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INTERFACE CONSIDERATIONS OF SOLID MOTORS WITH RAE A, B, IMP H AND J, AND PIONEER F AND G SPACECRAFT

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INTERFACE CONSIDERATIONS OF SOLID MOTORS WITH RAE A, B, IMP H AND J, AND PIONEER F AND G SPACECRAFT

Daniel W. Dembrow
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Introduction

Solid propellant rocket motors were employed for the final propulsive stages for the Radio Astronomy Explorers, Interplanetary monitoring platforms H and J and Pioneer F and G spacecraft. Each of these spacecraft had unique missions and special instrumentation aboard, some in close proximity to the solid propellant rocket motor. The nature of the functions to be performed by the spacecraft and the type of experiments to be conducted imposed special requirements on the rocket motor, special attention to the mechanics of attachment, careful analysis of the sequence of motor operation with respect to other spacecraft functions and meticulous concern with the rocket's effect upon spacecraft and experiments. Some interface problems with the six spacecraft listed above are described along with an account of the solutions to those problems.

Radio Astronomy Explorers

The Radio Astronomy Explorer (RAE) spacecraft was employed in two versions. RAE-A was launched into a cis-lunar orbit on July 4, 1968 from the Western Test Range to attain a circular 6000 km earth orbit (see Figure 1). RAE-B was launched into a lunar anchored orbit on June 19, 1973 from Eastern Test Range to attain a circular lunar 1100 km orbit at an inclination of 60° post grade (see Figure 2). Although the missions were quite dissimilar, both spacecraft used the same model solid propellant rocket motor. RAE-A used it as an apogee kick stage to increase the velocity from a transfer orbit to a circular one. RAE-B used it as a retro stage to decrease the velocity in the moon's vicinity to permit lunar capture.

A sketch of the RAE-A spacecraft is shown in Figure 3. The figure shows several of the potential interface problems. Note the proximity of the solar paddles to the exit plane of the rocket motor. The spacecraft can be seen to be symmetrical indicating the intent to be carefully balanced. Presence of libration damper booms indicate that some coning could be accommodated. Note also the buried location of the motor, suggesting possible thermal problems. Finally, the presence of the four antennae to be deployed to their 750 foot lengths implied that no disturbing forces could be tolerated by the time of main antenna deployment.

A brief review of the launch sequence shows the variety of potential interface problems. The RAE-A spacecraft was launched from a Thor Delta launch vehicle (referred to as the DSV 3J). The launch vehicle used its second stage guidance and control system to aim the remaining stages into the proper transfer orbit. Prior to firing the third stage solid propellant rocket motor, the spacecraft was spun up to 74 rpm. Shortly after third stage motor burnout the spacecraft with fourth stage motor

