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FLIGHT TEST PLAN

PROJECT SOL RAD SCOUT

Report No. 3-13839/2R-2

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ABSTRACT

Flight vehicle configuration, detailed test objectives, the required data measurements, the predicted trajectory data, and the flight test plans for the first SOL RAD SCOUT launch are presented in this report. NASA/DOD Scout Research Vehicle S-111 will inject the Solar Radiation Satellite (SR-5) into a nominal 390 nautical mile circular orbit at a nominal inclination of 75.4 degrees where measurements of solar X-rays and the Lyman Alpha effect will be made. The vehicle will be launched from the Scout Pad at Point Arguello, Pacific Missile Range, by the Blue Scout Branch of the 6595th Aerospace Test Wing.

SUMMARY

The ~~NASA/DOD Scout Research Vehicle (SR 5) Solar Radiation Satellite~~ Solar Radiation 4B Satellite is scheduled to be launched on a South~~eastern~~western flight azimuth from the Scout Pad of the Probe Launch Complex at Point Arguello, Pacific Missile Range. NASA/DOD Scout Vehicle S-111 will be used to place the SOL RAD SATELLITE into orbit. This booster is basically a four stage, solid propellant vehicle comprised of four rocket motors joined by transition sections and supported by a base section. The first stage is an Algol Senior, developed by Aerojet General Corporation. Control of this stage is accomplished by jet vanes and aerodynamic fin tips located in the base section around the Algol Senior nozzle. The second stage is a Castor XM-33-E5, developed by Thiokol. Control during second stage burning is accomplished by a Hydrogen Peroxide reaction system located in the upper "B" transition section surrounding the second stage nozzle. Third stage is an Allegany Ballistic Laboratory motor, Antares X-259A1. Control is provided by a hydrogen peroxide reaction system located in upper "C" transition section which is similar to the second stage control systems. The fourth stage, an Allegany Ballistic Laboratory Altair X-248-A5S, will be used to inject the SOL RAD payload into orbit. The fourth stage and payload derive their stability from being spin stabalized by tangential spin-up rockets prior to fourth stage ignition.

The vehicle loads are carried by the following components: Base section, first stage motor case, lower "B" transition section, second stage nozzle, second stage motor case, lower "C" and upper "C" transition sections, third stage motor case, lower "D" and upper "D" transition section, fourth stage motor case, satellite support ring and payload.

Control steering commands for the first three stages are provided by a strapped-down three-axis minature **integrating** Gyro system for altitude control and another three-axis rate Gyro system for rate control. Programming is done in the pitch

plane only while yaw and roll are held [illegible]

Radar Tracking of the Scout vehicle to the point of fourth stage separation will be accomplished by utilizing a C-Band Radar Beacon, Avion model 149C. The vehicle telemetry system is a PAM/FM/FM type with fifteen channels, two of which are commutated.

The NASA/DOD Scout will place the SOL RAD payload into a nominal 490 Nautical Mile circular orbit. The orbit will have an inclination of approximately 75.4 degrees. The SOL RAD SATELLITE, developed by the Naval Research Laboratory, will obtain measurements of solar X-rays in four bands and measurements of the Lyman Alpha air glow from the Northern sky.

Powered flight data acquired by the PMR will be used by AFSSD and the associated contractors for evaluating the performance of the launch vehicle.

Operation of the satellite systems and collection of data from them will be accomplished by the NRL.

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1. INTRODUCTION

1.1 Scope

This document presents the flight test plan and mission objectives for the Solar Radiation HD Satellite and the NASA/DOD Scout Research Vehicle S-111 [redacted] This vehicle will be launched from the Scout Pad at Point Arguello, Pacific Missile Range, California. The flight trajectory and instrumentation required to meet the objectives of the launch are specified in this report. Items such as evaluation parameters, special or unique operating procedures, and other information necessary to define the flight test are also given.

1.2 Purpose

In addition to providing a comprehensive account of the flight test, the specific objectives of this FTP are:

a) To specify mission objectives.

b) To provide information on this flight test to supplement the general data and support requirements previously defined for the complete program.

c) To designate that information required for flight test evaluation.

d) To serve as the primary flight test document for all contractors and agencies concerned with the launch.

Objectives (a) and (b) are considered the primary objectives and are concerned specifically with the launch of Scout vehicle S-111 whereas the latter two objectives, under which information peculiar to the subject operation is also provided, are considered secondary. The secondary objectives are treated somewhat more extensively in this report than they will be in future SOL RAD SCOUT FTP's where only the changes in the secondary objectives will be described. It is anticipated that this form of presentation will result in this report serving, in addition to its primary use, as a general reference for future SOL RAD SCOUT operations.

Since this document is submitted well in advance of the actual operation, it was necessary to describe a nominal vehicle. Rather than submit the nominal trajectory which appears in the SSVT-24 PRD, the final trajectory which is based on actual vehicle weights and predicted engine performance will be forwarded as Appendix B to the report, approximately 15 days prior to the scheduled launch date. A detailed tabulation of the final trajectory will be forwarded at this time as an appendix to the Operations Requirements. Also presented with this Appendix will be the actual pitch program, control system frequency tolerances, and required control system settings.

1.3 Test Responsibilities

A general outline of responsibilities assigned to the groups participating in this project is given below:

a) Transit/Anna Directorate AFSSD
   Responsibility - Vehicle Program Direction

b) Standard Launch Vehicle I Directorate, AFSSD
   Responsibility - Booster and Facilities

c) Naval Research Laboratory, Satellite Techniques Branch
   Responsibility - SOL RAD Satellite

d) Director of Plans, 1st STRATAD (SAC)
   Responsibility - Base Support Programming and Policies

e) Commander CSD #1 WCMR, VAFB
   Responsibility - Contract Administration and Management at VAFB

f) Director Civil Eng., 6595th Test Wing
   Responsibility - Test and Support Facilities

g) Technical Support Office, 6595th Test Wing

Responsibility - Communications, Instrumentation, Safety, Maintenance, Transportation and Supply Support at VAFB.

h) Blue Scout Branch, 6595th Aerospace Test Wing

Responsibility - Program Support

i) Blue Scout Military Launch and Test, 6595th Aerospace Test Wing

Responsibility - Vehicle Assembly, Checkout and Launch

j) Chance Vought Astronautics

Responsibility -systems engineering and technical assistance for all phases of launch vehicle preparation

1.4 Applicable Documents

Requirements, detailed procedures, and general test material relating to this launch are presented in several documents, some of which remain to be published. Section 7 of this report covers these documents in detail. Documents of primary concern are:

a) Program Requirements Document
b) Operations Requirements
c) Operations Directive
d) Launch Test Directive
e) Preliminary Countdown Document
f) Range Safety Report
g) PMR Ground Safety Report

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2. FLIGHT VEHICLE CONFIGURATION

2.1 General

The Scout Research Vehicle is basically a four-stage, solid propellant, booster system which features low cost of manufacture and launch operations and flexibility of application. For certain types of mission profiles, a fifth stage may be added in the payload area. The three basic missions to which the vehicle may be successfully adapted are: *More realism here please. include inclination*
(1) satellite launcher (e.g., 150-lb. object in a 300 nautical mile orbit):
(2) high altitude probe (e.g., 90-lb. object to approximately 4,000 nautical miles):
(3) high velocity re-entry tests.

The basic airframe configuration is comprised of the four rocket motors tied together by transition sections and supported by a base section. The third and fourth stage motors and the payload are protected from aerodynamic heating. Principal fairings consist of two wiring tunnels from the base section to the uppermost transition section. The major assemblies are identified as follows:

Base Section "A"
Algol - 1D rocket motor (Aerojet Senior)
Transition Section "B" (lower and upper sections)
Castor rocket motor (Thiokol XM-33-E5)
Transition Section "C" (lower and upper sections)
Antares rocket motor (Alleghany Ballistics X259-A1)
Transition Section "D" (lower and upper sections)
Altair rocket motor (Alleghany Ballistics X-248-A5S)
Payload Assembly

Figure 1 is a photograph of Scout Research Vehicle ST-5 which had essentially the same configurations as S-111. A general arrangement of the subject vehicle is shown in Figure 2 and an outboard profile is displayed on Figure 3. Figure 4 is a general arrangement of the SOL RAD payload.

