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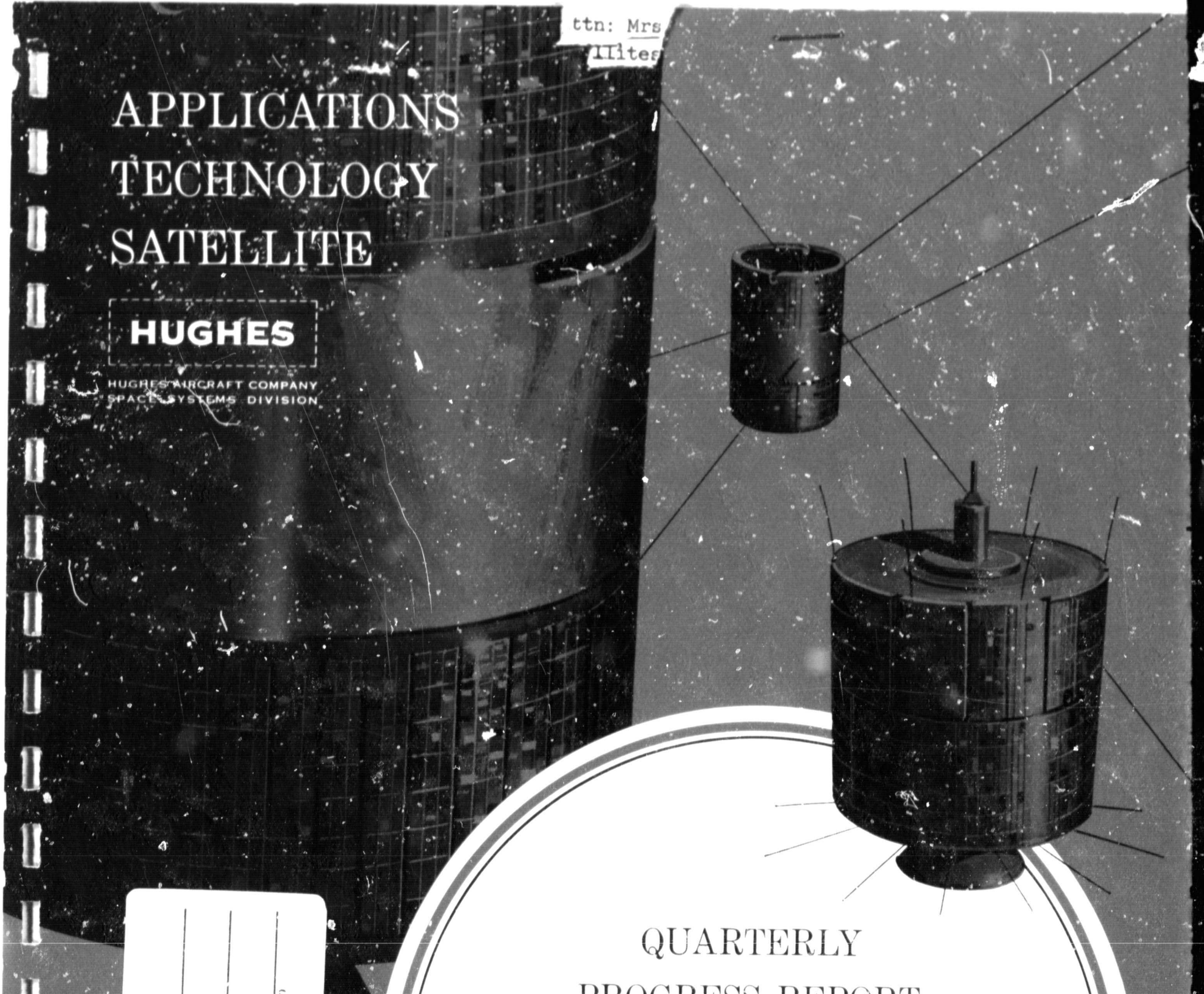
# APPLICATIONS TECHNOLOGY SATELLITE

**HUGHES**

HUGHES AIRCRAFT COMPANY
SPACE SYSTEMS DIVISION



# QUARTERLY PROGRESS REPORT NUMBER 18

1 September through 30 November 1968

NASA Contract NAS 5-382

SSD 80523R

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# APPLICATIONS TECHNOLOGY SATELLITE

## QUARTERLY PROGRESS REPORT NUMBER 18

1 September through 30 November 1968

NASA Contract NAS 5-3823

HUGHES
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SPACE SYSTEMS DIVISION

SSD 80523R

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# 1. PROGRAM SUMMARY

The Applications Technology Satellite (ATS) spacecraft program provides a relatively large, adaptable payload capability designed to achieve long life in circular, medium-altitude orbits or in the synchronous, equatorial orbit. Gravity-gradient stabilization maintains vehicle orientation in the medium-altitude orbit, and either gravity or spin stabilization, depending on mission objectives, is used for vehicle orientation in the synchronous orbit. Table 1-1 shows the basic spacecraft parameters and payloads.

The ATS development and launch program provides for the fabrication of basic spacecraft and system test equipment, using the techniques and subsystems developed under the Advance Technological Development Program (NAS 5-2797). Appropriate combinations of equipment are integrated with the basic spacecraft to complete the flight vehicles. The planned program includes vehicle and subsystem developmental design verification testing and flight acceptance testing of each flight vehicle. In addition, launch center operations and mission operations in support of the vehicle launches and initial orbital operations are included.

On 6 December 1966 at 10:12 a.m. EST, the first spacecraft (ATS-1) was successfully launched from Cape Kennedy. On the second apogee, the apogee motor was fired, placing the spacecraft into its planned synchronous orbit. At approximately 12 noon on 15 December 1966, the satellite was on station over the Pacific Ocean. To date, the spacecraft and all on-board experiments have been operating successfully.

On 5 April 1967, the second spacecraft (ATS-2) was launched from Cape Kennedy. The Agena launch vehicle successfully established a transfer ellipse to the intended 600-n. mi. apogee. However, the Agena failed to provide a second burn at apogee. As a result, the spacecraft is in a highly elliptical orbit (perigee 100 n. mi. ) instead of the circular orbit required for proper performance of the gravity-gradient stabilization system. All subsystems and experiments on the spacecraft have been turned on and are operating properly.

On 5 November 1967 at 6:36 p.m. EST, the third spacecraft (ATS-3) was successfully launched from Cape Kennedy. At 10:37 a.m. EST on 6 November, the apogee motor was fired, placing the spacecraft into its planned synchronous orbit.

On 10 August 1968, the fourth spacecraft (ATS-4) was launched from Cape Kennedy. The Centaur launch vehicle failed to provide a second burn at the second equatorial crossing (first ascending node). As a result, the spacecraft was never separated from the Centaur vehicle and remained in a low-altitude parking orbit (110 by 413 n. mi. with a 26-degree inclination). The ATS-4/Centaur vehicle re-entered the atmosphere on 17 October 1968. All subsystems had been in operation on the spacecraft with no known anomalies.

The remaining launch sequence consists of one synchronous-altitude gravity-gradient stabilized (S/G) launch, ATS-E.