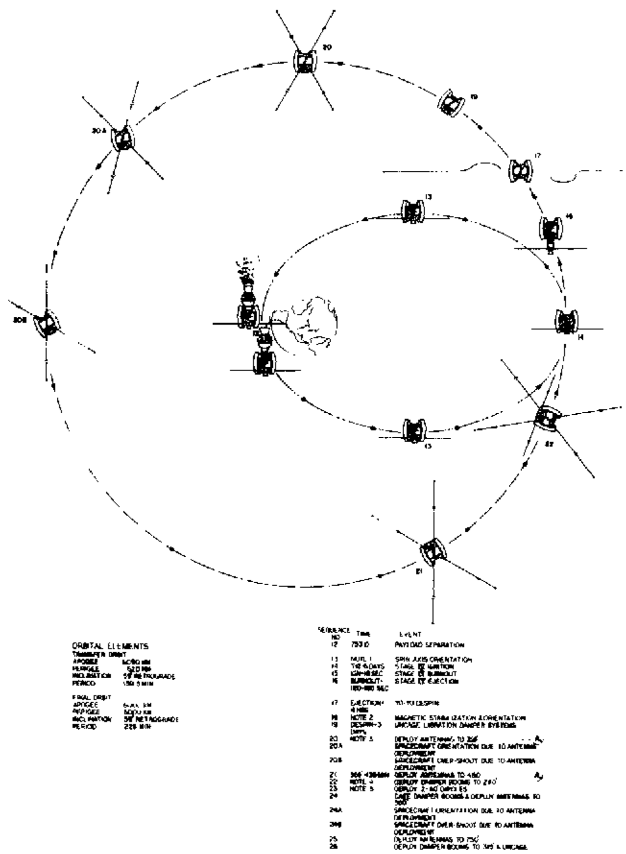

Figure 1. RAE-A Launch and Injection Sequence

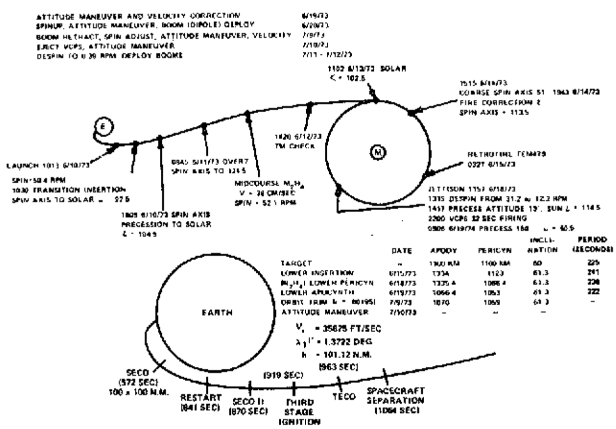

Figure 2. RAE-B Launch and Injection Sequence

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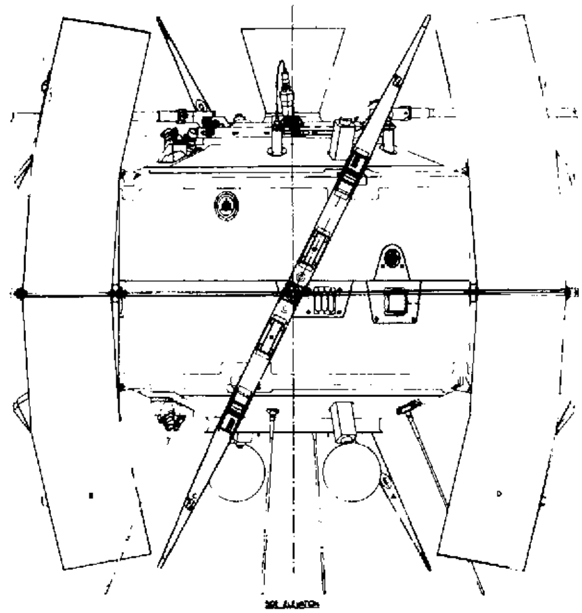


Figure 3. RAE-A Spacecraft

attached was separated from the third stage. Spacecraft continued to coast to apogee having entered an elliptical orbit of 554 km by 6100 km with an anomalistic period of 160 minutes. During this period, coning was reduced by libration dampers and a magnetic orientation system aligned the spin axis vector. Based upon solar aspect sensor and tracking data information, calculations were made by ground computers to optimize orbital circularization. A ground command was sent to fire the fourth stage motor at the proper point in apogee based on computer results to inject the spacecraft into a 6000 km circular orbit. The circular orbit anomalistic period was 228 minutes but it could have taken as long as 8 days before the transfer orbit parameters were accurately determined to define the optimum time for effecting the transition to the circular orbit. Following the fourth stage burn period, the motor was jettisoned. The time between burnout and jettisoning had to be sufficiently long to prevent the post burnout chuffing or outgassing to cease, yet short enough so that the temperature rise from the motor walls did not degrade nearby instruments or disturb spacecraft components which had to perform functions subsequently. Yoyo devices were then employed to despin the spacecraft to 2 rpm or less. The spacecraft was then magnetically despun to zero rpm and oriented through ground commands. Gravity gradient stabilization was employed requiring the deployment of four antenna booms when the spin axis of the spacecraft was aligned to within 10° of the local horizontal in the northern hemisphere. After gravity gradient capture, the libration damper boom was deployed followed by the release of the dipole antenna to attain the final spacecraft configuration. From the account of the orbital injection sequence, it is apparent that the following requirements had to be imposed upon the fourth stage solid rocket motor:

1. An incremental velocity capability which in turn fixes I_{sp}, weights loaded and empty, or the propellant mass ratio.