2.2 Major Assemblies, Booster System

Major assemblies of the vehicle are defined as the base section, the three transition sections, the heat shield, and the rocket motors. The following brief description of these units is intended to acquaint the reader with the purpose and intrinsic features of each. The subsystems that are identified as being contained in or incorpora ed on these assemblies and whose function is vehicle control or communication are described in detail in Section 4.

2.2.1 Base Section "A"

This section forms the aft portion of the first stage and contains the four cruciform fins, jet vanes, and associated components of the first stage hydraulic control system. Also, contained in this assembly are the ignition system motion switches and the guidance system umbilical plug. The airframe is constructed of a semi-monocoque aluminum shell supported with steeland aluminum ring frames, longerons and a bulkhead installation at the nozzle exit. Attachment takes place at the aft bolting ring of the ALGOL ID motor.

The fins are mounted on the aft portion of Base Section "A", 90 degrees apart and at zero angle of incidence. The control surfaces, steel fin tips and molybdenum jet vanes are connected by a common shaft which is supported by the fin structure.

Three large hinged access doors are incorporated in this section. On these doors are mounted the various subsystem components in a manner that permits complete checkout and installation of the section on the Algol motor without disturbing any of the subsystem components. This arrangement further provides excellent accessibility to the components during operation of the various subsystems. For this discussion, the doors will be identified as Doors II, III and IV. When looking forward with the Tower Fin at the 12 o'clock positon, Door II is in the upper left quadrant; Door III is in the lower left quadrant; and Door IV is located in the lower right quadrant.

Equipment mounted on these doors are as follows:

Door II	Hydraulic reservoir, accumulator, and pressure switch.
Door III	Electro-hydraulic motor pump, filter, and external electric and hydraulic power connectors.
Door IV	First stage control system power supply and two associated amplifiers.

2.2.2 First Stage Rocket Motor

The Algol Rocket motor (Aerojet Senior), Model 1D, was first installed in vehicle number S-110. Improvements over the model 1C motor include a lighter weight nozzle and the use of tap pads instead of studs for installation of the wiring tunnels. Nominal physical characteristics and performance of the motor are presented in Table2-2. Dual explosive linear shaped charges which can accomplish first stage thrust termination are mounted longitudinally along the motor case.

2.2.3 Transition "B"

This section is the transition structure between the first stage and second stage motors. It consists of an upper and lower section separated by a blowout diaphragm which ruptures or deforms when the second stage motor ignites. The lower section which remains with the first stage upon separation, is an aluminum semi-monocoque structure with attaching rings at the end. It is connected to the head attachment ring of the first stage motor. The Auto destruct components for the first stage motor are located in this section. The upper section consists of split halves of a non-structural glass fiber laminate shell reinforced with internal aluminum frames. The two halves are frame spliced from the outside with a minimum number of attachments along the skin splice, reducing the complexity of the assembly procedure. Accessibility to the second stage reaction control system components which are located within this section is provided by access doors. The primary load carrying member through this section is the second stage nozzle.

2.2.4 Second Stage Rocket Motor

The Castor rocket motors (Thiokol XM-33-E5) is the same motor as was installed on previous NASA Scout vehicles except that tap pads are used in place of studs for wiring tunnel attachments. Explosive charges, similar to the units found on the first stage motor, are incorporated on the Castor motor case. Table 2-2 describes the nominal Castor motor.

2.2.5 Transition "C"

This section consists of two monocoque shells joining the second

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and third stage motors through a blowout diaphragm. All structural loads are carried in the glass fiber laminated outer shell. The assembly for S-111 incorporates two configuration changes which will also be effective on subsequent vehicles. These changes are structural beef-ups to make the section compatible with the new third stage motor and installation of the ignition destruct batteries in upper "C" section. (previously installed in "D" section). Other items contained in Upper "C" are the third stage reaction control system components, guidance system components, and Ignition/Destruct components.

2.2.6 Third Stage Rocket Motor

Third stage propulsion is provided by the Antares rocket motor (ABL X-259-A1). NASA Scout number S-110 first utilized this particular motor which has several improvements over the previously installed third stage motor. These changes are; a motor case of increased structural strength, the use of Thermolag (an ablative coating) in place of a heat shield for thermal protection, head insertable igniters in place of tail insertable, and a wiring tunnel attachment arrangement similar to the first two stages. Thrust termination is accomplished by the linear shaped charges located along the motor case. Nominal dimensions and characteristics are given in Table 2-2.

2.2.7 Transition "D"

The third and fourth stage motors are joined by this transition structure, which is comprised of an upper and lower section. The lower section is a steel structure supported by longerons and end rings. This section includes a spin table and four spin motors forward of the spin table. Also contained within this section are components of the telemetry system, the guidance system, the radar beacon system, and the ignition components. The upper section is a magnesium structure which transmits spin loads which originates in the portion of lower "D" section forward of the spin table. The upper and lower sections are joined by a ring clamp which is secured by four explosive bolts. Separation of the fourth stage and payload is accomplished when the explosive bolts are fired by a command from the guidance system timer. A smooth separation is insured by 32 miniture springs which are normally compressed within the separation plane. One and one-half seconds after the explosive bolts are activated the third stage pitch and yaw reaction control motors are fired, thus imposing a deceleration to the third stage due to the 30° forward cant of the motors. Fourth stage ignition, accomplished by a delay squib, occurs after above described separation events.

Configuration changes which are effective on vehicle S-111 and subsequent include; structural beef-up of the entire section, access door for installation of side insertable squibs for the third stage motor, and component re-arrangement and shelf redesign to allow access to third stage motor igniter.

2.2.8 Fourth Stage Rocket Motor

The fourth stage and payload are accelerated to injection velocity by the Altair rocket motor (ABL X-248-A5S). A nominal Altair motor is described in Table 2-2.

TABLE 2-1

SOL RAD SCOUT NOMINAL WEIGHT DATA

Payload	110.00	See Note 6
Fourth Stage Inert	64.50	
Fourth Stage Burnout	174.50	
Fourth Stage Consumed	466.00	
Fourth Stage Ignition	640.50	
Third Stage Inert	789.30	See Note 5
Third Stage Burnout	1429.80	
Third Stage Consumed	2626.00	See Note 4
Third Stage Ignition	4055.80	
Second Stage Inert	2146.00	See Note 3
Second Stage Burnout	6201.80	
Second Stage Consumed	7549.00	See Note 2
Second Stage Ignition	13750.80	
First Stage Inert	4068.30	See Note 1
First Stage Burnout	17819.10	
First Stage Consumed	19200.00	
First Stage Ignition	37019.10	

Notes	1	3.00 #	Thermolag (Antares)
		4065.30 #	Motor Case and Hardware
	2	7455.00 #	Propellant Consumed (Castor)
		90.00 #	H_2 O_2 consumed
		4.00 #	Thermosorb (Castor Nozzle)
	3	1974.00 #	Motor Case and Hardware
		170.00 #	4th Stage Payload Heat Shield
		2.00 #	Thermolag (Antares)
	4	2616.00 #	Propellant consumed (Antares)
		10.00 #	H_2 O_2 consumed
	5	786.30 #	Motor Case and Hardware
		2.00 #	Thermolag (Antares)
		1.00 #	Spin-Motor Propellant consumed
	6	100.00 #	Payload
		10.00 #	Separation Mechanism

TABLE 2-2

TYPICAL NASA/DOD SCOUT MOTOR SPECIFICATIONS

	ALGOL	CASTOR	ANTARES	ALTAIR
Length (feet)	29.98	20.32	10.16	12.18
Diameter (feet)	3.38	2.6	2.5	1.8
Weight - Empty (Pounds)	4068.30	2145.96	789.35	65.00
Weight - Propellant (Pounds)	19200.00	7338.00	2526.00	465.00
Weight - At Ignition (Pounds)	23268.30	9483.96	3315.35	530.00
Weight - At Burnout (Pounds)	4068.30	2028.96	699.35	64.00
Specific Impulse	238.15 VAC	262.91 VAC	276.70 VAC	251.27 VAC
Thrust (Pounds)	110712 VAC	49123 VAC	18097 VAC	2822 VAC
Burning Time	41.3	39.9	40.0	41.4

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3. TEST PLAN

3.1 Preflight Tests

Sub-system tests, calibration checks, functional tests, and combined systems test will be accomplished in the missile assembly building. During the latter portion of the countdown, while the vehicle is being assembled in the OAB, mechanical and electrical tests are performed on the launcher. At this same time operational test s on the blockhouse equipment are accomplished. After the vehicle is moved to the launch pad, RF functional tests and compatability checks will be performed. A "dry-run" countdown will be scheduled prior to launch day.