# TABLE 1-1. BASIC SPACECRAFT PARAMETERS

| Physical Configuration | S/G<br>56-inch-diameter Cylinder | S/S<br>57.6-inch-diameter Cylinder | M/G<br>56-inch-diameter Cylinder |
|---|---|---|---|
| Weight | 1730 pounds at launch; 775 pounds in 24-hour equatorial orbit | 1625 pounds at launch; 775 pounds in 24-hour equatorial orbit | 702 pounds at launch and in orbit |
| Apogee motor | Scaled JPL Starfinder | Scaled JPL Starfinder | N/A |
| Control systems | 5-pound thrust $N_2H_4$ system to get on station<br>$5 \times 10^{-4}$ pound thrust subliming solid system (inversion)<br>$10^{-5}$ pound thrust resistojet system (east-west stationkeeping) | 5-pound thrust $H_2O_2$ system to get on station<br>5-pound thrust $H_2O_2$ system for station-keeping east-west and north-south<br>5-pound thrust hydrazine (in addition to $H_2O_2$) for stationkeeping on ATS-C | $5 \times 10^{-4}$ pound thrust subliming solid system for inversion |
| Electrical power | N/P solar cell array<br>130 watts initial<br>Two 6 amp-hr batteries (22 cells each) | N/P solar cell array<br>175 watts initial<br>Two 6 amp-hr batteries (22 cells each) | N/P solar cell array<br>130 watts initial<br>Two 6 amp-hr batteries (22 cells each) |
| Telemetry | Four 2.1-watt transmitters<br>2 at 136.470 MHz<br>2 at 137.350 MHz<br>Two 6 amp-hr batteries (22 cells each) | Four 2.1-watt transmitters<br>2 at 136.470 MHz<br>2 at 137.350 MHz<br>Two encoders; GSFC PCM standard | Four 2.1-watt transmitters<br>2 at 136.470 MHz<br>2 at 137.350 MHz<br>Two encoders; GSFC PCM standard |
| Command | Two receivers at approximately 150 MHz<br>Two decoders; GSFC FSK standard | Two receivers at approximately 150 MHz<br>Two decoders; GSFC FSK standard | Two receivers at approximately 150 MHz<br>Two decoders; GSFC FSK standard |
| Communications | Two triple-mode repeaters<br>4-watt TWT power amplifier<br>18-dB receiving antenna<br>18-dB transmitting antenna<br>6301, 6212 MHz ground to spacecraft<br>4195, 4120 MHz spacecraft to ground | Two triple-mode repeaters<br>4- or 12-watt TWT power amplifier<br>8- or 18-dB receiving antenna<br>18-dB transmitting antenna<br>6301, 6212 MHz ground to spacecraft<br>4195, 4120 MHz spacecraft to ground | Two triple-mode repeaters<br>4-watt TWT power amplifier<br>10-dB transmitting and receiving antenna<br>6301, 6212 MHz ground to spacecraft<br>4195, 4120 MHz spacecraft to ground |
| Payloads | 1<br>Ion engine<br>Gravity-gradient stabilization and instrumentation<br>Image orthicon camera<br>2<br>Gravity-gradient stabilization and instrumentation<br>Ion engine<br>EME<br>Millimeter wave<br>L-band repeater<br>Solar damage experiment | 1<br>Environment measurement<br>VHF repeater<br>Resistojet<br>Spin-scan cloud camera<br>Nutation experiment<br>2<br>Mechanical despun antenna<br>Resistojet<br>Image dissector camera<br>Self-contained navigation experiment<br>Reflectometer<br>Multicolor spin-scan cloud camera<br>VHF repeater<br>Plume temperature measure experiment<br>Solar cell damage experiment<br>Coherent source Faraday rotation measurement | Environmental measurements experiment<br>Albedo experiment<br>Meteorological package<br>Gravity-gradient stabilization and instrumentation |

## 2. SUBSYSTEM DEVELOPMENT AND FABRICATION

### SPINUP SYSTEM

The flight acceptance test on the ATS-E spinup system has been successfully completed and installed on the spacecraft.

### DESPIN SYSTEM

A review of the ATS-D despin data indicates that the spin period of the spacecraft and attached booster changed from 9.4 to 9.8 seconds when the first-stage despin bobs were deployed. The predicted change was from 9.4 to 10.1 seconds.

Possible mechanical failure of components that would allow only one of the despin bobs to deploy has been investigated. No failure mode has been found which would provide a believable explanation to the data.

An analysis is being performed to determine whether the initial velocities induced in the bobs by the temperature compensator springs and the released mechanism could account for the lack of agreement between the predicted and the actual spin period change. These added velocities would cause the bobs to experience a series of "jerks" during deployment resulting in a final spin rate possibly differing from that normally expected. A digital program has been initiated in order to study this phenomenon.

At the start of the report period, fabrication of the despin system was completed to the existing configuration, except for final adjustment of the bob assembly (on "hold" by Engineering). At the close of the report period, the status had not changed. Also, there are indications that changes may be required to the configuration.

### REACTION CONTROL SYSTEMS

#### Hydrogen Peroxide ($H_2O_2$) Reation Control System

On 22 November, the ATS-3 hydrogen peroxide system was activated to determine system function. Ten maneuvers with 200 pulses each were commanded. There was no evidence of system operation. Additional maneuvers are being considered for determining the amount of residual propellant in the peroxide reaction control system.

## Hydrazine ($N_2H_4$) Reaction Control Systems (RCS)

The installation of the $N_2H_4$ RCS S/N F002 in the ATS-E structure was completed. The system is presently dry and under a blanket pressure of 50 psi. No additional effort is required until the system is serviced with referee propellant prior to spacecraft systems vibration testing. As part of the ATS-E support operations, the T-6 test vehicle was serviced with referee propellant.

Depletion of the residual hydrazine in the ATS-4 spacecraft was implemented. Nine steady-state firings were commanded to the radial motor with a total burn time of 41 minutes. During the last few firings, there was evidence of tank unporting, permitting both hydrazine and pressurant gas to flow to the motor. System pressure at the conclusion of the firing sequence was down to 56.4 psia.

The report period marks the completion of 1 year operation of the hydrazine RCS on ATS-3 with all functions of the system being satisfactory. At present, 16,902 pulses have been accumulated by the radial thruster with no apparent degradation of performance. The axial thruster has been pulsed 280 times.

The $N_2H_4$ servicing cart sustained minor damage in transit from AFETR back to Hughes. The cart is currently undergoing repairs. The scales will be replaced, components cleaned, reassembled, and certified.

## Long-Term Storage Test of Quality Assurance System A Hydrazine ($N_2H_4$)

A long-term storage test of the QA system A, modified for $N_2H_4$ service in accordance with DP 30929-001, was satisfactorily terminated on 21 August 1968. Following completion of system matchmate firing tests on 22 June 1967, the system was reserviced to 31.5 pounds of $N_2H_4$ and pressurized to 120 psig with 1 percent helium - 99 percent $N_2$ gas. The system was then to have remained in an uncontrolled environment for an indefinite period of time. However, in October 1967, the system was subjected to a previously reported series of low inlet pressure firings on the axial and radial motors down to a minimum system pressure of 10 psig. After these tests and without reservicing (less than 5 pounds were required for the tests), the system was repressurized to 120 psig and the long-term storage test was resumed. Throughout the initial 4-month and the subsequent 10-month storage periods, there was no evidence of system pressure change when corrected for changes in temperature. This lack of pressure change indicates proper utilization of compatible materials of construction that have remained leakage-free over the ambient storage environment. The similar performance of the ATS-3 $N_2H_4$ RCS in orbit for the past 12 months serves to corroborate these findings. The QA system A hydrazine was offloaded on 21 August 1968 and an offload particle count sample of 1000 milliliters was taken without agitating the RCS. Correcting the sample to 100 milliliters produced the results shown in Table 2-1.

TABLE 2-1. PARTICLE COUNT RESULTS FROM OFFLOAD SAMPLES OF QA SYSTEM A $N_2H_4$

| Range, microns | Specification Maximum | Results for 100-Milliliter Sample |
|---|---|---|
| 5 to 10 | 5000 | 60 |
| 10 to 25 | 1500 | 34 |
| 25 to 50 | 750 | 10 |
| 50 to 100 | 250 | 13 |
| 100 to 200 | 15 | 2 |
| Over 200 | 0 | 0 |
| Metals over 100 | 0 | 0 |
| Fibers | - | 1 |

Immediately after offloading, the system was vacuum-dried and pressurized to 50 psig nitrogen storage blanket pressure in preparation for storage.

## THERMAL CONTROL

### ATS F-1/F-3 Flight Data Comparison and Analysis

A study was initiated to determine the causes of the large temperature differences observed between the ATS F-1 and ATS F-3 flight data. It had been expected, since the vehicles were almost indentical in both design and mission, that the temperatures recorded during flight would be similar. This was not the case and the F-3 flight data indicated that some major change to the spacecraft had occurred which significantly affected the overall vehicle heat balance. Examination of the flight data and the results of a relatively simple thermal model of the vehicle led to the conclusion that some portion (approximately 25 percent or one segment) of the forward thermal barrier had been dislocated from its prescribed configuration. The resulting opening in the $\theta$ = 270-degree area exposed portions of the spacecraft to a direct view of space causing these areas to run colder, thus establishing a gradient across the vehicle.