2. Delineation of plume boundary so as not to damage the nearby solar cells.

3. Limit to maximum axial acceleration so as not to damage various delicate antenna mechanisms.

4. Non-magnetic material so as not to disturb the magnetic orientation system or magnetic sensors.

5. Balance during the transfer orbit and remain in balance post burnout so as not to disturb the libration damping mechanism.

6. Spin effect. The motor must be capable of functioning properly under spinning conditions yet not contribute to spinup during the burning phase.

7. Thrust misalignment limited so as not to produce coning in excess of that capable of being accommodated by libration damper mechanism.

8. Temperature of the chamber walls must remain low enough until motor is jettisoned so as not to damage nearby components or instruments.

9. Residual thrust and outgassing must be completed prior to the time of jettison to prevent possible collision with spacecraft.

10. Satisfactory ignition after 8 days in hard vacuum.

11. Withstand the temperature drop that would occur in shadow with aft section of motor radiating to outer space during the orbit transfer period.

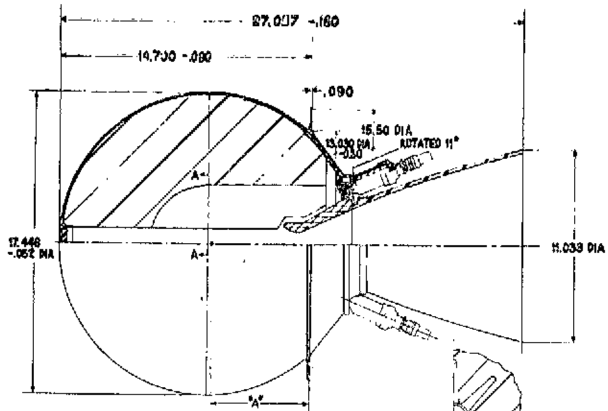
12. Withstand Delta Launch Vehicle environment of temperature, shock, vibration, acceleration, etc.

13. Use method of mounting a motor whose diameter is larger than the mounting flange.

To provide the main propulsion system for the RAE-A mission, the rocket motor illustrated in Figure 4 was developed. Manufactured by Thiokol Chemical Corp., Elkton, Maryland, it was designated as the TE-M-479. In Figure 5 a drawing shows the method of mounting and the provisions made for jettisoning the motor.

The same model rocket motor was used for RAE-B. The Figure 6 drawing shows the RAE-B spacecraft. The major change from RAE-A was the introduction of two additional propulsion systems: (1) cold gas system for attitude control and (2) hydrazine system for velocity adjustment. These modifications also permitted some of the delicate instrumentation which had previously been mounted on the center tube close to the rocket motor to be relocated outboard, thus making it less susceptible to heat soakback. In this manner the requirement for jettisoning the rocket motor before peak temperature no longer held. The case temperature of the rocket motor was monitored in flight until the heat soakback effect was dissipated.

Of the potential interface problems listed above, many were resolved by proper motor design. For example, the