3.2 Launch Conditions

The limiting conditions for launch are primarily wind force limitations.

3.2.1 Launch Time

The present launch date is 4 April 1962. Liftoff time will be scheduled for , with an allowable tolerance of 4 hours.

3.2.2 Wind Force

3.2.2.1 Ground Conditions

The maximum allowable ground winds for the vehicle on the launcher is 32 ft/sec. while in the launch position. Should the winds exceed the above limit, the vehicle should be recovered, lowered, and secured to the transporter.

3.2.2.2 Altitude Conditions (Preliminary)

The limiting altitude conditions are direct tail winds with velocities of 160 ft/sec. at 27,000 feet and 200 ft/sec. at 35,000 feet. At these velocities the probability is 0.01 that the allowable bending moment will be exceeded.

Figure   is a graphic presentation of probability versus peak wind speed. This graph and a refinement of the above numbers will be submitted along with final trajectory data, approximately 15 days prior to the scheduled launch date.

3.3 Flight Test

The missile will be launched in a vertical position from the Scout Pad of the Probe Launch Complex at Point Arguello, Pacific Missile Range. The proposed flight azimuth will be 164 degrees. An airborne timer which initiates the pitch program for the vehicle will be activated approximately 0.21 seconds after ignition. Any deviation from the planned flight path will be sensed by the guidance unit (basically a precision auto pilot) and commands will be sent to the control elements to realign the vehicle so as to null the error signal. Pitch programming is accomplished from 1.0 second after first stage ignition until fourth stage separation. The commands result in the vehicle following a gravity turn trajectory.

The first stage will burn out after approximately 41 seconds and the vehicle will enter its first coast phase, the jet vanes and/or aerodynamic fin tips give the controlling moments for guidance through this portion of the flight. At T + 72 seconds, second stage will ignite, separating from the first stage motor at the blowout diaphram, and burn for a nominal 40 seconds. At this time, the vehicle enters its second coast period. The hydrogen peroxide system located in the upper "B" transition section is used for control during second stage burning and this second coast period. Third stage ignition occurs at T + 117 sec and the motor will burn for 40 seconds with burnout at T + 157 sec. Separation of second and third stages is also accomplished by a blowout diaphram and a smaller hydrogen peroxide system in upper "C" section provides third stage control. When the third stage burns out at T + 157 seconds, the vehicle enters its third and longest coast period, 442 seconds, and will coast up to orbital altitude with the third stage still attached. The fourth stage is ignited at T + 599 sec and injects the fourth stage and payload into orbit with burnout at T + 640. Stability for this stage is provided by the fourth stage being spun up to 180 rpm just prior to ignition. Separation of the fourth stage is achieved when the ring clamp is released just after spin up. After fourth stage motor burnout the payload is separated by a timer firing explosive nuts. This releases the spring loaded payload from the payload separation ring mounted on the fourth stage.

A ground track of the first pass of this vehicle is shown in Figure . Table 3-1 presents a list of flight events from liftoff of the vehicle to satellite separation.

3.4. Powered Flight Range Safety

The nominal boost trajectory for the SOL RAD mission does not require the vehicle to overfly any inhabited land masses during ascent to orbital altitude.

Stage one, two, and three expended booster cases all impact in open water areas. Stage four expended case has sufficient velocity at burnout to go into orbit with the payload. Nominal impact point locations are °N and °W, N and W, and N and W for the first, second, and third stage booster cases respectively.

Further Range Safety information can be found in Reference ( ).

TABLE 3-1

NOMINAL SOL RAD SCOUT SEQUENCE OF EVENTS
COMMANDED PITCH RATE

Time (Seconds)	Events	Pitch Rate Command Deg./Sec.	How Accomplished
0.00	Stage 1 Ignition	$\dot{\theta}_C = 0.00$	Blockhouse Command
0.21	Start Timer		Fly-Away
	Remove Ignition Holdouts		Lanyard
1.00	Pitch Program	$\dot{\theta}_{C_1} = -4.7383$	Timer Function 1
4.50	Pitch Program	$\dot{\theta}_{C_2} = -0.6159$	Timer Function 2
27.50	Pitch Program	$\dot{\theta}_{C_3} = -0.3133$	Timer Function 3
41.30	Stage 1 Burnout		
54.00	Pitch Program	$\dot{\theta}_{C_4} = -0.4262$	Timer Function 4
72.25	Stage 2 Ignition		Timer Function 5
	Separate 1st Stage		2nd Stage Ignition
	Activate "B" Controls		Timer Function 5
	Remove 1st Stage Controls		1st Stage Separation
80.00	Pitch Program	$\dot{\theta}_{C_5} = -0.2893$	Timer Function 6
94.00	Pitch Program	$\dot{\theta}_{C_6} = -0.1726$	Timer Function 7
112.01	Stage 2 Burnout		
	Separate Payload Heat Shield		Timer Function 8
	Activate "C" Burn Controls		Timer Function 8
	3rd Stage Squib Ignition		Timer Function 8
117.02	Stage 3 Ignition		Ignition Squib Delay
	Separate 2nd Stage		3rd Stage Ignition
	Remove 2nd Stage Controls		2nd Stage Separation
133.00	Pitch Program	$\dot{\theta}_{C_7} = -0.1040$	Timer Function 9
157.02	Stage 3 Burnout		
	Activate "C" Coast Controls		Timer Function 10
167.00	Pitch Program	$\dot{\theta}_{C_8} = -0.1000$	Timer Function 11
210.00	Pitch Program	$\dot{\theta}_{C_9} = -0.0000$	Timer Function 12

TABLE 3-1 (Continued)

NOMINAL SOL RAD SCOUT SEQUENCE OF EVENTS
COMMANDED PITCH RATE

Time (Seconds)	Events	Pitch Rate Command Deg./Sec.	How Accomplished
593.0	4th Stage Spin Motor Ignition.		Timer Function 13
	4th Stage Ignition delay Squibb Ignition. (six sec. delay)		Timer Function 13
594.5	Separation bolt ignition		Timer Function 14
	Separate 4th Stage		
595.5	Fire "C" Controls for Retro		Timer Function 15
599.0	4th Stage Ignition		
640.42	Stage 4 Burnout (Injection)		
	Payload Separation		Payload Separation Timer

4. FLIGHT VEHICLE GUIDANCE AND INSTRUMENTATION

4.0 General

Vehicle guidance and control is accomplished by a three axis reference package located in the Lower D Transition section and by separate control systems in each of the first three stages. All guidance commands originate from the internal system after the guidance system intervalometer is activated when a holding voltage is removed at separation of the flyaway umbilical.

The Instrumentation System of the NASA/DOD Scout Research Vehicle is defined as all of the instrumentation required to monitor the performance of the first three stages of the Scout vehicle and to provide for radar tracking and command-destruct functions. The system is comprised of the following four subsystems: (1) a fifteen (15) channel telemetry system; (2) a radar beacon; (3) a command-destruct system; and (4) all the cabling, recaps, transducers and other equipment required for ground control and monitoring. A summary of the operating frequencies is given in Table 4-1.

The development, design, and fabrication of the Instrumentation System incorporates standard materials and components that have exhibited high flight reliability.

The SOL RAD payload instrumentation consists of a transponder telemetry system which transmits solar X-rays and Lyman Alpha data acquired during the orbit.

4.1 Guidance and Control Systems

The guidance and control system provides three axes of attitude reference and control. Azimuth and Roll orientations are referenced to the launch attitude of the vehicle and maintained throughout the flight. The pitch axis is initially referenced to the launch attitude, however, the pitch reference is changed throughout the flight by action of the pitch programmer. Any deviation from the reference orientation is sensed in the guidance system and an appropriate correction commanded by the control system. All controling signals arise from airborne equipment. The system is completely adaptable to the three basic flight trajectories. For the

TABLE 4-1

VEHICLE FREQUENCY SUMMARY

NASA/DOD SCOUT (All Stages)	FREQUENCY (mc/sec)
1. Command Destruct	416 (C)
2. PAM/FM/FM Telemetry Transmitter	249.9
3. Avion 149 C Transponder	5491 mc (Interrogation) 5566 mc (Response)
SOL RAD SATELLITE	
1. Payload Telemetry Transponder	122.9 mc (Interrogation) 136.890 mc (Response)

purpose of this discussion the Guidance portion of the system is defined as those electronic components contained in the system and the Control portion is restricted to the final control assemblies, i.e., reactions thrust systems, aerodynamic surfaces, and jet vanes.