The ATS F-3 is a spin-stabilized vehicle designed for synchronous orbit operation. Thermal control is accomplished by passive techniques using conduction and thermal radiation to establish an overall heat balance with the spacecraft's environment. ATS F-3 was identical to ATS F-1 with the exception of particular pieces of electrical equipment. These equipment changes did not affect the total heat balance of the vehicle. Early mission flight data (winter solstice) did not give a clear indication of the problem; however, shortly after equinox, it became apparent that the vehicle had undergone some change which disturbed the thermal design. It appeared that the problem, which manifested itself as low temperatures and a gradient across the diameter of the vehicle, was located in the forward area. Since F-1 and F-3 had the same complement of temperature sensors, with the exception of the two located on the forward thermal barrier, a point-by-point comparison of flight data obtained from the two vehicles could be made. Table 2-2 presents this comparison for the sensors located in the forward area. All F-1 data were within the predicted ranges, and no abnormal gradients were recorded. During winter solstice, with the sun shining on the forward end of the vehicle, little difference was noted between the F-1 and F-3 data. However, close examination showed that a gradient had been established across the forward bulkhead with the temperature in the $\theta$ = 270-degree area running about 9 degrees colder than the temperature in the $\theta$ = 90-degree area. (Compare TC77 to TC78). During equinox and summer solstice, sensors showed lower readings for every point as well as confirming the 7° to 10°F gradient across the forward bulkhead. It is interesting to note that the sensor on the thermal barrier at $\theta$ = 270 degrees completely drops off scale at equinox, a further indication that the problem was associated with this piece of hardware.

Two possible occurrences were hypothesized that could result in temperature changes of the magnitude recorded. The first of these was a localized blackening of the thermal barrier caused by plume impingement during the firing of the apogee motor. This blackening would significantly change the thermal surface properties of the barrier resulting in an increase in both the solar absorptance ($\alpha$) and the IR emittance ($\epsilon$). The second possible occurrence was the dislocation of one or more of the four segments of the thermal barrier, thus providing a direct view to space for some portions of the internal forward area. The effect of the removal of a segment of the thermal barrier would be to increase the heat that would be radiated to space across this segment's interface by at least 100 percent.

To attempt to determine the validity of either of these hypotheses, a simple thermal model was constructed which contained the main controlling parameters that determine the thermal balance for the spacecraft. To establish a baseline, an attempt was made to reasonably match the F-1 data. This was accomplished with only minor perturbations to the original vehicle design parameters. Table 2-3 presents the results of the final match of the F-1 flight data with the F-1 thermal model results. Only one model result falls outside the desired range determined by the flight data. This is the prediction for the thrust tube interior (apogee motor inert) during equinox and is due to the inaccuracies in modeling the thermal insulation that surrounds the apogee motor. Due to its isolation, the effect of this area on other areas within the spacecraft is negligible, and the results are included here only for completeness.

A word of explanation is necessary concerning the average forward solar panel temperature which is based on the flight data. Due to sensor location, the recorded temperatures for the forward panel do not directly reflect an average temperature. A nonlinear axial gradient exists in the forward panel due to its extended nature, and a method for establishing the

## TABLE 2-2. COMPARISON OF ATS F-1 AND ATS F-3 FLIGHT DATA, FORWARD AREA OF VEHICLES

| Description | Temperatures, °F | | | | | |
|---|---|---|---|---|---|---|
| | 65-Degree Sun Angle, Winter Solstice | | 90-Degree Sun Angle, March Equinox | | 115-Degree Sun Angle, Summer Solstice | |
| | F-1 | F-3 | F-1 | F-3 | F-1 | F-3 |
| Forward bulkhead, θ = 35 degrees, TC 77 | 74/82 | 72/78 | 58/65 | 48/54 | 56/62 | 45/51 |
| Forward bulkhead, θ = 260 degrees, TC 78 | 74/82 | 63/69 | 58/65 | 41/46 | 56/62 | 37/41 |
| Forward solar panel area 2 temperature, θ = 34 degrees station 49" | 76/84 | 76/84 | 56/64 | 28/36 | No data | 7/15 |
| Forward solar panel area 4 temperature, θ = 188 degrees, station 35" | 66/74 | 62/70 | 55/63 | 33/41 | 45/51 | 19/27 |
| Axial jet valve, θ = 90 degrees, TC 75, hydrazine ($H_2O_2$ F-1) | 70/80 | ~75 | 55/65 | ~54 | 40/48 | ~39 |
| Axial jet valve, θ = 270 degrees, TC 76, $H_2O_2$ (both vehicles) | 70/80 | 66/75 | 55/65 | 45/54 | 40/48 | 26/34 |
| Forward thermal barrier, θ = 90 degrees, TC 89* | -- | ~96 | -- | ~24 | -- | ~6 |
| Forward thermal barrier, θ = 270 degrees, TC 92* | -- | ~112 | -- | < 0 | -- | < 0 |

*No temperature sensor in this area on ATS F-1.

TABLE 2-3. COMPARISON OF ATS F-1 FLIGHT DATA AND ANALYTICAL RESULTS FROM ATS F-1 THERMAL MODEL

| Description | Temperatures, °F | | | | | |
|---|---|---|---|---|---|---|
| | 65-Degree Sun Angle, Winter Solstice | | 90-Degree Sun Angle, March Equinox | | 115-Degree Sun Angle Summer Solstice | |
| | F-1 | Analysis | F-1 | Analysis | F-1 | Analysis |
| Center bay area (tanks) | 60/72 | 68 | 52/64 | 63 | 60/72 | 65 |
| Rib area (TWTs) | 50/74 | 68 | 52/77 | 65 | 77/100 | 80 |
| Thrust tube interior (apogee motor inert) | 32/42 | 37 | 16/26 | 33* | 33/43 | 41 |
| Aft solar panel | 56/64 | 63 | 60/68 | 62 | 59/67 | 59 |
| Forward bulkhead, 0/180-degree segment (TC 77) | 74/82 | 76 | 58/65 | 62 | 56/62 | 59 |
| Forward bulkhead, 180/360-degree segment (TC 78) | 74/82 | 76 | 58/65 | 62 | 56/62 | 59 |
| Forward solar panel (area 2 temperature) | 76/84 | 71 | 56/64 | 58 | No data | 49 |
| Forward solar panel (area 4 temperature) | (75)←<br>66/74 | Average **<br>71 | →(60)←<br>55/63 | Average **<br>58 | →(45)<br>45/51 | 49 |
| Forward thermal barrier, 0/180-degree segment*** | -- | 87 | -- | 52 | -- | 43 |
| Forward thermal barrier, 180/360-degree segment*** | -- | 87 | -- | 52 | -- | 43 |
| Antenna area | 28/42 | 38 | 36/52 | 52 | 85/99 | 95 |

* Analytical result outside of flight data range (16° to 26°F) due to inaccurate modeling of motor thermal insulation.

** Calculated forward solar panel average temperature based on flight data. Model designed to predict this average.

*** No temperature sensor in this area on ATS F-1.

average temperature was used which had previously been formulated for solar panel studies. A tolerance equal to half the total flight data range (±4°F for the F-1 data) is applied to the average. The thermal model was designed to predict average temperatures due to the coarseness of its nodal breakup. Therefore, model results falling within the prescribed range determined by the average and its tolerance establish an adequate match.

Plume impingement was first investigated as the possible cause of the problem. One set of computer runs for the three sun angles eliminated this from further consideration. It was assumed that the barrier was blackened over 50 percent of its total area and that the blackening was concentrated all in $\theta$ = 180 to 360-degree segment. For the equinox and summer solstice conditions, the results showed a gradient of 10° to 15°F with absolute temperatures approximately 5°F above the flight data range for the $\theta$ = 90-degree area of the forward bulkhead. Most other temperatures fell within the desired ranges. For the winter solstice condition, a gradient of 25°F resulted on the forward bulkhead with most internal temperatures being 5°F or more above the flight data ranges. Also during this period, the model predicted that the blackened area of the barrier would reach a temperature of 155°F. Examination of these results showed that it would be necessary to assume more blackened area (increase IR emittance) to give the desired match with the flight data for the equinox and summer solstice conditions and less blackened area (decrease the predominance of the sun load) to obtain a match with the winter solstice data. These two opposing requirements are obviously incompatible.