MOTOR CONDITION	"A" CG IN.	WEIGHT LB M
LOADED	5.75	174.6
EMPTY	1.23	20.4

Figure 4. TE M 479 Rocket Motor

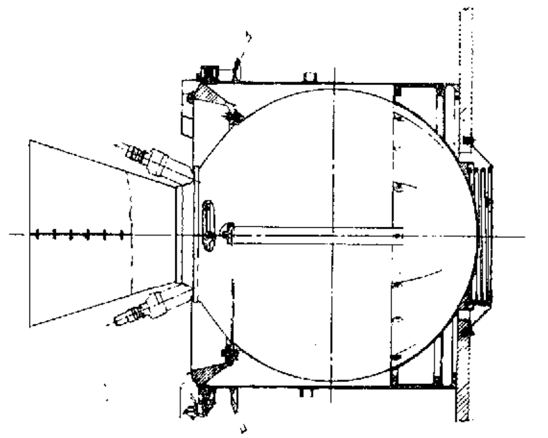


Figure 5. RAE-A Method of Mounting and Jettisoning Motor

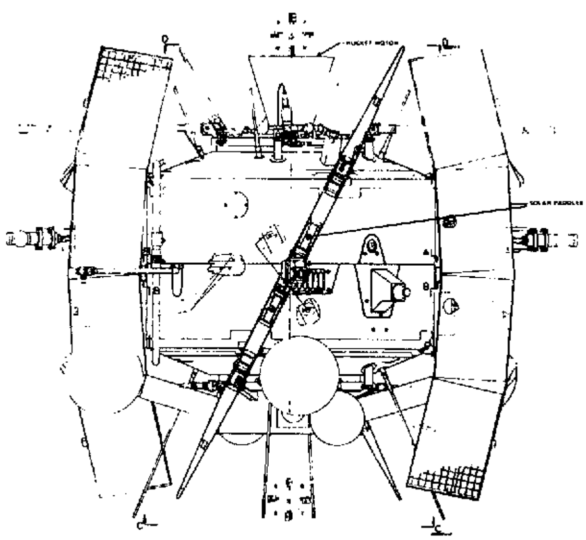


Figure 6. RAE-B Spacecraft

incremental velocity required was attained by selecting a propellant whose I_{sp} was well established and fixing upon the spacecraft weight and propellant weights to give the proper propellant mass fraction. The metallic components of the motor were made of non-magnetic materials, namely Titanium alloy (Ti 6 Al 4V) for the chamber, aluminum for the aft closure, and either nickel or other non-magnetic iron alloys, for the pyrogens. The motor and insulation were fabricated so as to be axially symmetrical thus limiting the imbalance to within the tolerable specification limits. A contoured nozzle rather than a conical one was designed so as to reduce the turning angle at the exit plane to limit the exposure angle of the nearby solar cells.

Of the remaining interface problems, many were resolved by an experimental test program. Balance measurements were made before and after loading with propellant. Residual magnetic field measurements were made of the metallic components of the motor prior to loading with propellant. Performance of the motor was established by testing under spinning and simulated altitude conditions. (Ref. 1.) These tests included a measurement of side forces during burning to determine the effect of thrust misalignment measurements. (Ref. 2.) The tests also included low pressure instrumentation to determine the post burnout exhaust and outgassing effects. The plume influence on nearby solar cells was first analyzed, then experimentally confirmed during the performance tests using a "rat-trap" device to protect the polished optical surface from the "blowback" effect. The motor was heavily instrumented with temperature sensors to delineate the temperature-time history. Thermal damage to the ejector piston was observed (ref. 1) but the extent of damage was tolerable.

RAE-B was launched 5 years after RAE-A. Propellant was removed and the TE M479 motors re-loaded for use on RAE-B. A design assurance test to determine the influence of a relief boot section removal was conducted with the same basic instrumentation as used on RAE-A tests (Ref. 3) but without the presence of an experimental spacecraft. Added were radiometers to determine contour and emissive characteristics of the plume.

The only solid rocket motor instrumentation used in flight was a temperature sensor mounted to the chamber wall. In flight, the peak temperatures did not exceed 300°C as compared to 280°C during ground static tests.

IMP

IMP H and J spacecraft were launched about 1 year apart, H on Sept. 23, 1972 and J on Oct. 26, 1973 both using a Delta launch vehicle from ETR to give low eccentricity cis-lunar - 30 - 40 earth radii orbits. IMP H and J were the last two of a series of ten interplanetary monitoring platforms whose purpose was to measure energetic particles, plasmas and magnetic fields in the earth-moon environment. The missions were unique in that the spacecraft had to be placed into very high altitude earth orbits of low eccentricity to continuously monitor the region about halfway between earth and moon (30-40 earth radii for IMP H and 23-46 earth radii for IMP J) remaining approximately 180 degrees apart for more than one year to provide simultaneous measurements of complementary phenomena on opposite sides of the earth. To place these spacecraft into the proper phase relationship, a three stage Delta launch vehicle was used in both instances. The spacecraft were first placed into an eccentric elliptical transfer orbit with apogee of 37 earth radii. The spacecraft were oriented during the orbital coast period, and the kick motor fired near first apogee to place the spacecraft into final orbits. With no further thrusting capability, it was impossible to correct for any errors from kick motor firing. The orbit injection was especially difficult for IMP J inasmuch as any error in velocity from the launch vehicle or kick stage would result in a rapid departure from the desired 180° phase relationship with IMP H. (Ref. 4.)