4.2.1 Guidance Subsystem

Attitude reference for the guidance subsystem is provided by a guidance unit in which three miniature integrating gyros (MIG's) are secured directly (strapped down) to the vehicle airframe. The guidance unit also contains the necessary electronics to operate the gyros and process the error signals. A pitch programer, consisting of a multi-tap, regulated D. C power supply, is used in conjunction with an intervalometer to supply voltage steps to the torques of the pitch MIG gyro. These voltages and the intervalometer times are preset to values which produce an attitude reference change in the pitch gyro, resulting in a zero "G" trajectory. The intervalometer also provides precise scheduling of other flight events such as ignition and heat shield ejection. A rate gyro unit containing three miniature rate gyros is included to provide stability and damping in the system. Due to vehicle body bending the rate gyros have been located in Upper "C" transition section.

The error signals produced by the guidance unit are transmitted to an electronic switching unit, which control the valves of the reaction control systems and produces a proportional D.C signal to operate the first stage servo system. Two battery power supplies and a solid state 400 cycle inverter are used to supply the necessary inflight power requirements of the guidance system. A relay box is included to accomplish power and ignition switching.

The components of the Guidance subsystem are located as follows:

Guidance Reference Unit-Lower "D" Transition Section

Program Power Supply-Lower "D" Transition Section

Poppet Valve Amplifier Demodulator-Lower "D" Transition Section

Intervalometer-Lower "D" Transition Section

Power Control Relay Box-Lower "D" Transition Section

Single Phase Inverter-Lower "D" Transition Section

Power Supply 28 VDC-Lower "D" Transition Section

Power Supply 37 VDC-Lower "D" Transition Section

Rate gyro Unit-Upper "D" Transition Section

Servo Amplifier (2)-Base "A" Transition Section

4.1.2 Control Subsystem

4.1.2.1 First Stage

Control during the period of first stage burning is provided by a combination of interconnected jet reaction vanes and aerodynamic fin tip control surfaces. During the ensuing coast period, the tip surfaces provide all of the control forces. Proportional signals from the guidance package and resulting proportional surface deflections characterize this system. The four sets of control surfaces operate in pairs to control the vehicle. The side surfaces move together to control the pitch attitude of the vehicle and the upper and lower surfaces move together to control the yaw angle. Roll attitude control is accomplished by differential deflections of the upper and lower surfaces.

4.1.2.2 Second and Third Stage

Stabilization and control of the second and third stages is accomplished with reaction motors. These motors are fueled with 90%

Hydrogen Peroxide pressurized with a high pressure nitrogen charge. Eight motors are located in upper "B" transition section to control the second stage. The motors are arranged as shown in Figure . The pitch and yaw motors fire singularly and the roll motors fire in pairs to achieve vehicle corrections. Ten reaction motors are located in upper "C" transition section for third stage control. During third stage burning, the large pitch, yaw and the four roll motors are used and operate similarly to the second stage. To conserve fuel during the coast phase of third stage, the pitch control is transferred to the small motors, the yaw motors are shut off, the roll motors are throttled down to a low level thrust and the yaw control is transferred to the roll motors, see Figure . The roll motors are then firing in pairs on opposite sides of the vehicle for roll control and on the same side for yaw control.

4.1.2.3 Fourth Stage

The fourth stage, including payload does not have an active control system, but receives its spatial orientation from the control exerted on the vehicle by the first three stages. Stabilization of the fourth stage and payload is accomplished by spinning the entire stage prior to separation from the third stage. Spin up is achieved by four small rocket motors located just below the separation plane. The rate at which the fourth stage spins will vary with payload weight.

4.1.3 Operation of the Guidance and Control System

The proportional control signals and forces of the first stage system and its integrated feedback loop are typical of a present-day airplane attitude reference and stability control system. Familiarity with the operation of such a system is so wide spread that elaboration here seems unwarranted. However, the reaction thrust systems of the second and third stages, while not a new concept, are novel to a degree and a brief description of their mode of operation follows. An error signal originating in the guidance package opens the motor valves which allows hydrogen peroxide to pass from the tanks, through the expulsion tubes, peroxide lines and motor control valves. Then, it is distributed over the decomposition chambers' silver screen catalyst beds and decomposes into super-heated steam and oxygen. The hot gaseous products expand through a convergent-divergent nozzle and produce a reaction thrust. This process continues until the thrust produced by the motors has satisfied the trajectory correction requirements of the vehicle. After the necessary correction has been effected, the guidance package will interrupt the signal to the peroxide control valve- thus, closing the valve and interrupting the flow of the peroxide.

4.1.4 Guidance System Setting

Table 3-1 lists the nominal sequence of events as programed by the Intervalometer. Figure presents the sequence in graphical form. The nominal gain settings for the guidance and control system are found in Table 4-2.

4.1.5 Control System Settings

A list of representative settings for the reaction control system is set forth in Table 4-3. The frequency response characteristics are set forth in Figure .

TABLE 4-2

GUIDANCE SYSTEM GAINS

TABLE 4-3

NOMINAL CONTROL SYSTEM SETTINGS

Second Stage Control System

Thrust Levels - 100,000 Feet:

One Yaw or Pitch Jet and two Roll Jets operating.

Pitch or Yaw Jet	510 $\pm$ 30 lbs thrust
Roll Jets (2)	46 $\pm$ 6 lbs thrust

One Pitch, One Yaw and two Roll Jets operating

Pitch and Yaw Jets	490 $\pm$ 30 lbs thrust
Roll Jets (2)c	44 $\pm$ 4 lbs thrust

Useable Fuel Capacity: 176 lbs 90% H_2O_2 minimum.

Response Times:

	Rise Time		Decay Time	
	10%	90%	90%	10%
Pitch or Yaw Jet	0.110	0.130	0.070	0.090 sec.
Roll Jet	0.045	0.090	0.040	0.110 sec.

Third Stage Control System

Thrust Levels - 200,000 Feet:

Large Pitch Jet	44 $\pm$ 4.4
Small Pitch Jet	2.2 $\pm$ 0.8 - 0.4
Yaw Jet	4.4 $\pm$ 4.4
Roll Jet (Burn)	14. $\pm$ 1.4
Roll Jet (Coast)	3.0 $\pm$ 1.0

Useable Fuel Capacity: 17.5 lbs 90% H_2O_2 minimum

Response Times:

	Rise Time		Decay Time	
	10%	90%	90%	10%
Pitch or Yaw Jet	0.045	0.090	0.040	0.110 sec.
Roll Jet	0.090	0.110	0.040	0.110 sec.
Small Pitch	0.045	0.090	0.040	0.090 (25%) sec.

4.2 Vehicle Telemetry System

A 15-channel PAM/FM/FM telemetry system is located in Transition Section "D" (lower). All measurements made on the first three stages for flight performance evaluation will be telemetered by this unit. In broad terms these measurements are the various temperatures, pressures, accelerations, events, guidance package outputs and control systems functions required for performance evaluation and flight analysis for the first, second, and third stages. The detailed measurement list and associated channel assigment is presented in Appendix A. A general system block diagram and typical simplified measurement circuits are illustrated by Figure

The following discussion is offered to acquaint the reader with the design concept of the Scout vehicle telemetry system. While it is somewhat greater in detail that other sections of this report it should not be interperted as a handbook of instructions or should the nominal values presented be used in the operational checkout of the system. Applicable instruction documents are provided for this purpose.

The overall system utilizes both continuous and commutated methods of signal monitoring. The major components in each measurement signal path are similar, i.e., end instrument (sensor), voltage controlled sub-carrier oscillator, mixer-amplifier, and transmitter. Further signal conditioning is done where necessary to convert sensor outputs to the basic 0 to 5 V.D.C. full-scale measurement scheme.

The commutator employed is a two-pole, 30-channel, 5 R.P.S. (30 X 5) switch, and conforms, along with the standard sub-carrier frequencies, to I.R.I.G. standards.

The 15 sub-carrier channels are mixed and amplified to provide the composite intelligence wave-form for frequency modulation of the transmitter carrier. Figure is a block diagram of this arrangement. The standard F.M. transmitter has a nominal output power of 10 watts.