The second hypothesis of the dislocation of a segment of barrier was then examined. The assumption was made that 25 percent (equivalent to one segment of the forward thermal barrier) had been removed leaving openings which provided portions of the spacecraft over the $\theta$ = 180- to 360-degree segment of the barrier for modeling purposes (a large number of pinholes). The results of the analysis are presented in Table 2-4. A good match was obtained for most internal areas of the vehicle with the exception of the internal thrust tube (apogee motor inert) which again experienced inaccuracies in the modeling of the thermal insulation. During winter solstice, the resulting bulkhead predictions were 5° to 10°F low; however, the desired gradient was established. Reducing the view to space from the bulkhead would remedy this problem. Forward solar panel temperatures show a fair match with the flight data during winter solstice and equinox and a very poor match for summer solstice. (Compare average of analytical results for the two forward solar panel nodes with the average solar panel temperature based on the flight data.) Exposing the forward panels to a larger view of space would allow more illumination during winter solstice (raising the temperature) and greater radiation capabilities during equinox and summer solstice (lowering the temperature). Although these two corrective actions for obtaining the desirable match for the bulkhead and forward solar panel temperatures with the respective flight data points seem to be contradictory, in actuality they are not and can be obtained by assuming different distributions of the openings. The initial assumption of equally distributed openings over the entire $\theta$ = 180- to 360-degree segment of the barrier could not physically occur, but model limitations forced this simplifying assumption. A good match for the thermal barrier could not be

## TABLE 2-4. COMPARISON OF ATS F-3 FLIGHT DATA AND ANALYTICAL RESULTS FROM ATS F-3 THERMAL MODEL

| Description | Temperatures °F | | | | | |
|---|---|---|---|---|---|---|
| | 65-Degree Sun Angle, Winter Solstice | | 90-Degree Sun Angle, March Equinox | | 115-Degree Sun Angle, Summer Solstice | |
| | F-3 Data | Analysis | F-3 Data | Analysis | F-3 Data | Analysis |
| Center bay area (tanks) | 54/65 | 63 | 49/58 | 57 | 53/62 | 58 |
| Rib area (TWTs) | 35/75 | 65 | 33/76 | 61 | 62/96 | 77 |
| Thrust tube interior (apogee motor inert) | 34/44 | 33* | 15/25 | 28* | 40/50 | 36* |
| Aft solar panel | 51/61 | 61 | 52/61 | 59 | 53/62 | 56 |
| Forward bulkhead, 0/180-degree segment (TC 77) | 72/78 | 67** | 48/54 | 50 | 45/51 | 46 |
| Forward bulkhead, 180/360-degree segment (TC 78) | 63/69 | 60** | 41/46 | 41 | 37/41 | 38 |
| Forward solar panel (area 2 temperature) | 76/84 | 63** | 28/36 | 47** | 7/15 | 37** |
| Forward solar panel (area 4 temperature) | (73)←<br>62/70 | Average***<br>68** | →(36)←<br>33/41 | Average***<br>37** | →(19)<br>19/27 | 26** |
| Forward thermal barrier 0/180-degree segment | ~96 | 82** | ~24 | 38** | ~6 | 28** |
| Forward thermal barrier 180/360-degree segment | ~112 | 81** | < 0 | 34** | < 0 | 24** |
| Antenna area | 29/49 | 36 | 32/60 | 49 | 83/100 | 93 |

* Analytical results outside of flight data ranges due to inaccurate modeling of motor thermal insulation.

** Analytical results outside of flight data ranges due to model simplifications and inaccuracies.

*** Calculated forward solar panel average temperature based on flight data. Model designed to predict this average.

obtained due to analytical simplifications which predict an average temperature for the entire barrier rather than the temperature at the flight data point. Even with these shortcomings, the model did produce a good match for most of the internal bulk nodes of the vehicle, thus providing a good degree of confidence in the results.

After the results presented in Table 2-4 were obtained, further analysis was not attempted. As outlined above, several areas of the model were constrained due to the analytical simplifications. Further analysis using the present model would have only led to further uncertainties due to a lack of model detail. The next step would naturally be to construct a more complete and complicated thermal model which would break down areas of concern into a finer nodal description. Due to the cost, time involved in such an undertaking, and uncertainty in results, it was not considered advisable to pursue the matter further. From a technical point of view, the value of further analysis is questionable since verification of additional findings, such as the exact orientation of the dislocated portion of the barrier, cannot be made. It is also doubtful that corrective action could be defined.

## Preliminary Thermal Input to ATS-E Flight Acceptance Test

Table 2-5 presents a list of recommended thermocouples for the ATS-E flight acceptance test. These thermocouples will verify the overall thermal performance of the vehicle as well as the individual performance of new experiments and some selected items that are particularly thermally sensitive.

### Recommended Test Procedures

During one of the short-form test periods, total internal power dissipation should be maintained at some nominal high-power value (135 to 145 watts) for at least 30 hours. This will provide adequate verification of the overall spacecraft thermal performance. Either during the above period or during some other periods of the test program, selected units should be operated for a sufficient period of time to reach a quasi-thermal equilibrium over their surroundings. Two to 4 hours should be sufficient. For those units most sensitive to spacecraft orientation, this should be during their periods of solar illumination. The EME and the millimeter-wave experiment fall into this category. Some judgment will be required in order to fully accomplish these unit operations, but this program and test data should be sufficient to establish both overall spacecraft performance and individual unit performance.

### ATS-E Flight Thermistors - Update

Table 2-6 updates the flight thermistor requirements for ATS-E previously presented. Only one location change has been made: thermocouple 33 has been relocated to the magnetic field monitor mounting surface on the exterior of the forward bulkhead. All the blanks that previously existed have been filled in, thus completing the table.

TABLE 2-5. RECOMMENDED THERMOCOUPLE LIST FOR ATS-E

| Item | Quantity |
|---|---|
| Hydrazine | 4 |
| EME | 4 |
| Millimeter-wave | 3 |
| Switching converter | 1 |
| Solar panels | 2 |
| Bellyband | 2 |
| Batteries | 4 |
| Bus voltage limiters | 2 |
| Spares | 2 |
| Total | 24 |

## POWER SUBSYSTEMS

### ATS-E Power Subsystem

All units were processed through quality control final inspection and installed on the spacecraft with the exception of payload power switch 475320-100, F-1. This payload power switch was modified subsequent to the major control item test.

### ATS-E Payload Power Switch

A waiver was requested and received from NASA authorizing environmental testing of payload power switch 465320-100, F-1 as follows:

1) Thermal testing at 40°F for 4 hours followed by 100°F for 4 hours, one cycle only.

2) Vibration testing to NASA Specification S2-0102, Table III (three axis, sine). (Reference: NASA Letter, TW-125, from A. D. Brown to A. C. Bryant, dated 11 October 1968.)

Thermal testing was completed 18 October, and vibration testing was completed 23 October.

TFR 84133 was written against the payload power switch after the first (X-axis) vibration run. The trip currents were slightly out of specification (19.3 and 19.5 amperes versus the maximum allowable value of 18.2 amperes). The cause was attributed to improper grounding of the test