Figure 7 shows the IMP spacecraft. Figure 8 shows the orbital launch and injection sequence. Both launches used a Delta 1604 and a Thiokol TE-M-521-5 model kick motor (Fig. 9) derived from the IDCSP kick motors (Ref. 5). Potential interface problems are apparent by inspection of the spacecraft drawings. The solid rocket motor lies half buried within spacecraft and was retained post burnout. To eliminate thermal problems, excessive cooling during the shadowed coast phase, electrically heated blankets were employed (Fig. 10). To provide for installation of these blankets, they were divided into three sections - head end, body section, and nozzle end - requiring three thermostats mounted separately on the rocket motor wall. In addition, flight instrumentation was incorporated on the rocket motor using spacecraft telemetry to transmit information. Selected was a pressure transducer and a longitudinal accelerometer for monitoring motor performance.

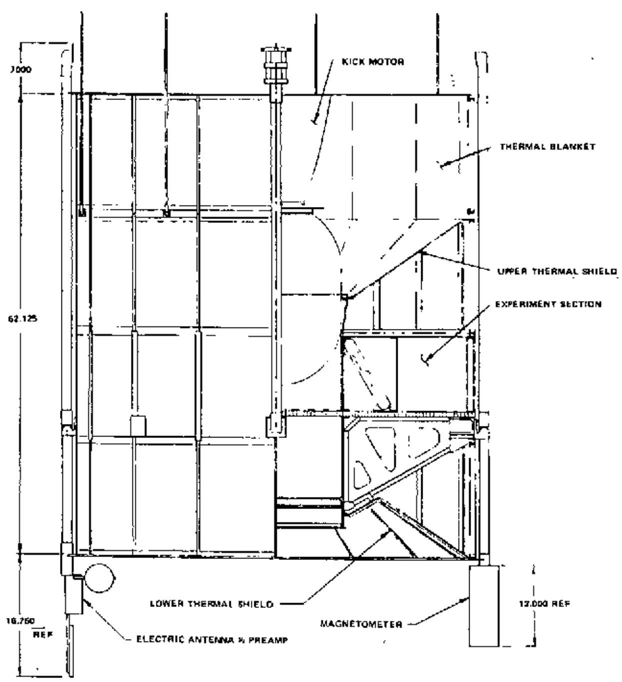


Figure 7. IMP-H General Layout

Interface problems were:

1. Unusual residual magnetic field constraints - namely less than 1/2 gamma at 36" after exposure to a 15 gauss permanent magnetic field followed by a 50 gauss deperm - requiring all metallic components of motor and instrumentation to be non-magnetic.