Power requirement for the complete telemetry system is approximately 3.5 amps at 28 V.D.C. This is furnished by a battery pack of 19 silver-zinc cells. This battery is capable of supplying power to the system for approximately one hour.

Potentiometer and Thermistor type sensors in the system receive their excitation from a 10 V.D.C. solid state converter. This precision power source is also used for commutator calibration and reference voltages and for bias requirements in the gyro output signal conditioning phase sensitive demodulators.

A second solid state converter provides the 250 V.D.C. required by the R.F. transmitter. This precision power source is also used as the excitation voltage for the on-off matrix function voltage dividers. All other components in the telemetry system receive their power from the 28 V.D.C. battery.

Pressure measurements are acquired by the use of potentiometer-type pressure transducers. The precision voltage source is impressed across a voltage divider network and the transducer wiper derives its stimulus from varying pressure. A typical circuit for this type measurement and those described in the next several paragraphs are described on Figures   thru   .

A rectilinear potentiometer is installed in each of the four fins to measure the deflection of the fin tips and jet reaction vanes.

Temperature measurements are obtained through the use of thermistor elements (negative coefficient semi-conductor material) that are connected to a voltage divider network simliliar to the above potentiometer types. Output voltage variation due to thermistor resistance change to temperature is within the 0 to 5 V.D.C. measurement range.

Rate and displacement function are 400 cps, single-phase, low level A.C. voltage signals exhibiting a phase reversing characteristic. A phase sensitive demodulator is employed in this type circuit to produce the basic system requirement of 0 to 5 V.D.C. output for full scale measurement range. The minimum A.C. signal for full scale output from the demodulator is ± 20 MV (RMS). The reference phase required for the demodulators is the primary 15 VRMS, 400 cps guidance power.

Acceleration measurements in the vehicle are detected by magnetic type (variable-reluctance) transducers. These units require a D.C. power input, supply their own regulation, and full scale output is 0 to 5 V.D.C. Power is direct from the telemetry system battery.

The guidance pitch program voltage measurement requires a special isolated system since a guidance requirement is that this voltage must be maintained above vehicle ground. The pitch program output signal is impressed across a high resistance network, and the signal then modulates a low-level (milli-volt) sub-carrier oscillator. This oscillator is a differential amplifier input type with no reference to ground.

Guidance system 400 cps, 15 V.R.M.S. single-phase power is monitored by use of a voltage divider and the resulting ± 2.5 V.A.C. (Peak-to-Peak) signal modulates a standard high-level oscillator. This oscillator differs from the other 5 V.D.C. input type in that the oscillator band-width (± 75%) is adjusted for ± 2.5 V.D.C. instead of 5 V.D.C. This technique provides for monitoring of both signal voltage, amplitude and frequency.

Various other on-off type functions are monitored by relay matrix. Typical of these are the control motor operation for stages 2 and 3. Chamber pressure and valve operation signals are combined in a typical fourth order matrix to completely identify bi-directional guidance control. This type of circuit is also used to monitor the Command-Destruct Receiver and ejection of the fourth stage heat shield.

4.3 Vehicle Tracking System

An Avion 149 C-5 "C" band radar beacon with 400 watts peak pulse power and 700 watts maximum peak pulse power is installed in the vehicle Transition Section "D" (lower). This transponder assists the ground radar in acquisition and tracking of the vehicle to the point of fourth stage separation. The unit, which is pressurized and completely transistorized, utilizes an antenna horn that provides a 15 db gain. This combination of features allows for maximum tracking range capability and improves range safety aspects.

4.4 Vehicle Destruct System

The Command-Destruct System is comprised of two completely separate circuits each with dualized associated components, either of which will destroy the first three stages of the vehicle. Thrust termination of the first three stages is accomplished by side-case splitting the rocket motors. An additional function of the system is to disable all unfired rocket ignition circuits upon reception of the destruct signal.

In addition to the command-destruct system which is initiated by a coded radio command, the vehicle is equipped with an auto-destruct system for the first two stages. Essentially this is a power-lanyard switch circuit in "B" and "C" transition sections paralled across the destruct system safe/arm units. Premature separation or break-up of the first two stage transition sections will also initiate the destruct of those stages. A pressure switch is included in the auto-destruct battery circuit to preclude destruction of the expended stage at normal stage separation.

The minimum power of the Command-Destruct System is 500 watts and power amplification increases this value to 10,000 watts. The transmitter carrier wave is frequency modulated by code tones to operate a relay-interlock network in the destruct receiver upon proper command.

The bow-tie antenna pairs are mounted diametrically opposite one another to provide near omni-directional coverage about the vehicle. Each antenna pair serves one receiver.

"Safe" and "Arm" operation of the Safe/Arm units is by rotary solenoid operation through hard lines to blockhouse arming console switches.

4.5 Vehicle Ignition System

The ignition system design for the vehicle is based on dual battery systems (ignition-destruct uses common batteries) which incorporate a duality of other components up to and including the end igniter elements of the rocket motors.

Remote "Safing" and "Arming" of the igniter circuits from the blockhouse arming console is provided by means of arming relays in the vehicle. Also provisions are included for blockhouse monitoring of each of the ignition squib bridges by circuit resistance.

The pair of ignition-destruct silver cell batteries are remotely activated by command from the blockhouse. These batteries supply 22.5 volts open circuit, and 16 volts under the anticipated peak load. Approximately two minutes (maximum) are required to fully charge the cells after battery activation. Heater requirements are 80 watts at 28 V.D.C. Provisions are made to monitor the battery voltage no-load and under simulated load, in the blockhouse.

Ignition of the vehicle first stage is by hard line command from the blockhouse programmer. Subsequent stages are fired by command from the guidance intervalometer.

4.6 Solar Radiation Satellite

Detailed information as to measurements, channel assignments, or other items pertaining to the satellite mission are not available. In general, however, the following items are of interest.

A mechanical separation system is employed to eject the satellite from the fourth stage subsequent to burn-out. This is accomplished by a battery-explosive nut circuit activated by a timer. The timer is initiated by signal from the power control box at fourth stage ignition. Timer control is variable from 0 to 30 minutes. Compressed springs eject the satellite upon explosive nut activation.

The satellite has a spin-up system of rocket motors and gas jets which are controlled by radio command from the ground.

Intelligence from the satellite transmitter is in pulse code from (PCM), other details are not presently available.

5. TEST OBJECTIVES

5.1 General

The SOL RAD (SR-5) is the fifth in a series of solar radiation satellites, and the NASA/DOD Scout is the eleventh in a series of similar vehicles to be utilized for boosting instrumented payloads to various probe altitudes or into near earth orbits.

The SOL RAD payloads will perform the function of obtaining measurements of solar X-rays in four bands and measurements of the Lyman Alpha effect at an altitude of approximately 390 N.M.

NASA/DOD Scout vehicle will place this satellite into the prescribed orbit and will also demonstrate the integrity of the various subsystems and techniques used in this flight. In addition to testing standard hardware, some of which was only recently incorporated on Scout S-110, this flight will demonstrate the structural integrity of the "beefed-up" "C" and "D" Transition sections. These are the major configuration changes which are effective on Scout S-111 and subsequent vehicles.

5.2 Precedence of Objectives

5.2.1 First Order Test Objectives

First order test objectives are those which are the basic reason for conducting the flight test. These objectives are of such a nature that inability to achieve them must result in the flight being delayed or aborted. Each system related to a first order objective must be operating properly at launch.

5.2.2 Second Order Test Objectives

Second order test objectives are those required to determine the over-all performance of the test vehicle or specific systems. A second order objective can be deleted without compromising the attainment of the first order objectives. The flight may be delayed to achieve a second order objective if it will not jeopardize the first order objectives. A maximum amount of time may be allowed for the repair of equipment pertaining to these objectives.

5.2.3 Third Order Test Objectives

Third order test objectives are those which furnish additional data for over-all vehicle evaluation, or for special investigations. Any "hold" time required for these items will be up to the discretion of the Test Director. The launch should not be delayed if it is found that a third order test objective will not be met.

5.2.4 Instrumentation Priorities

The priorities listed in Table 5-1 have been included as a guide for determining "hold" conditions and data acquisition plans. Priorities were assigned on the basis of the importance in obtaining the test objectives.

5.3 Payload First Order Test Objectives

(NRL Responsibility. CVC not provided with this information.)

