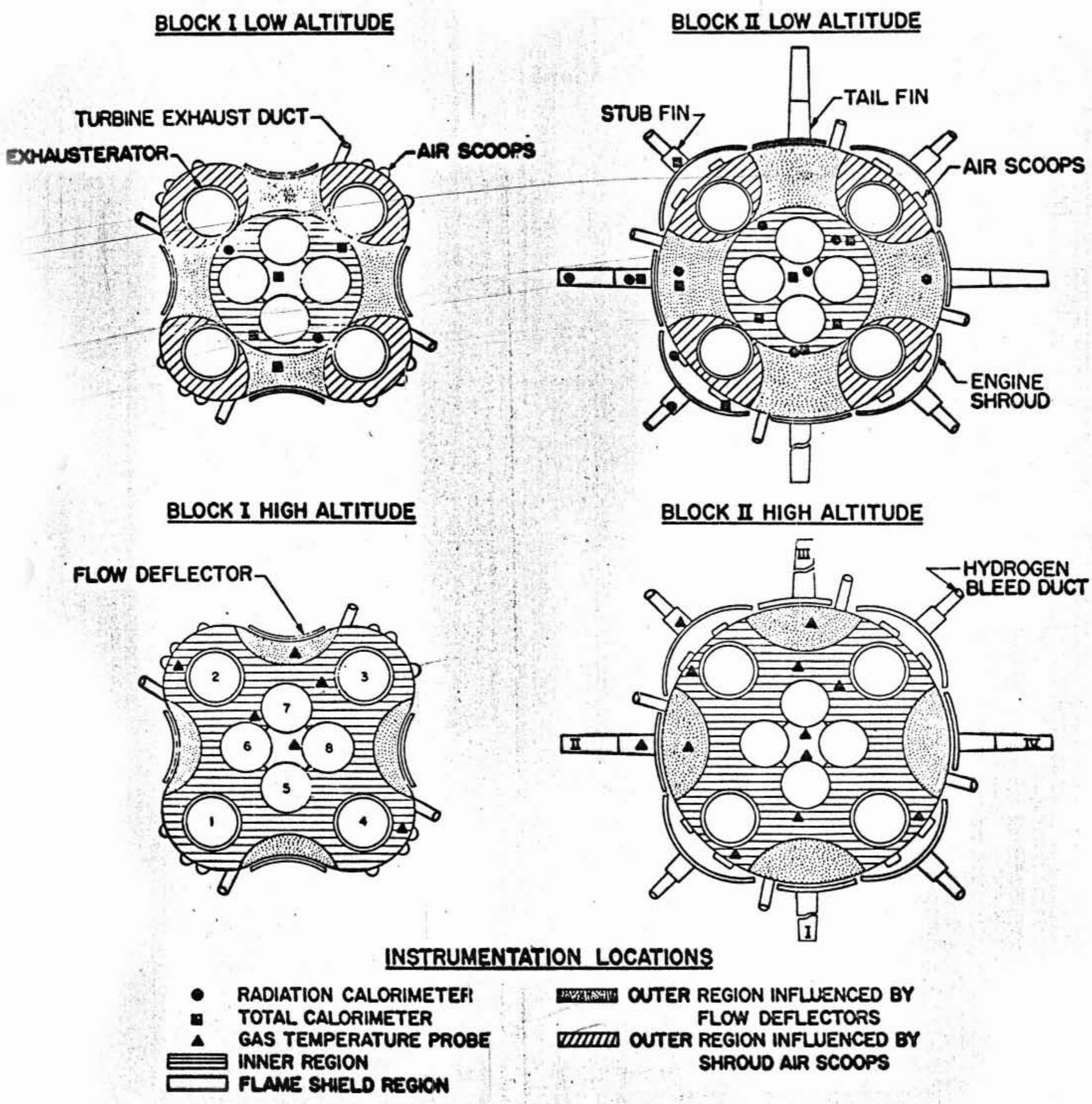


FIGURE 15 S-I STAGE BASE HEATING REGIONS

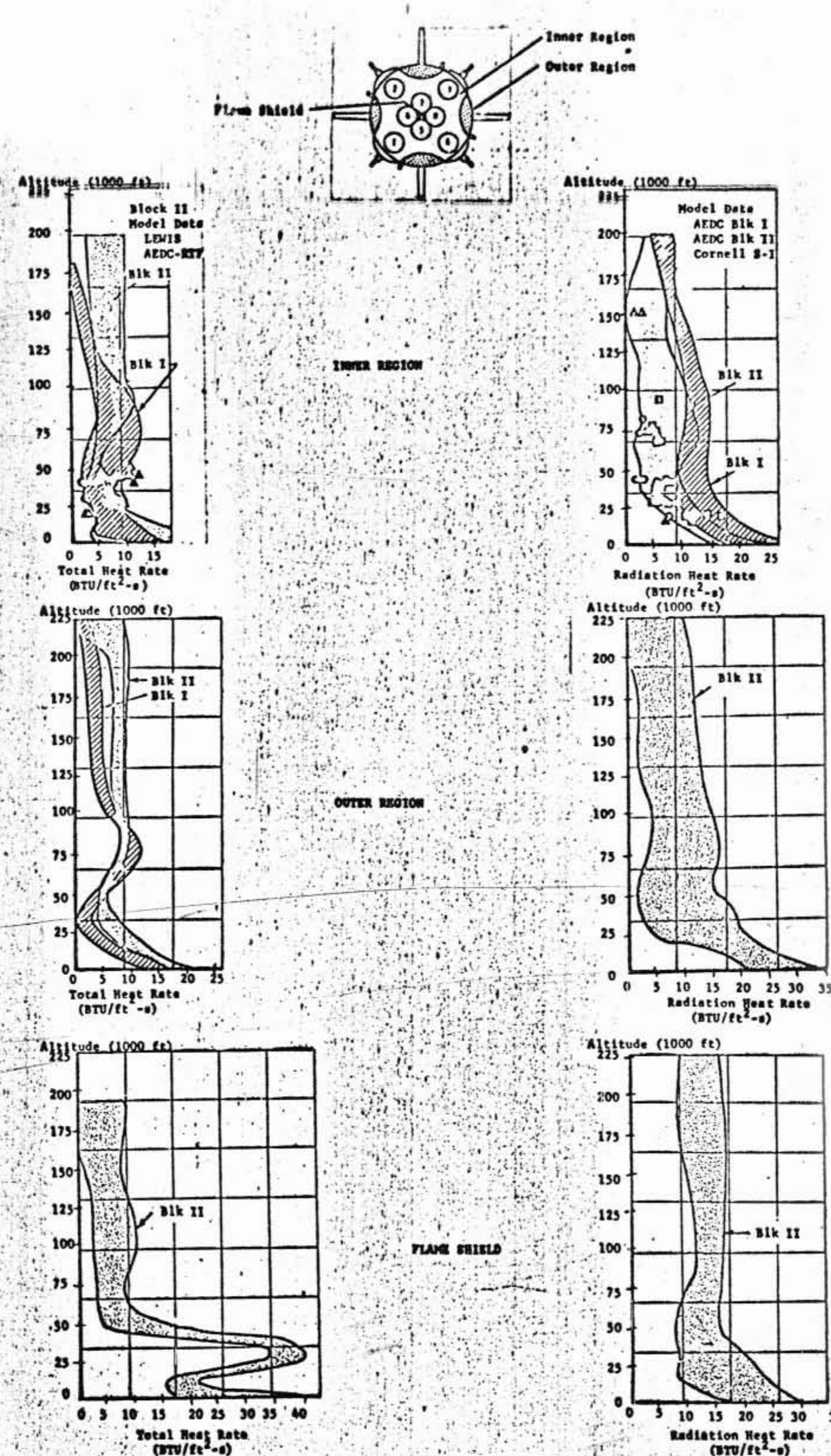


FIGURE 16 S-I STAGE BASE HEATING RATES

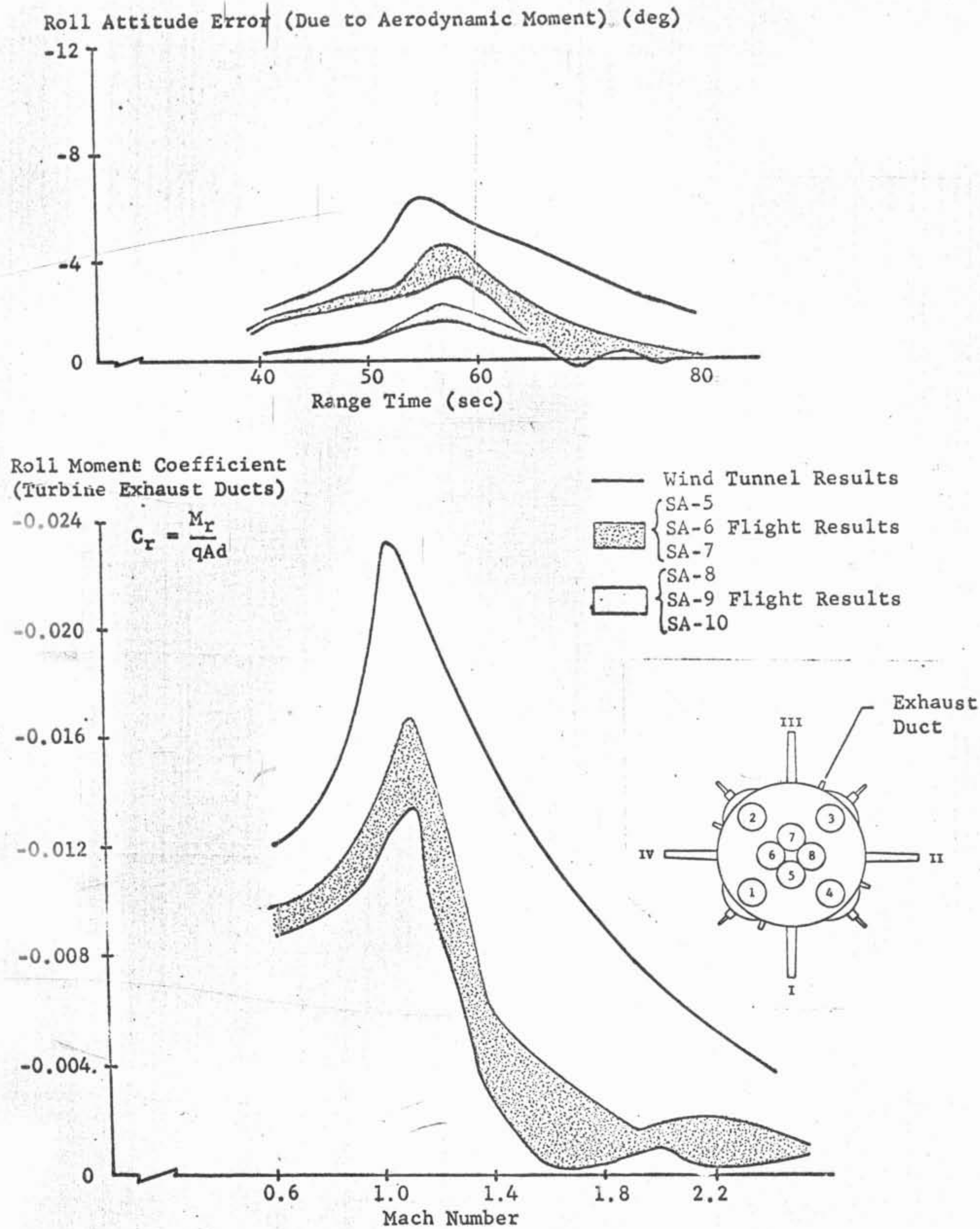


FIGURE 17 ROLL MOMENTS OF BLOCK II VEHICLES

Regions in the base area which experienced similar environments in flight can be sub-divided into five major areas: inner region, outer region, engine shroud, flame shield and fin base. These are shown in Figure 15.

Figure 16 presents a summary of the results obtained from the flights in three areas: the heat shield inner and outer regions and the flame shield. High heat rates were encountered during liftoff due to high radiation levels induced by engine exhaust impingement on the launch complex flow deflector. Following the liftoff phase, when normal low altitude radiation from the exhaust plumes was incurred, the total heating rates on the heat shield dropped to a level of 8 BTU/ft^2 sec. or below. At an altitude of approximately 30,000 to 35,000 feet the total heating rates began to increase as the reverse flow from the inboard engines produced a convective flow in the base region. Maximum heating occurred between 60,000 and 90,000 feet. Above this altitude, the rapid expansion of the exhaust plumes caused a cooling of the exhaust gases and a subsequent reduction in radiation and total heat rates on the heat shield.

Radiation heating rates to the outer region were slightly higher than for the inner region as expected. Since the total heating rates for both regions were similar, this would indicate that slightly more convective cooling was present in the outer region.