2. Thermal problems due to the presence of a thermal blanket and its attendant insulation characteristics.

3. Balance problem due to the presence of a thermal blanket with wiring along one side of the motor.

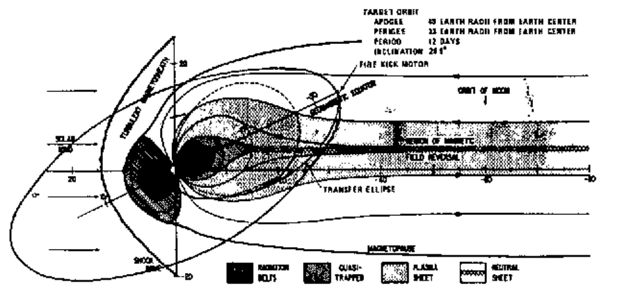


Figure 8. IMP H & J, Launch and Injection Sequence

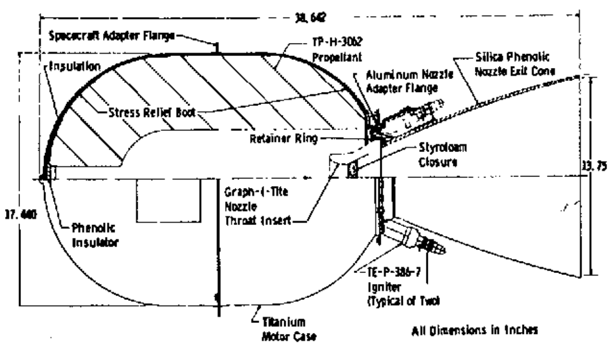


Figure 9. Thiokol Chemical Corporation TE-M-521-5 Rocket Motor

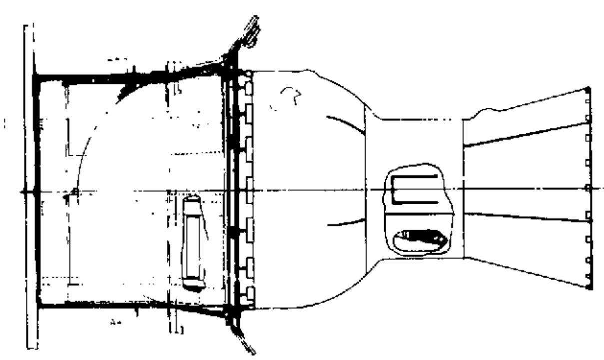


Figure 10. Blanket Assembly

4. Electronic instrumentation problem due to use of spacecraft telemetry for transmitting the kick motor performance data.

5. Requalification of the solid rocket motor to more severe launch vehicle environmental conditions necessitated by a decision to launch on a Delta 1604 rather than milder environment of a Delta 1603.

6. Sensitivity of the IMP J mission to accurate prediction of kick motor performance. A deviation of more than $1/2\sigma$ would result in an unfavorable phase drift.

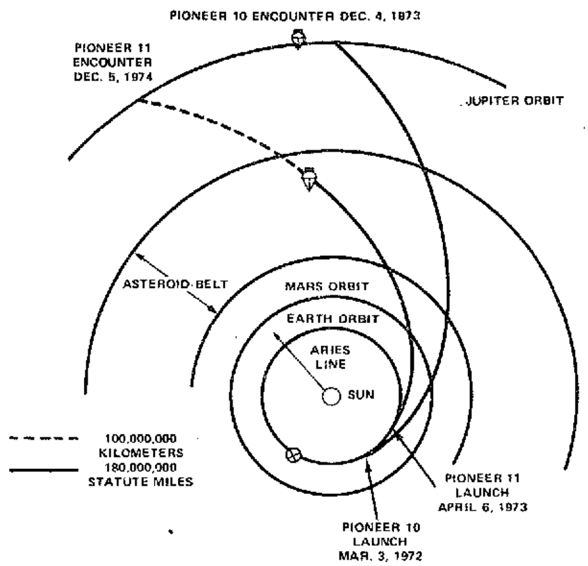
These problems were solved by both testing and design changes where required. (Ref. 6.) The unusual magnetic field constraint was met by employing throughout non-magnetic materials. A double wrapping of conetic shielding was used around the servo-type accelerometer to meet its requisite residual magnetism. Thermal functioning tests were performed on the thermal blanket during ground static test firings. Balance was attained by mounting copper ballast weights to offset imbalance. Flight instrumentation was designed to make use of either of two methods of telemetry - (1) FM/FM or (2) PCM/FM. A flight type plug-in patch switch enabled one to exercise an option of either continuous FM or digital (PCM) telemetering based upon ground system checkouts prior to launch. The coding electronics of one of the experiments on board was used to provide the appropriate modulating signal. A design assurance test was conducted to establish that the rocket motor could withstand the more severe Delta 1604 environment as well as the boot cutback. (Ref. 7.) Flights were both successful, instrumentation measurements agreeing closely with those made on the ground.

Pioneer

Pioneers F and G were launched from ETR using Atlas Centaurs with a Delta spin table and TE M 364-4 as a third stage for injection into a swingby orbit of Jupiter and the outer solar system planets before leaving the solar system (Fig. 11). Pioneer F (10) was launched on March 3, 1972 and entered the encounter phase with Jupiter on Dec. 3, 1973. Pioneer G (11) was launched on April 6, 1973 and is scheduled to encounter Jupiter around Dec. 2, 1974. Figure 12 shows a sketch of the Pioneer spacecraft as mounted on third stage. The experiments whose location entailed interface problems are shown in Fig. 13. Note the location of instruments on the periphery thus posing potential problems due to exposure to radiation from the spin rockets or the third stage rocket motor. Employed as the third stage rocket motor was Thiokol's TE 364-4 motor, an elongated spherical motor developed from Surveyor technology (Fig. 14). Among the associated interface problems were: (1) excessive radiation to sensitive instruments, (2) possible contamination from the exhaust gases of the third stage motor, either from plume backscatter or from the normal plume expansion in a vacuum, (3) possible spinup exceeding predicted spin rates thus affecting outboard experiments.