TABLE 5-1
Vehicle Test Objectives

Objective	Order	Key Evaluation Parameters	Expected Value	Instrumentation Priority	Data Required Via	Data Acquisition Times Required	Ground Data Acquisition Stations	Ground Data Acquisition Priority
Orbit Injection Determination Demonstrate that satisfactory injection of the payload into the circular orbit has been achieved.	1	Receive Satellite data after first pass. a) Nominal Altitude b) Orbital Inclination	 390 Naut. Mi 75.4°		SOL RAD Ground Station	T+650 sec, Continuous		1
Overall Vehicle Performance Demonstrate that the vehicle possesses the capability to follow the predicted trajectory	1	Altitude Range Velocity Flight Path Angle	As defined by final trajectory data.		Vehicle Tracking	T-0 to 4th Stage separation	FPS-16 (023002)	1
							FPS-16 (023001)	2
Evaluate Guidance System Performance Demonstrate that the guidance system will maintain proper attitude during flight.		a) Pitch Program Voltage b) Roll Rate c) Pitch Rate d) Yaw Rate e) Roll Disp. f) Pitch Disp. g) Yaw Disp. h) Guidance Volt. Reference	0 - 615 MV ± 15°/sec ± 8°/sec ± 8°/sec ± 5°/sec ± 4°/sec ± 5°/sec 400 cps	1	Channel 1.7 Channel 3.90 Channel 5.40 Channel .56 Commutated 40 KC Channel 22.00	T-0 to LOS	Point Arguello T/M Station	1
							Probe Launch Complex, T/M Van	1/2
								(1 - until T/M Station at Arguello receives signal)
Evaluate Control System Performance Demonstrate control system can cause vehicle to follow instructions from guidance system.	1	a) Second & Third Stage Yaw Motors	ON-off	1	Channel 0.96	T+70 to T+600	Point Arguello T/M Station	1
		b) Second & Third Stage Upper Roll Motors	ON-off	1	Channel 1.30	"	Probe Launch Complex, T/M Van	1/2
		c) Second & Third Stage Large Pitch Motors	ON-off	1	Channel 0.73	"		
		d) Second & Third Stage Lower Roll Motors	on-off	1	Channel 3.00	"		

TABLE 5-1 (Continued)

Objective	Order	Key Evaluation Parameters	Expected Value	Instrumentation Priority	Data Required Via	Data Acquisition Times Required	Ground Data Acquisition Stations	Ground Data Acquisition Priority
(Cont.)	1	e) Third Stage Small Pitch Motors	on-off	1	Channel 2.30	T+70 to T+600	Point Arguello T/M Station	1
		f) Tower fin position	± 18.5°	1	Commutated Channel 30 KC	T-0 to T+73	Probe Complex T/M Van	1/2
		g) Left fin position	± 18.5°	1	"	"		
		h) Bottom fin position	± 18.5°	1	"	"		
		i) Right fin position	± 18.5°	1	"	"		
		j) Second Stage N_2 pressure	500 - 3000 psia	2	"	T-0 to T+117		
		k) Third Stage N_2 pressure	500 - 3000 psia	2	"	T-0 to LOS		
Evaluate First, Second, and Third Stage Motors	1	a) Long. Acceleration	-1 to +12 "g"	1	Channel 14.50	T-0 to LOS	Point Arguello T/M Station	1
		b) Normal Acceleration	± 1 "g"	2	Channel 10.50	"	Probe Complex T/M Van	1/2
		c) Transverse Acceleration	± 1 "g"	2	Channel 7.35	"		
		d) 1st. Stage Headcap Press.	0-500 psia	2	Commutated; 30 KC (Channel 24)	T-0 to T+45		
		e) 2nd Stage Headcap Press.	0-550 psia	2	40 KC (Channel 2, 10, 18)	T-0 to T+115		
		f) 3rd Stage Headcap Press.	0-350 psia	2	40 KC (Channel 6, 14, 22)	T-0 to T+160		

TABLE 5-1 (Continued)

Objective	Order	Key Evaluation Parameters	Expected Value	Instrumentation Priority	Data Required Via	Data Acquisition Times Required	Ground Data Acquisition Stations	Ground Data Acquisition Priority
Evaluate Airborne Telemetry System	2	a) Receive T/M signals	See Appendix "A"	1	13 continuous and 2 Commutated Channels	T-0 to LOS	Point Arguello T/M Station	1
							Probe Complex T/M Van	1/2
Evaluate Airborne C-Band Transponder	1	a) Acquisition by FPS-16 Radar	—	1	—	T-0 to 4th Stage Separation	FPS-16 (023002)	1
							FPS-16 (023001)	2
Evaluate Ignition System		a) Longitudinal Acceleration	-1 to +12 "g"	1	Channel 14.50	T+73 to T+120	Point Arguello T/M Station	1
							Probe Complex T/M Van	1/2
Evaluate Heat Shield Ejection	2	a) T/M Event	—		Commutated 40KC Channel 1, 11, 23	T+115	Point Arguello T/M Station	1
							Probe Complex T/M Van	
Evaluate Command-Destruct System (If Required)	1	a) T/M Event	—	1	Commutated 40KC Channel 1, 11, 23	T-0 to LOS	Point Arguello T/M Station	1
		b) Longitudinal Acceleration	-1 to +12 "g"	1	Channel 14.50	T-0 to LOS	Probe Complex T/M Van	1/2
		c) Motor Headcap Press:						
		First Stage	0-500 psia	2	Commutated 30 KC Channel 24	T-0 to T+45		
		Second Stage	0-550 psia	2	40 KC Channel 2, 10, 18	T-0 to T+115		
		Third Stage	0-350 psia	2	40 KC Channel 6, 14, 22	T-0 to T+160		

TABLE 5-1 (Continued)

Objective	Order	Key Evaluation Parameters	Expected Values	Instrumentation Priority	Data Required Via	Data Acquisition Times Required	Ground Data Acquisition Station	Ground Data Acquisition Priority
Evaluate Thermal Structural Environment	3	a) Trans "C" Amb. Temp.	70-130° F	3	Commutated 30 KC (Channel 8)	T-0 to T+600	Point Arguello T/M Station	1
		b) Trans "D" Amb. Temp.	80-130° F	3	(Channel 13)	"	Probe Complex T/M Van	1/2
		c) Trans "B" Nozzle Insul. Temperature	70-160° F	3	(Channel 14)	T-0 to T+115		
		d) Base "A" Nozzle Insul Temperature	70-150° F	3	(Channel 15)	T-0 to T+73		
		e) Trans "B" Inside Skin Temperature	70-300° F	3	(Channel 16)	T-0 to T+115		
		f) 3rd Stage N_2 Temp.	70-160° F	3	(Channel 7)	T-0 to T+600		
Evaluate First Stage Hydraulic System	2	a) Hydraulic Accumulator Pressure	3000 psia	2	Commutated 30 KC (Channel 23)	T-0 to T+73	Point Arguello T/M Station	1
							Probe Complex T/M Van van	1/2

6. DATA ACQUISITION AND ANALYSIS

6.1 General

Three methods will be used at PMR to acquire performance and tracking data from the NASA/DOD Scout Vehicle. These are telemetry, FPS-16 radar, and camera tracking. Meteorological data will be used in a secondary sense to define the wind disturbances prior to launch and to provide supporting data for trajectory analysis.

The Solar Radiation Payload Lab and Ground Station will receive telemetry data from the satellite. The analysis of this information will be the responsibility of the payload experimentor.

6.2 Telemetry Data

Telemetered data, transmitted on one link frequency of 249.9 at an effective radiated power of 10 watts, will be received by the telemetry ground station. This system will be radiating vehicle data during ground checkout and flight up to fourth stage separation. It is possible that telemetry signals may continue past fourth stage ignition if the components located in transition section "D" are not damaged by the ignition of the fourth stage.

The telemetry system, as described in Section 4 , will be used to transmit operation parameters of the vehicle's subsystems and to determine the spacial orientation of the vehicle. These measurements will also be used to define any malfunctions or abnormal occurances which might be encountered.

The PMR will record the link frequency on magnetic tape and will plot the parameters on the oscillograph format that is described in SSVT-24 OR. The slow speed, 0.5 in/sec. plotters should be available in the 24-hour data package.