Aerodynamics

An interesting flow phenomena was discovered from the flights of the Block II vehicles. On the first flight, SA-5, the vehicle rolled in an unexpected, but systematic fashion, to a trim condition which reached a maximum roll angle of slightly more than 3 degrees at 56 seconds of flight. It was finally theorized that this may have been due to the slight assymmetry of the base configuration (Figure 17) resulting from the locations of the turbine exhaust ducts. After the flight of SA-5, wind tunnel tests were run. These tests with a complete model verified the trend, but over-predicted the magnitude. Removal of the turbine exhaust ducts from the model reduced the roll moment to zero verifying that these were in fact causing the moment.

The magnitude of the roll moment in flight was calculated to be on the order of 65,000 ft-pounds. The effect was repeated on all subsequent flights. No truly adverse effects were caused by this moment. However, the roll control gain was increased on SA-8, -9 and -10 which caused the trim angle to be reduced by about 50%.

Vehicle Modifications

While the Saturn I program was outstanding by its success there were still changes required during the flight test phase. Some of the changes are summarized in Figures 18 and 19. You will note two categories; pre-planned modifications and unexpected modifications. The later are consequences of post-flight evaluation. In addition to hardware changes is the increased knowledge of the vehicle systems performance capabilities obtained from flight testing. This is indicative of the job that remains even with a basic design concept that proved as sound as the Saturn I.

FIGURE 18

SATURN I SYSTEMS MODIFICATIONS DURING TESTING

Pre-Planned Modifications

Unmodified Vehicles	Area to be Modified	Modified Vehicles	Modification
SA-1 & SA-2	S-I Propellant Utilization	SA-3 & SA-4	LOX Depletion (S-I Stage)
SA-1 thru SA-3	Angle-of-Attack Control	SA-4 & Subs	Accelerometer Control
SA-1 thru SA-5	Electrically Differentiated Attitude Rates	SA-6 & Subs	Rate Gyro Control
SA-1 thru SA-6	Mark III Gear Case	SA-7 & Subs	Mark III H (heavy duty) Gear Case
SA-5 & SA-6	Non-Utilization of S-IV Backup LOX Tank Pressurization System	SA-7 & Subs	Backup bottles eliminated
SA-6 & SA-7	Path Adaptive Guidance	SA-9 & Subs	Iterative Guidance Mode

FIGURE 19

UNEXPECTED MODIFICATIONS

Vehicle Affected	Area or System Affected	Modified Vehicles	Modifications
SA-1	Sloshing instability	SA-2 & Subs	Addition of Baffles
SA-2	Turbine erosion due to a long LOX lead time	SA-3 & Subs	Link valves installed to prevent LOX lead
SA-2	LOX emergency vent valve opened during flight	SA-5 & Subs	Reduced number of active coils in heat exchanger
SA-1 thru SA-4	Roll moment	SA-5 & Subs	Different type of S-I actuators used
SA-2	Periodic tilt disturbance	SA-3, SA-4 SA-5 & SA-6	Reduced tilt cam loading
SA-3 & SA-4	Bending oscillations after CO	SA-5 & Subs	Changed control gains
SA-3	Open circuit in guidance signal processor	SA-4	Redesigned module packaging
SA-3 thru SA-10	RF attenuation during retro/ullage rocket firing	SA-4 & Subs	Utilized playback TM recorder
SA-5	Obscurred lens of onboard movie cameras	SA-6 & SA-7	Air purge over lenses
SA-5	Null shifts present in two S-IV servo actuators (Fluid contamination)	SA-6 & Subs	Different type of S-IV actuators used
SA-5 & SA-6	Orbital attitude perturbation	SA-7 & Subs	Initiated Non-Propulsive Vent System
SA-5 & SA-6	S-IV stage TM vibrations	SA-7 & Subs	S-IV TM shock mounted
SA-5 thru SA-10	Aerodynamic Roll Moment	SA-7 & Subs	Roll gain changed
SA-6	Ullage rocket failed to jettison	SA-7 & Subs	Elbow fitting replaced with a "T" fitting
SA-6	S-I propellant loading irregularity	SA-7 & Subs	Improved loading instructions
SA-7	S-IV LH_2 venting	SA-8, SA-9, & SA-10	Auxiliary Non-Propulsive Vent System
SA-7	Platform alignment affected by vibrations after S-I ignition	SA-8, SA-9, & SA-10	Eliminated pendulum alignment from loop prior to S-I ignition
SA-9	GOX impingement on Pegasus wings	SA-8 & SA-10	LH_2 vent and GOX vent interchanged