Interface problems were solved by both testing and design changes. (Ref. 8.)

Heat flux was obtained from calorimeters placed to measure the radiation from the exhaust plume during



January 7, 1974

Earth - Pioneer 10 Distance –	880.55 Million Km
Pioneer 10 - Jupiter Distance –	29.29 Million Km
Pioneer 10 - Sun Distance –	758.74 Million Km
Earth - Pioneer 11 Distance –	594.52 Million Km
Pioneer 11 - Jupiter Distance –	285.05 Million Km

Figure 11. Pioneer 10 and 11 Positions January 7, 1974

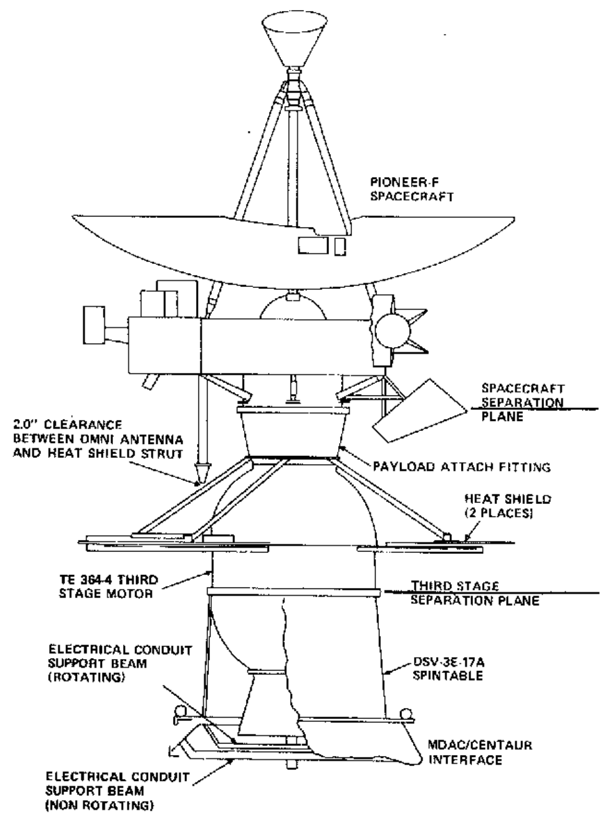


Figure 12. Pioneer-F Third Stage

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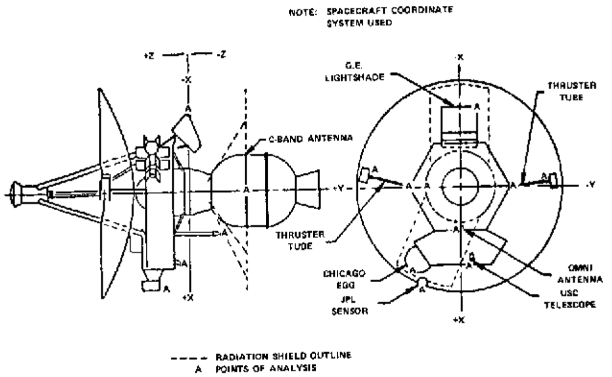

Figure 13. Locations of Experiments

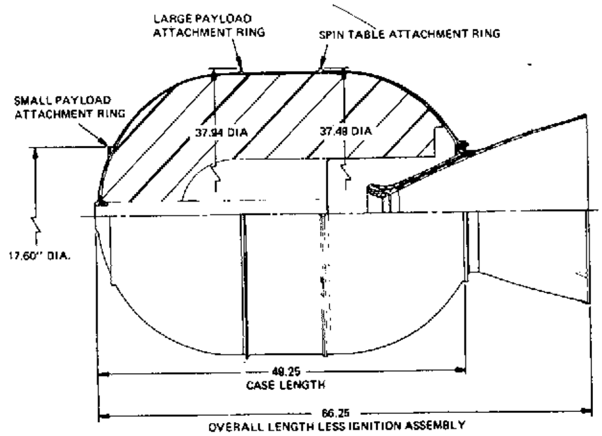

Figure 14. TE M 364-4 Rocket Motor

TE 364-4 static firing tests. As a result of these tests and analysis of the effect on most sensitive instruments, a heat shield was flown on Pioneer F along with measurement instruments to verify the magnitude of the radiation problem. The heat shield employed is shown in Fig. 13. The graph (Fig. 15) shows a comparison of the predicted and actual temperature rise and heat fluxes observed from the Pioneer F flight. As a consequence, Pioneer G flew with no heat shield, nor with instrumentation to measure temperature or heat flux.