6.3 Radar Tracking Data

The vehicle will be acquired and tracked by FPS-16 radar units. These units will be used with an Avion model 149C transponder-beacon carried in transition section "D". The transponder is rated at 400 watts peak pulse power with 700 watts peak power maximum. It is completely transistorized, pressurized, and coupled with an antenna horn that provides a 15 db gain.

The raw data obtained from the FPS-16 radar will be recorded on magnetic tape, Nixie film, and plotboards. The tape is considered to be the primary data source and this information in conjunction with meteorological data will be analyzed to define the vehicle's actual flight performance. Copies of the plotboards should be available in the 24-hour package.

The planned trajectory will require the use of the FPS-16 fixed tuned magnetrons. The PMR Frequency Schedular has assigned 5491 MC and 5566 MC as the airborne receive and transmit frequencies, respectively. These frequencies will be set in the equipment prior to delivery to the launch site.

6.4 Camera Tracking

Three color 35 MM cameras will be used to track the vehicle from T-5 sec. to the limit of tracking capability and three color 16 MM cameras will cover the interval of 0 to 1000 feet. These will be used to visually check the performance of the first stage and possibly to confirm second stage ignition.

Additional photographic requirements exist for documentary purposes. These will include 16 MM cameras for tracking the vehicle showing various satellite laboratory operations and recording the installation of the payload. Both color and BW stills of the vehicle on the launcher, the satellite checkout and installation will also be required.

6.5 Data Reduction

Data reduction for the vehicle will be accomplished in three distinct operations in order to provide the maximum amount of flight evaluation in the shortest period of time. These operations are real time, quick look and the final test report.

6.5.1 Real Time Data Acquisition and Evaluation

This information is collected, processed and evaluated while the vehicle is still in flight. It will be used primarily to obtain an estimate on the NASA/DOD Scout performance up to fourth stage separation. Three real time plotters, displayed in the telemetry ground station, will record the information so designated in Tabel A-1 thru A-3 following the plotter format describedin SSVT-24 OR. Monitoring these recorders will allow immediate reporting of events such as ignition of stages, pitch program steps, length of coast periods and ejection of the fourth stage heat shield. Copies of the plotboards will also be used in the real time evaluation of the vehicle performance.

6.5.2 Quick Look Data Acquisition and Evaluation

For this operation, selected data is collected as soon as possible after launch and is used by the Launch Test Working Group to prepare a "Quick Look", report on the performance of the vehicle. This gives more accurate, comprehensive information on the flight parameters and achievement of the test objectives than can be obtained in the real time evaluation. This quick look evaluation uses the results of the real time operation plus the reduced telemetry recordings, films and tracking information that are available for evaluation of various subsystems. The results of this evaluation will be used in any subsequent reports that may be required for the program offices or contractor's use.

6.5.3 Final Evaluation

Final evaluation is made after all records pertaining to the flight are available and can be used to fully analyze the test results. It is expected that this data will be reduced and available for use in the final evaluation by 3 days after the launch. This effort includes a detailed operation and system analysis but does not include payload performance. The results of this evaluation is combined into a Final Flight Test Report which will be available 21 days after the launch.

6.6 Orbital Determination

The Naval Research Laboratory will have the responsibility for tracking the SOL RAD Satellite, collecting and evaluating the data.

7. REFERENCES

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b. "PMR Ground Safety Report - NASA/DOD Scout Vehicle", Report No. AST/R-13430-1-4, Chance Vought Corporation, Astronautics Division, Los Angeles Facility. 15 December 1961. Confidential

c. "Inflight Termination Report" No. AST/R-87620-1-1. Chance Vought Corporation, Astronautics Division, Los Angeles Facility. 15 November 1961

d. "Scout Calibration and Checkout Manual for DYG739D1 Guidance and Control System", R-ED 11145. Honeywell Aeronautical Division, Los Angeles, California. 21 November 1961.

e. "Range Safety Information - NASA/DOD SCOUT VEHICLE - SOL RAD MISSION". Report No. 3-13430/2R-1. Chance Vought Corporation, Astronautics Division, Los Angeles Facility. 17 January 1962.

f. "Operations Requirements - SOL RAD Scout Vehicle Launch" No. SSVT-24. Headquarters, Pacific Missile Range, Point Mugu, California (Pending).

g. "Operations Directive - SOL RAD Scout Vehicle Launch" No. SSVT-24. Headquarters, Pacific Missile Range, Point Mugu, California (Pending).

h. "Preliminary Countdown Document - Air Force Scout Vehicle" Report No. AST/R-87620-1-2. Chance Vought Corporation, Astronautics Division, Los Angeles Facility. 16 November 1961.

i. "Launch Test Directive" - SOL RAD Scout Vehicle Launch. Headquarters, Pacific Missile Range, Point Mugu, California (Pending).

APPENDIX A

DATA MEASUREMENTS

APPENDIX A

DATA MEASUREMENTS

This appendix outlines the data measurements required to meet the test objectives and to provide adequate information for post-flight evaluation. Table A-1, A-2, and A-3 present the NASA/DOD Scout measurements that are to be telemetered from the vehicle. A priority is assigned to these functions in order to insure the vehicle is launched with adequate information for monitoring the flight. These priorities (1,2 and 3) are defined in Section 5 of this report.

Table A-4 in this section presents the ground monitor data required during the countdown. Details on the transducers and instrumentation system may be obtained from Section 4 of this report.

TABLE A-1 TELEMETRY MEASUREMENTS

(Transmitted by PAM/FM/FM Telemeter of Link Frequency - 249.9 MC)

IRIG CHANNEL	FREQ. KC	MEASUREMENT	END INSTRUMENT MEASUREMENT	OPERATIONAL PRIORITY	REAL TIME REQ.
2	0.56	Yaw Rate	± 8 deg/sec	1	X
3	0.73	2nd Stage pitch motor and 3rd Stage pitch motor	On - Off	1	X
4	0.96	2nd Stage yaw motor and 3rd Stage yaw motor	On - Off	1	X
5	1.30	2nd Stage upper roll motor and 3rd Stage upper roll motor	On - Off	1	X
6	1.70	Pitch Program Voltage	0-615 MV	1	X
7	2.30	3rd Stage small pitch motor	On - Off	1	X
8	3.00	2nd Stage lower roll motor and 3rd Stage lower roll motor	On - Off	1	X
9	3.90	Roll Rate	± 15 deg/sec	1	X
10	5.40	Pitch Rate	± 8 deg/sec	1	X
11	7.35	Transverse Acceleration	± 10 "g"	2	X
12	10.50	Normal Acceleration	± 10 "g"	2	X
13	14.50	Longitudinal Acceleration	- 4 to + 20 "g"	1	X
14	22.00	Guidance Voltage 400 cycle Reference	0-15 VRMS	1	
15	30.00	* See Table A-2	Volts @ 400 Frequency		
16	40.00	* See Table A-3			

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TABLE A-2 TELEMETRY MEASUREMENTS

(Transmitted by PAM/FM/FM Telemeter Commutated 40 KC

on 249.9 MC Link Frequency)

CHANNEL NO.	MEASUREMENTS	END INSTRUMENT MEASUREMENT	OPERATIONAL PRIORITY	REAL TIME REQ.
1,11,2	Command-Destruct, Heat Shield		1	
2,10,18	2nd Stage Headcap Pressure	0-800 psia	2	X
3,7,15,19,24	Yaw Displacement	± 5°	1	X
4,8,12,16,20	Roll Displacement	± 5°	1	X
5,9,13,17,21	Pitch Displacement	± 4°	1	X
6,14,22	3rd Stage Headcap Pressure	0-400 psia	2	X
25	Spare @ 0%	0 VDC	2	
26	50%	25 VDC	1	
28,29,30	100%	5 VDC	1	

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TABLE A-3 TELEMETRY MEASUREMENTS

(Transmitted by PAM/FM/FM Telemeter Commutated 30 KC

on 249.9 MC Link Frequency)

CHANNEL NO.	MEASUREMENTS	END INSTRUMENT MEASUREMENT	OPERATIONAL PRIORITY	REAL TIME REQ.
1,9,17	Tower Fin Position	$\pm 20^\circ$	1	X
2,10,18	Left Fin Position	$\pm 20^\circ$	1	X
3,11,19	Bottom Fin Position	$\pm 20^\circ$	1	X
4,12,20	Right Fin Position	$\pm 20^\circ$	1	X
5	Guidance Temp.	500° F	3	
6	Spare @ 0%	0 VDC	3/2*	
7	3rd Stage N_2 Temp.	500° F	2	
8	Trans. C Amb. Temp.	500° F	3	
13	Trans. B Amb. Temp.	500° F	3	
14	Trans. B Noz. Insul. Temp.	500° F	3	
15	Base A Noz. Insul. Temp.	500° F	3	
16	Trans. B Inside Skin Temp.	500° F	3	
21	2nd Stage N_2 Line Press.	0-3500 psia	2	
22	3rd Stage N_2 Line Press.	0-3500 psia	2	
23	Hyd. Accum. Press.	0-3500 psia	2	
24	1st Stage Headcap Press.	0-600 psia	2	X
25	Spare @ 0%	0 VDC	3/2*	
26	50%	2.5 VDC	1	
27	0%	0 VDC	1	
28,29,30	100%	5 VDC	1	

* Higher Priority applies depending on status of second spare @ 0% channel.