## TABLE 2-6. ATS-E FLIGHT THERMISTOR REQUIREMENTS

| Nomenclature | TC | Drawing | Location Station | Location θ,degrees | Part No. 9888XXX-X | Thermistor Temperature Range, °F Minimum | Thermistor Temperature Range, °F Maximum | Encoder | Channel | Bias Network |
|---|---|---|---|---|---|---|---|---|---|---|
| Solar panel temperature 1 (forward) | 1 | 475262-125 | 32 | 255 | 627-2 | -85 | 155 | 1 | 29 | S87Y |
| Solar panel temperature 2 (forward) | 2 | 475262-125 | 44 | 255 | 627-2 | -85 | 155 | 1 | 42 | S87Y |
| Solar panel temperature 3 (aft) | 3 | 475263-125 | -13.5 | 274 | 627-2 | -85 | 155 | 2 | 29 | S87Y |
| Solar Panel temperature 4 (aft) | 4 | 475263-125 | -4.5 | 180 | 627-2 | -85 | 155 | 2 | 42 | S87Y |
| (Deleted) | 5 | | | | | | | | | |
| (Deleted) | 6 | | | | | | | | | |
| Center cylinder thermal barrier | 7 | 3041913 | 17.2 | 27 | 639-1 | -265 | 250 | 1 | 30 | S82Y |
| Center cylinder thermal barrier | 8 | 3041915 | 17.2 | 205 | 639-1 | -265 | 250 | 2 | 30 | S82Y |
| Apogee motor mount bracket | 9 | 3041916 | 1 | 118 | 606-3 | 0 | 500 | 1 | 27 | S80Y |
| Apogee motor mount bracket | 10 | 3041916 | 1 | 298 | 606-3 | 0 | 500 | 2 | 27 | S80Y |
| TWT 1 and 2 structure | 11 | 3045846 | 25.5 | 340 | 627-1 | 0 | 140 | 1 | 62/6 | S84Y |
| TWT 1 and 2 structure | 12 | 3045846 | 16.5 | 340 | 627-1 | 0 | 140 | 2 | 62/6 | S84Y |
| TWT 3 and 4 structure | 13 | 3045846 | 25.5 | 220 | 627-1 | 0 | 140 | 1 | 62/21 | S84Y |
| TWT 3 and 4 structure | 14 | 3045846 | 16.5 | 220 | 627-1 | 0 | 140 | 2 | 62/21 | S84Y |
| Gravity-gradient TV electronics mounting | 15 | 3041916 | 29.0 | 10 | 627-1 | 0 | 140 | 2 | 62/20 | S84Y |
| Bus limiter mounting #3 | 16 | 3045846 | 8 | 340 | 627-1 | 0 | 140 | 1 | 63/18 | S84Y |
| ATC temperature 1 (forward) | 17 | 3045846 | 21 | 150 | 627-1 | 0 | 140 | 1 | 62/3 | S84Y |
| ATC temperature 2 (aft) | 18 | 3045846 | 13 | Z-axis | 627-1 | 0 | 140 | 1 | 62/20 | S84Y |
| ATC temperature 3 (forward) | 19 | 3045846 | 21 | 150 | 627-1 | 0 | 140 | 2 | 62/3 | S84Y |
| Antenna electronics structure | 20 | 3045846 | 25.5 | Z-axis | 627-1 | 0 | 140 | 2 | 62/43 | S84Y |
| TM encoder 2 | 21 | 3041916 | 15 | 40 | 627-1 | 0 | 140 | 1 | 62/2 | S84Y |
| TM encoder 1 | 22 | 3041916 | 19 | 220 | 627-1 | 0 | 140 | 2 | 62/2 | S84Y |
| Battery temperature 1 (4 cell, bus 1) | 23 | 3041916 | 8 | 220 | 627-1 | 0 | 140 | 1 | 62/50 | S84Y |

Table 2-6 (continued)

| Nomenclature | TC | Drawing | Location |  | Part No. 9888XXX-X | Thermistor Temperature Range, $^{o}F$ |  | Encoder | Channel | Bias Network |
|---|---|---|---|---|---|---|---|---|---|---|
|  |  |  | Station | θ, degrees |  | Minimum | Maximum |  |  |  |
| Battery temperature 3 (4 cell, bus 2) | 24 | 3041916 | 8 | 70 | 627-1 | 0 | 140 | 2 | 62/50 | S84Y |
| Battern temperature 2 (6 cell, bus 1) | 25 | 3041916 | 25.5 | 285 | 627-1 | 0 | 140 | 1 | 62/4 | S84Y |
| Battery temperature 4 (6 cell, bus 2) | 26 | 3041916 | 10.5 | 220 | 627-1 | 0 | 140 | 2 | 62/4 | S84Y |
| Command receiver 2 structure | 27 | 3045846 | 15.5 | 150 | 627-1 | 0 | 140 | 1 | 62/43 | S84Y |
| EME primary interface | 28 | 3041916 | 16 | 195 | 627-1 | 0 | 140 | 2 | 63/18 | S84Y |
| RCS tank temperature 1 | 29 | 3041916 | 19 | 135 | 627-1 | 20 | 100 | 1 | 63/28 | S85Y |
| RCS tank temperature 2 | 30 | 3041916 | 19 | 135 | 627-1 | 20 | 100 | 2 | 63/28 | S85Y |
| RCS axial jet | 31 | 3041916 | -12 | 255 | 627- | -85 | 155 | 2 | 63/3 | S84Y |
| RCS radial jet | 32 | 3041916 | 13 | 337 | 627- | 0 | 140 | 1 | 63/3 | S84Y |
| Magnetic field monitor* | 33 | 3041916 | 30.5 | 340 | 627-1 | 0 | 140 | 1 | 62/0 | S84Y |
| $N_2$ tank temperature (θ = 140 degrees) | 34 | 3041916 | 8 | 130 | 627-1 | 0 | 140 | 2 | 62/0 | S84Y |
| Thrust tube temperature 1 | 35 | 3045846 | 15 | 225 | 627-1 | 0 | 140 | 1 | 63/9 | S84Y |
| Thrust tube temperature 2 | 36 | 3045846 | 15 | 40 | 627-1 | 0 | 140 | 2 | 63/9 | S84Y |
| EME remote interface | 37 | 3041916 | 29 | 260 | 627-1 | 0 | 140 | 1 | 63/20 | S83Y |
| Millimeter-wave experiment interface | 38 | 3041916 | 15 | 340 | 627-1 | 0 | 140 | 2 | 63/20 | S83Y |
| Heat pipe temperature 1 (forward) | 39 | 475262-125 | 30.5 | 255 | 639-1 | -265 | 250 | 1 | 62/26 | S82Y** |
| Heat pipe temperature 2 (forward) | 40 | 475262-125 | 33.5 | 0 | 627-2 | -85 | 155 | 2 | 62/26 | S83Y |
| Heat pipe temperature 3 (forward) | 41 | 475262-125 | 37 | 96 | 627-2 | -85 | 155 | 1 | 62/35 | S83Y |
| Heat pipe temperature 4 (forward) | 42 | 475262-125 | 39.5 | 180 | 639-1 | -265 | 250 | 2 | 62/35 | S82Y** |
| Heat pipe temperature 5 (forward) | 43 | 475262-125 | 42.5 | 255 | 627-2 | -85 | 155 | 1 | 62/44 | S83Y |
| Heat pipe temperature 6 (forward) | 44 | 475262-125 | 46 | 2 | 627-2 | -85 | 155 | 2 | 62/44 | S83Y |
| Heat pipe temperature 7 (forward) | 45 | 475262-125 | 49 | 96 | 627-2 | -85 | 155 | 1 | 62/51 | S83Y |

Table 2-6 (continued)