Many methods for predicting spin rate increase during solid rocket motor burn have been hypothesized. Pioneer F had been qualified for spin rates as high as 100 rpm with an upper bound of 83 rpm expected in flight. Because of the questionable nature of the explanation for the spinup phenomenon, a nominal spin rate of 58 rpm was assigned to ensure that the upper limit would never be reached. Since spinup predictions had been in the past erroneous and the process for spinup was not fully understood, this estimate was chosen to supply an adequate safety margin. Nevertheless, the predicted spinup effect during burning was never attained, the most plausible explanation being the substitution of a tape wrap nozzle layup for the previously used rosette pattern.

Another interface problem was that of outgassing. Of principal concern was the possibility of the rocket motor acquiring sufficient velocity to overtake the spacecraft while still ejecting exhaust products that could contaminate the optical experiments on board.

Plume backscatter interface problems were resolved by examination of the results of plume backscatter

Table 1 Summary

Application	RAE A, B	IMP H, J	Pioneer F, G
Contract No.	NAS 5-9336	NAS 7-801	NAS 7-678
Motor	TE M 479	TE M 521	TE 364-4
Manufacturer	Thiokol	Thiokol	Thiokol
Impulse, lb-sec	44,500	71,000 to 71,700	650,000
Case Diameter, inches	17.4	17.4	37
Max Thrust, lb	3000	5000	17,100
Burn Time, sec	20	20	44
Spin, rpm	75	100	150
Thrust Rise Rate, lb/sec	—	<210,000	—
Ignition Time, sec	0.010	0.025 to 0.250	0.250
Shock	—	350 g's for 0.4 ms	1000 g's for 1 ms
Acceleration (a) Axial	15 g's	25 g's	17 g's
Acceleration (b) Lateral	2 g's	2 g's	3 g's
Max Weight, lbs	175	278	2472
Nominal Propellant Wt, lbs	154	247	2290
Propellant Mass Fraction	>0.88	>0.90	>0.92
Burst to MEOP	1.5	1.5	1.25
Balance, Static, in.-lb	0.37* 2.5	0.37* 2.5	—
Balance, Dynamic, lb-in^2	3.12* 12.5	3.12* 12.5	—
Temperature, °F	40° to 100°	0° to 110°	40° to 100°
Thrust Misalignment, Angular	0.002 radians	0.002	0.02
Thrust Misalignment, Radial	0.010"	0.010"	0.007"
Residual Magnetic Field			
Post 15 g exposure	18" 36"	18" 36"	—
and 50 gauss deperm	<2γ <1/4γ	<2γ <1/4γ	—
Other			300 days space aging

*Prior to loading with propellant.

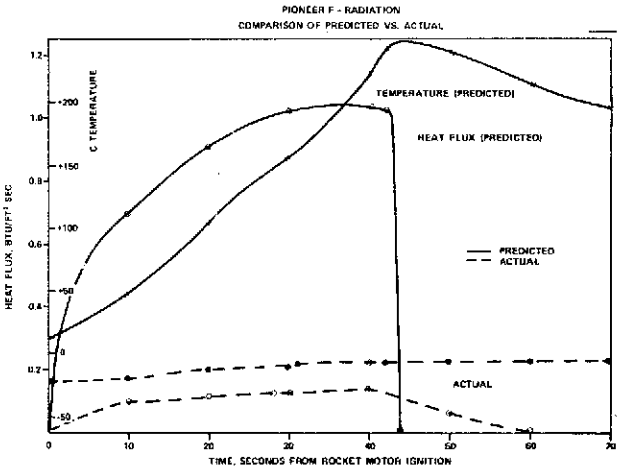

Figure 15.

measurements made at JPL. These results indicated that the magnitude of the effect was within the levels that could be tolerated.

A summary of the main features of the rocket motors employed for the above missions is shown in Table 1.

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