TABLE A-4

GROUND MONITORING FUNCTIONS

GROUND MONITOR DATA REQUIRED	PURPOSE	INTERIM
Second stage H202 Line Pressure	Check System Operation	Continuous
Third stage H202 Line Pressure	Check System Operation	Continuous
Second stage N2 Line Pressure	Check System Operation	Continuous
Third stage N2 Line Pressure	Check System Operation	Continuous
Command Destruct Channel 7, No. 1 & No. 2	Check System Operation	As Req'd.
Command Destruct Channel 7, No. 2 & No. 1	Check System Operation	As Req'd.
Zero, Half and Full Scale Calibration	Check System Operation	As Req'd.
Transtion "D" Amb. Temp.	Monitor Environment	Continuous
Transition "C" Amb. Temp.	Monitor Environment	Continuous
Transition "B" Amb. Temp.	Monitor Environment	Continuous
Third Stage Motor Skin Temp.	Monitor Environment	Continuous
Second Stage Motor Skin Temp.	Monitor Environment	Continuous
First Stage Motor Skin Temp.	Monitor Environment	Continuous
Second Stage N2 Pressure Switch	Check System Operation	Continuous
Third Stage N2 Pressure Switch	Check System Operation	Continuous
Second Stage Yaw Motor Chmbr. Press Switch	Check System Operation	Continuous
Third Stage Yaw Motor Chmbr. Press Switch	Check System Operation	Continuous
Second Stage Upper Roll Motor Chmbr. Press Switch	Check System Operation	Continuous
Third Stage Upper Roll Motor Chmbr. Press Switch	Check System Operation	Continuous
Second Stage Lower Roll Motor Chmbr. Press Switch	Check System Operation	Continuous
Third Stage Lower Roll Motor Chmbr. Press Switch	Check System Operation	Continuous
Second Large Pitch Motor Chmbr. Press Switch	Check System Operation	Continuous
Third Large Pitch Motor Chmbr. Press Switch	Check System Operation	Continuous
Third Small Pitch Motor Chmbr. Press Switch	Check System Operation	Continuous

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HEADQUARTERS
SPACE SYSTEMS DIVISION
AIR FORCE SYSTEMS COMMAND
UNITED STATES AIR FORCE
Air Force Unit Post Office, Los Angeles 45, California

5100*
1522

REPLY TO ATTN OF: SSVT/[redacted]/OS 9-4661, Ext. 3645

1522
SUBJECT: SOL RAD Scout Flight Test Plan — 1 February 1962

TO: Naval Research Laboratory
Attn: Mr. M. J. Votaw (Code 5170)
Washington 25, D.C.

1. The first draft of the subject report is forwarded for your review and comment. It will be noted that the test responsibilities (page 1-2)of BUWEPS and NASA have been omitted. These will be included in the final report.

[redacted]
Major, USAF
Director of TRANSIT/ANNA

1 Atch
SOL RAD Scout Flight Test Plan, dtd 26 Jan 62, CVC LOG #C-26 (C)

80134-BVE-210-05

Ref 594659

CLASSIFICATION OF THIS DOCUMENT WILL BE DOWN GRADED TO unclas UPON REMOVAL OF ENCLOSURES.

~~CONFIDENTIAL~~

05273

5170-35:MJV:js
NRL Prob R06-29
SER:

From: Director, U. S. Naval Research Laboratory, Washington 25, D. C.
To: Commander, Headquarters, Space Systems Division, Air Force Systems Command, United States Air Force, Air Force Unit Post Office, Los Angeles 45, California

Subj: Solar Radiation Scout Flight Test Plan

Ref: (a) SSD ltr SSVT/Capt. Browning/OS 9-4661, Ext. 3645 of 1 February 1962

1. In accordance with reference (a), the following information is submitted to improve the payload portion of the Flight Test Plan:

Solar Radiation IV B/s primary purpose is to measure the intensity of x-ray radiation from the sun. The 58-pound satellite will also study Lyman-Alpha ultraviolet radiation in the earth's night sky. Although externally similar to the Laboratory's earlier solar measurement spheres, the satellite embodies modifications based on analysis of the previous experiments.

Four x-ray detectors have been built into the ball to accomplish the sphere's primary purpose. Three of the detectors, protected by special magnets to deflect the charged particles of the Van Allen Belt that could saturate them, will cover the 2 to 8, 8 to 12, and 8 to 16 Angstrom wavelengths of the spectrum. (The Angstrom unit is a unit of measure used in expressing the length of light waves.) The fourth detector, designed to cover the 0.5 to 3 Angstrom wavelength range, has a relatively thick beryllium window that makes the special magnet unnecessary. The 2-8 and 8-16 Angstrom detectors are of the same type as those flown in Solar Radiation III, while the other two detectors will extend the wavelength and sensitivity range to more penetrating and more intense x-ray radiation than was manageable in the earlier solar measurement satellite experiments. These detectors will provide information about the hardness (intensity of penetration) of the x-ray emission spectrum and hence the electron temperature of the emitting source. The hardness of the emission spectrum plays a major role in the production of ionospheric disturbances due to events such as solar flares and active prominence regions.

The satellite's "night sky" experiment, which is to measure Lyman-Alpha radiation, consists of four detectors connected in parallel and looking out in the same direction from the satellite. Lyman-Alpha is an intense radiation from the sun and has a weakening

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SER:

effect on day-time radio communications when absorbed by the earth's upper atmosphere. The Lyman-Alpha line of hydrogen is in the extreme ultraviolet region of the spectrum between visible light and x-rays. Providing four times the sensitivity of a single detector, the four units will measure scattered solar Lyman-Alpha radiation from neutral atomic hydrogen in the earth's atmosphere. This experiment, which will remain dormant through most of the satellite's lifetime since it is meant to operate in darkness, will be operated by command from one or more ground telemetry stations. The latter will be able to activate and switch the experiment onto one of the telemetry channels of the satellite when Solar Radiation IV B is in the earth's shadow or when the sphere assumes a predetermined angle in relation to the axis of its flight. Monitoring the latitude and longitude aspects of the "night glow" in the earth's atmosphere caused by solar Lyman-Alpha radiation will provide scientists with an insight into the relationship between the solar radiation-affected upper atmosphere of the earth and the land below it.

COMPONENTS OF SOLAR RADIATION IV B

The 20-inch diameter aluminum shell containing six 9-inch solar cell patches is identical to Solar Radiation III. The electronics is in these experiments, Solar Radiation IV B has two spin-up systems. A cold gas jet system-consisting of a 3-inch diameter storage vessel charged with nitrogen to a pressure of 2000 psi and two explosively-operated valves-automatically goes into effect immediately after satellite separation. A switch fires the valves and allows the nitrogen to discharge through two nozzles on the satellite equator to produce a spin of about 80 revolutions per minute. A back-up system consisting of four miniature rockets mounted on opposite sides of the sphere's equator will be used if the cold gas jet fails. If the cold gas jets are successful, the rockets will be used to extend the useful life of the satellite after the original spin has decayed below a minimum useful level. This could double the useful life of the satellite.

Power Supply - Six 9-inch diameter solar cell patches are symetrically located on the surface of the sphere, three above and three below the equator, so that there will always be at least one patch facing directly into the sun. Each patch contains 174 cells and provides about two watts of power for satellite operation. The storage battery system has 18 cells in series with center tap grounded. This supplies 24 volts to the satellite transmitter and two 12 volt sources to the rest of the equipment.