| Nomenclature | TC | Drawing | Location Station | Location θ, degrees | Part No. 9888XXX-X | Thermistor Temperature Range, °F Minimum | Thermistor Temperature Range, °F Maximum | Encoder | Channel | Bias Network |
|---|---|---|---|---|---|---|---|---|---|---|
| Heat pipe temperature 8 (forward) | 46 | 475262-125 | 51 | 55 | 639-1 | -265 | 250 | 2 | 62/51 | S82Y** |
| Heat pipe temperature 9 (aft) | 47 | 475263-125 | 3 | 94 | 639-1 | -265 | 250 | 1 | 62/60 | S82Y** |
| Heat pipe temperature 10 (aft) | 48 | 475263-125 | 0 | 357 | 627-2 | -85 | 155 | 2 | 62/60 | S83Y |
| Heat pipe temperature 11 (aft) | 49 | 475263-125 | -3 | 274 | 627-2 | -85 | 155 | 1 | 62/62 | S83Y** |
| Heat pipe temperature 12 (aft) | 50 | 475263-125 | -6 | 180 | 639-1 | -265 | 250 | 2 | 62/62 | S82Y** |
| Heat pipe temperature 13 (aft) | 51 | 475263-125 | -9 | 94 | 627-2 | -85 | 155 | 1 | 62/63 | S83Y** |
| Heat pipe temperature 14 (aft) | 52 | 475263-125 | -12 | 357 | 627-2 | -85 | 155 | 2 | 62/63 | S83Y** |
| Heat pipe temperature 15 (aft) | 53 | 475263-125 | -15 | 274 | 627-2 | -85 | 155 | 1 | 63/14 | S83Y** |
| Heat pipe temperature 16 (aft) | 54 | 475263-125 | -17 | 180 | 639-1 | -265 | 250 | 2 | 63/14 | S82Y** |
| Inversion engine 1, tank | 55 | 475071 | 24 | 164 | 627-1 | 0 | 160 | 1 | 62/8 | S83Y |
| Inversion engine 1, pressure transducer | 56 | 475071 | 24 | 164 | 627-1 | 0 | 160 | 2 | 62/8 | S83Y |
| Inversion engine 1, line | 57 | 475071 | 24 | 164 | 627-1 | 0 | 160 | 1 | 62/54 | S83Y |
| Inversion engine 1, nozzle | 58 | 475071 | 24 | 164 | 627-1 | 0 | 160 | 2 | 62/54 | S83Y |
| Inversion engine 2, tank | 59 | 475071 | 24 | 195 | 627-1 | 0 | 160 | 1 | 62/9 | S83Y |
| Inversion engine 2, pressure transducer | 60 | 475071 | 24 | 195 | 627-1 | 0 | 160 | 2 | 62/9 | S83Y |
| Inversion engine 2, line | 61 | 475071 | 24 | 195 | 627-1 | 0 | 160 | 1 | 62/24 | S83Y |
| Inversion engine 2, nozzle | 62 | 475071 | 24 | 195 | 627-1 | 0 | 160 | 2 | 62/24 | S83Y |
| Apogee motor temperature 1 | 63 | 3046448 | 3 | 0 | 639-1 | 0 | 700 | 1 | 26 | S81Y |
| Apogee motor temperature 2 | 64 | 3046448 | 3 | 0 | 639-1 | 0 | 700 | 2 | 26 | S81Y |

*New location and/or additional thermistor required for ATS-E.

**Bias networks mounted external to SCU on terminal boards.

equipment The trouble was corrected, and subsequent tests were run on all three vibration axes without a discrepancy.

The payload power switch was delivered to the ATS-E spacecraft for installation 25 October after quality control final inspection and NASA buyoff.

ATS-E Switching Converter

The following tasks were performed during this report period:

1) Fabrication of parts for flight unit
2) Parts procurement
3) Quality control screening and certification of parts
4) C&M testing and burn-in of parts
5) Test console modification and checkout
6) Specification preparation
7) Fabrication of inductors, burn-in, and test
8) Preliminary tests on the spacecraft using the breadboard model converter

Purchase requisitions were submitted and stock transfers made on parts for the switching converter A special stores was set up for this unit to ensure that all parts are readily available. Because of schedule limitations, a large number of parts (resistors, for example) that could not be purchased were transferred from Hughes stock Many of these parts had to undergo 168-hour burn-in to qualify them for ATS use. Burn-in of parts is complete through 96 hours for resistors. Test data are under review at the present time to determine whether additional burn-in time will be required.

A preliminary test using the breadboard converter on the Y-1 spacecraft indicated no unusual problems associated with spacecraft systems operation. The ATS-E spacecraft testing of the breadboard model switching converter was completed successfully also. These tests showed that the noise on the spacecraft bus was reduced by the addition of the converter.

The converter design was reviewed 23 September with the Hughes Program Office and again on 25 September with NASA. As a result, the decision was made to use the Darlington circuit (rather than the boost-drive).

A meeting was held with ATS systems engineering on 7 October that resulted in the specification of a maximum noise of 0.500 volt peak to peak and a minimum efficiency of 92.0 percent.

Modification of the test console began 10 October 1968. Completion has been rescheduled for 13 December to allow for minor changes to be made in conjunction with specification preparation.

Final check and release of drawings was delayed by a design change required to eliminate a heat problem that was discovered with one of the transistors in the converter circuit. Release of drawings was completed 26 November.

Fabrication of printed circuit boards is complete. Fabrication of terminal boards, case, cover, chassis, and brackets is under way and should be completed by 13 December 1968.

## ATS-E Solar Panel Development Tests

Two 3 x 20 cell samples successfully completed thermal-vacuum testing as follows:

1) +190° to -200°F, five cycles
2) +190° to -225°F, five cycles
3) +190° to -250°F, five cycles

No visual or electrical damage was evident after inspection.

## Solar Cell Grading

The entire order for 11,200 new solar cells has been received and graded.

## Solar Panel Array Fabrication

All arrays have completed machine soldering and rework, except for several that require retesting.

Solar arrays were essentially completed during this report period. Subcontractor delays in delivery of substrates to Hughes has caused reschedule of in-house work to complete the solar panels. Based on delivery of substrates to Hughes by 27 December 1968, the solar panels will be completed by 14 March 1969.

## Additional Thermistors on ATS-E Solar Panels

Work on solar panel drawings required by added thermistors was completed.

## Heater Blanket

Fabrication of the heater blanket was completed.

## Solar Panel Changes

An ECR and necessary EOs are being processed to cover minor changes in the solar panel fabrication specification and array test specification. The array test specification now takes into account the change in battery charge arrays from 11 cells in series to 12 cells in series and the single-cell open circuit voltage of 0.5328.

### ATS-3 Solar Cell Experiment

Data reduction has been completed for the first 280 days in orbit, and a rough draft of a report has been prepared.

### ATS-E Power System Analysis

A series of curves has been drawn showing various power systems parameters as a function of solar panel voltage for the case where the heat pipes are functioning normally at a temperature of 60° F during equinox. Analysis for other conditions is now under way.

## WIRING HARNESS

### ATS-E Development

All of the ECR package is released with the exception of ECR 397261 (third harmonic generator), 85 percent complete; ECR 397271 (solar cell experiment), 60 percent complete; and ECR 397272 (L-band repeater), which will start 10 January 1969.

Division 27 is preparing an ECR to reposition the power divider 105206-100 to increase the bend radii of the four coax cables to the third harmonic generator.

A wooden mockup of the forward bulkhead was built by Division 27 to determine an optimum routing for the hybrid balun and the coax cables to the third harmonic whips. The routing of the hybrid balun has been resolved; however, the routing of the cables to the third harmonic whips has not been determined.

Engineering liaison support is being maintained during harness fabrication, assembly, rework, checkout, and installation.

### ATS-E Fabrication

The wiring harnesses were completed during September.

The new harnesses for the magnetometer and coax cables for the millimeter wave and third harmonic generator have been completed. The existing harnesses are being reworked on the spacecraft to reflect the ECR package modification.

## HEAT PIPE — SOLAR PANEL PROJECT

### Development Testing

A frozen, hot dry-wick test was successfully conducted to confirm the ability of the heat pipe to recover from the condition of heat application similar to that which will be encountered during spacecraft solar thermal-vacuum testing.

A 3-month period of accelerated life testing began in September at the D. W Douglas Laboratories. In this test, two pipes are being exposed to continuous operation at higher pumping rates than normal and one pipe is being subjected to daily freeze/thaw cycles accompanied by reversal of heat flow.

The D. W. Douglas Laboratories have proposed to increase the period and rate of these tests to simulate a 3-year orbital life of the heat pipes. The proposed increase would extend the test period by 4 months.

### Heat Pipe Fabrication

Production of all test, flight, and spare heat pipes at the D. W. Douglas Laboratories was completed and the Hughes resident inspector has been recalled.

### Substrate Fabrication

#### Forward Panel

One of the short heat pipes at the forward end of the panel was mislocated 0.50 inch circumferentially and would have protruded into the cutout for the libration damper. The shape of the cutout was modified to accommodate this pipe without compromise to the dynamic envelope of the damper.

The inner face of the panel was slightly damaged when it was removed from the layup mandrel. Bits of teflon from the parting sheet removed by a drilling operation while the panel was on the mandrel "balled-up" and caused vertical wrinkles extending from the center of the panel to its lower edge. The resulting core crushing was as much as 0.045 inch locally, but analysis showed it not to be a serious degradation of strength.

As a result of the exposure of this panel to excess curing temperature during bonding, the local NASA representative formally rejected it subject to reconsideration after thermal acceptance testing. The Hughes conditions for unqualified acceptance were satisfied by McDonnell Douglas with a supplementary, hot wet-wick test of a development heat pipe above the overcure temperature of 278° F, and the Hughes lien was lifted.

At a review with NASA personnel on 26 September, NASA agreed to accept Hughes' disposition of the overcure, provided the substrate survived an additional exposure of leakage as well as the usual conditions of acceptance testing and recovery from eclipse.

### Aft Panel

By deleting the drilling operation while the panel was on the mandrel, wrinkling of the inner skin was avoided. In general, the aft panel has fewer manufacturing defects than the forward.

### Spares

Although they were never needed, McDonnell Douglas produced one set of spare heat pipes for either a forward or aft panel. All other components were readily available in local supply had it been necessary to build another assembly.

## Vibration Acceptance Testing

Testing was begun with the intention of vibrating each panel individually on a "hard" vibration test fixture limited to the subcontract specification levels. After the two structural failures described below, Hughes elected, with NASA concurrence, to transfer the vibration testing to Hughes where both panels were vibrated simultaneously on the T-6 vehicle to spacecraft qualification levels.

Testing at McDonnell Douglas was interrupted for 1-1/2 days due to breakage of the omni-antenna test boom. It was replaced with the ATS-E flight boom, which Hughes intend to qualify as a secondary purpose of the forward substrate vibration test.

Testing at McDonnell Douglas was suspended a second time due to the partial unbonding of one ballast weight accompanied by honeycomb core failure directly beneath it. Its location at the interface of the Hughes and McDonnell Douglas responsibilities and the lack of instrumentation in the immediate vicinity combined to make a clear determination of the cause and responsibility for the failure almost impossible. McDonnell Douglas and Hughes eventually split the cost of this failure equally.

The lack of a clear understanding of the cause of this failure led to the adoption of a redesign which was admittedly quite conservative — the attachment of ballast weights by bolting to threaded inserts in the panel structure behind doublers on the inner skin. In this way, the ballast loads are carried directly into the core and outer face sheet, and the inserts and doubler stabilize the panel locally. Attachment of the ballast weights is by four bolts on the same pattern as the studs on the ballast pads, which they now replace.

The local repair of the area damaged in test was performed at the same time as the addition of the threaded inserts and doublers. While in the autoclave for this purpose, McDonnell Douglas attempted to measure the leakage of helium and ammonia from the heat pipes. This seemed to be the simplest way to satisfy the NASA requirement for an additional temperature exposure for acceptance of the earlier overcure.

Unfortunately, the test was inconclusive because ammonia released by the curing foam masked any possible leakage from the heat pipe, and the high absolute pressure of the curing system desensitized the helium detector.

At this time, the decision was made to transfer testing to the T-6 vehicle at Hughes to produce the most realistic simulation of the spacecraft environment and reduce the risk of failure due to inadvertent overtest. McDonnell Douglas retained responsibility for the structural integrity of the substrates except for uninstrumented points and those which exceed the 30 to 35 g limitation below 200 Hz. They had personnel in attendance during all handling and testing of the substrates at Hughes.

The amount of ballast weight on the aft panel during test was reduced from 30 to 15 pounds, reflecting the changed mass properties of the spacecraft. Even so, there was about a 6 X margin over the anticipated flight weight.

During vibration testing on the T-6 vehicle, no attempt was made to control the vibration input levels based on the response of the substrates. The vibration inputs to the vehicle were those resulting from the specified inputs at the Centaur adapter. Since McDonnell Douglas had designed to a specified load limit of 30 g lateral and 35 g longitudinal, they would not accept responsibility for the integrity of the substrate if its response exceeded these values. When test response did exceed these levels at any point, the substrate was examined for damage and analyzed for the safety margin remaining at the observed load value. If the substrate appeared sound after this examination, its continued flightworthiness was jointly certified by McDonnell Douglas and Hughes before NASA would agree to continue testing.

During the vibration testing at Hughes, a crack developed in the omni-antenna boom which resulted in a repair of the boom.

During the longitudinal axis sinusoidal vibration test, structural failures developed in the aft panel at the two bulkhead mounting points in line with the ballast weights ($\theta = 180$ and 212 degrees). Post-test investigation revealed the following significant facts:

1) Previously used bathtub fittings were employed to attach the substrate to the T-6 bulkhead, which required the elongation of three of the mounting holes to properly engage those in the substrate.

2) Four mounting screws were loose in the aft panel and 13 in the forward panel. Two screws in the forward panel had backed completely out.

3) The threaded inserts in the forward panel had their locking features worn away by previous use.

4) Forward panel mounting screws were too short to engage the locking feature.

Following these failures, the forward panel had its threaded inserts replaced and was reinstalled with longer screws, using loctite for additional thread locking assurance. Testing of the forward panel was concluded with the use of a fiberglass aft panel on the T-6 vehicle.

Analysis of the aft panel showed that failure of the mounting points could only be explained by assuming a combination of adverse loads, minimum statistical strength, and poor installation. An improvement in any one factor would presumably have sufficed to ensure structural adequacy. Hughes, however, elected to improve both the local design strength of the panel mounting points and the efficiency of the installation.

The substrate was redesigned and repaired to add 0.060-inch doublers with close tolerance mounting holes and additional core fill at each mounting point. Addition of these doublers tripled the local tearout strength of the panel mounting points.

The installation was redesigned to use new bathtub fittings at all 15 mounting points and to equip them with close tolerance mounting holes fitted on assembly to the substrate and individually drilled to fit the bulkhead. In this way, the ability of all mounting points to carry the load was assured.

With the repairs and redesign described above, the aft panel was successfully retested in the longitudinal axis. Examination of the dynamic response of the panel after test showed it to be essentially identical to that before repair. On this basis, testing in the lateral axes was not repeated.

A chronology of the significant events in the vibration test program is shown in Table 2-7.

Thermal Acceptance Testing

McDonnell Douglas submitted a test plan that relies on a prior heat balance test to establish the value of heater panel emissivity, rather than the use of an optical emissometer, which Hughes prefers. While the method is acceptable to Hughes and NASA, Hughes was not a party to the tests that established the values and therefore asked McDonnell Douglas for some assurance of their validity.

McDonnell Douglas has agreed to perform a heat balance check during the acceptance test to confirm the emissivity values. They will rerun the previous tests if these results do not agree within 7 percent.

## TABLE 2-7. ATS-E HEAT PIPE SUBSTRATES VIBRATION TEST CHRONOLOGY

| Date | Event |
|---|---|
| 19 September 1968 | Commenced testing at McDonnell Douglas |
| 23 September 1968 | Antenna boom cracked |
| 24 September 1968 | Antenna boom repaired |
| 25 September 1968 | Ballast bonding failure<br>Testing suspended for redesign and repair of forward substrate |
| 11 October 1968 | Decision to transfer testing to Hughes |
| 15 October 1968 | Received forward panel from McDonnell Douglas |
| 17 October 1968 | Received aft panel from McDonnell Douglas |
| 23 October 1968 | Commenced vibration testing on T-6 |
| 28 October 1968 | Omni boom cracked during Y-axis sine |
| 29 October 1968 | Omni boom repaired |
| 30 October 1968 | Aft panel mounting points failed during Z-axis sine |
| 4 November 1968 | Aft panel shipped back to McDonnell Douglas for repair |
| 7 November 1968 | Concluded testing of forward panel |
| 8 November 1968 | Returned forward panel to McDonnell Douglas |
| 20 November 1968 | Aft panel returned from McDonnell Douglas |
| 27 November 1968 | Completed testing of aft panel |

## CONTROL ELECTRONICS

### Squib Driver

Squib driver unit 475306-101 is currently undergoing a retrofit of a capacitor in the time delay circuitry. The purpose of the change is to reduce the risk of failure to mission critical events with the use of a more reliable type of component. The change is from metalized paper (988501-161) to metalized polycarbonate (988509-95).

Because a limited quantity of space-qualified capacitors is available, only three of the four squib driver units are being retrofitted. Twelve circuits are involved. The specific circuits being changed were carefully chosen on the basis of tolerance of time delay, matching of redundant circuits, and priority of application confidence.

The repair involves breaking into an encapsulated assembly, and the technique was developed with the use of an engineering prototype model. To date, the flight units have been disassembled, the capacitors replaced, and the units functionally tested. During December, the units will be reassembled in new containers, encapsulated, and environmentally tested.

### Magnetic Damper

The magnetic damper unit X3161990 is currently in the assembly phase. All work is in accordance with the ATS program quality requirement (PQR) and waiver letter W-79. This includes the preparation of a released assembly drawing and 13 department-controlled drawings. Test procedures for the unit and terminal-type board subassemblies are also available. Approximately 90 percent of the purchased components are tested and aged by Hughes Components and Materials. Four boards using discretes are assembled up to the point of part shortages, and three boards using ICs are in subassembly test. The latter are using Fairchild uA9040 and uA9041 low-power DTL supplied by the Customer to Hughes; only one of each type (3 percent) failed functional tests.

### Payload Power Switch

The payload power switch 475320-100 S/N F-1 has been requalified for space flight during October.

### Ground Equipment

The STATS synchronous controller 475561-100 is to be updated to include the command address change similar to the ECR 332971 for the GCE synchronous controller 475559-100. The engineering information is currently being checked.

## COMMUNICATION SUBSYSTEM

### Transmitters

A laboratory model, 12-watt TWT, and TWT power supply, which uses solid encapsulation rather than foam in the high-voltage circuitry, has been in a life test since January 1968.

The units have been in a vacuum of $4 \times 10^{-5}$ millimeters of mercury 24 hours per day; the units are turned on four times each day and performance data recorded. After each turnon, the following data are recorded.

1) Unregulated input line voltage
2) Regulated output voltage
3) Input line current
4) RF power output
5) RF power input
6) Temperature (°F) of TWT collector
7) Temperature at power supply pass transistor stud (series regulator transistor).
8) Temperature at TWT heat sink (collector base)
9) Temperature at power supply heat sink (in vicinity of power transistors)

The performance has not degraded, and no irregularities have been noted. A series of tests has been run with the TWT and TWT power supply heat sinks at 120°F for 8 hours, and with the TWT power supply heat sink and TWT collector at 120°F for 8 hours, on alternate days for a 2-week period. This continuous test schedule has not been followed throughout the whole life test program because of the extensive manpower coverage requirement. Most of the life test program has consisted of frequent, but relatively brief turnons and performance documentation. Throughout the whole life test period, however, the power supply and TWT have remained in a vacuum continuously.

### Repeaters

The new scroll and SPLs for the L-band repeater have been released, and the preliminary specifications have been issued and discussed with NASA.

Design has begun on the following items:

| | | |
|---|---|---|
| 1) | Bandpass filters | 475118 - 133 and 135<br>475706 - 100 |
| 2) | Circulators | 475198 - 100 and 101 |
| 3) | High-level mixers | 475199 - 100 and 101 |

| | | |
|---|---|---|
| 4) | X2 multiplier | 475701 - 100 |
| 5) | Isolators | 475702 - 100<br>475705 - 100 |
| 6) | Driver amplifier | 475707 - 100 |
| 7) | Amplifier, narrow band | 475708 - 100 |
| 8) | Cross-strap module | 475709 - 100 |
| 9) | Tunnel-diode amplifier | 475177 - 131 |
| 10) | TWT electronic power conditioner | 475711 - 100 |
| 11) | TWT Faraday switch | 475185 - 130 |
| 12) | Power monitor | 475712 - 100 |
| 13) | Diplexer, transmitter/receiver | 475197 - 100 |
| 14) | RF switch | 475173 - 130 |
| 15) | Power amplifier | 475703 - 100 |
| 16) | X6 multiplier | 475704 - 100 |

Breadboard units are scheduled for the last week of January 1969; flight units are requested for the last week of February 1969. The following units have been assembled for breadboard use and are in the process of being aligned and tested to design specifications.

| | | |
|---|---|---|
| 1) | X2 multiplier | 475701 - 100 |
| 2) | Power amplifier | 475703 - 100 |
| 3) | Amplifier narrow band | 475708 - 100 |

The following units are existing in the ATS repeater, but will require defoaming and various amounts of modification.

| | | | |
|---|---|---|---|
| 1) | Master oscillator | 475122 - 139 | Crystal change |
| 2) | Limiter amplifier | 475137 - 130 | Transformer change |
| 3) | Filter amplifier | 475144 - 130 | Transformer and inductor change |
| 4) | Three mode regulator | 475710 - 100 | Diodes and resistor changes |

## Third Harmonic Generator

The third harmonic generator 475203 - 100 SN/F-4, power divider 475206 - 100 SN/F-2, and the payload regulator 475308 - 126 SN Y-4 have completed postfoam tests. The interim payload regulators 475308 - 103 SN/Y-1, - 105 SN/Y-1, - 105 SN/Y-2, and - 101 SN/Y-1 will have completed their postfoam tests by 20 December 1968. The flight units 475308 - 123, -124, -126, and -110 will complete fabrication by 3 January 1969

Transponder Simulator 475550 - 106 SN/G4

The repeater will be assembled (the triple-mode dc regulators 475129 - 100 were refabricated, and the master oscillator power amplifier 475142 - 100 was reworked) 16 December 1968 and will be predelivery-tested 10 December 1968. A diode will be added to the -28 volt dc input to the repeater and control circuitry to eliminate failure due to wrong dc polarity.

## BIBLIOGRAPHY

L. A. Hennis, "ATS F-1/F-3 Flight Data Comparison and Analysis," Hughes Aircraft Company, 12 September 1968.

G. A. Kane, "Preliminary Thermal Input to ATS-E Flight Acceptance Test," Hughes Aircraft Company, 19 September 1968.

L. A. Hennis, "ATS-E Flight Thermistors - Update," Hughes Aircraft Company, 23 October 1968.

# 3. SPACECRAFT DEVELOPMENT, FABRICATION, AND TEST

## SPACECRAFT DEVELOPMENT

### Stress and Dynamics

#### Magnetometer Boom

Both the T-6 boom and the ATS-E boom failed during qualification vibration testing of the heat pipe-solar panels. A review of X-rays indicates that in both cases removal of excess material from the interior of the main mast tube during rework to the magnetometer configuration contributed to the failures.

Information obtained during vibration testing of the heat pipe-solar panel on the T-6 spacecraft has resulted in revision of the design loads for the magnetometer boom. Analysis is currently under way to determine if the design is adequate for the revised loads and also if it is adequate for the increased loads due to the heavier, longer omni antenna.

#### ATS-E Centaur Adapter

Strength margins for the adapter for a 1900-pound spacecraft were recalculated and published.

Inspection of the ATS-E Centaur adapter revealed that it does not meet flatness or concentricity requirements at station -9.00 and is unacceptable for separation conditions. An attempt will be made to bring the adapter to within satisfactory tolerances by redriving rivets near station -9.00, while the adapter is held in the vendor's tool. If this is unsuccessful, another attempt will be made by replacing selected rivets while the adapter is held in the tool.

## Mass Properties

### ATS-E

Major weight increases in the ATS-E spacecraft that occurred during this report period were a result of the following changes:

1) Addition of a millimeter transmitter
2) Addition of third harmonic generator
3) Increased weight of the millimeter-wave package from 34 to 36 pounds
4) Addition of a switching converter
5) Addition of the L-band repeater experiment
6) Addition of a solar cell experiment

The spacecraft weight status as of 5 December 1968 is shown in Table 3-1.

The spacecraft system/subsystem weights and payload weights are listed in Table 3-2.

The mass property data in Table 3-3 indicates roll-to-pitch inertia ratios of:

1) 0.68 after spinup, prior to apogee motor ignition
2) 0.94 prior to apogee motor ejection
3) 1.15 after apogee motor ejection

TABLE 3-1. SPACECRAFT WEIGHT STATUS AS OF 5 DECEMBER 1968

| System | Subsystem | Maximum Weight, pounds | Current Weight, pounds | Difference (maximum versus current), pounds |
|---|---|---|---|---|
| ATS-E | Spacecraft | 1880.00 | 1879.46 | -0.54 |
| (S/G-2) | Interstage | 85.00 | 82.59 | -2.41 |
| F-5 | Centaur payload | 1965.00 | 1962.05 | -2.95 |

## TABLE 3-2. CURRENT WEIGHT STATEMENT

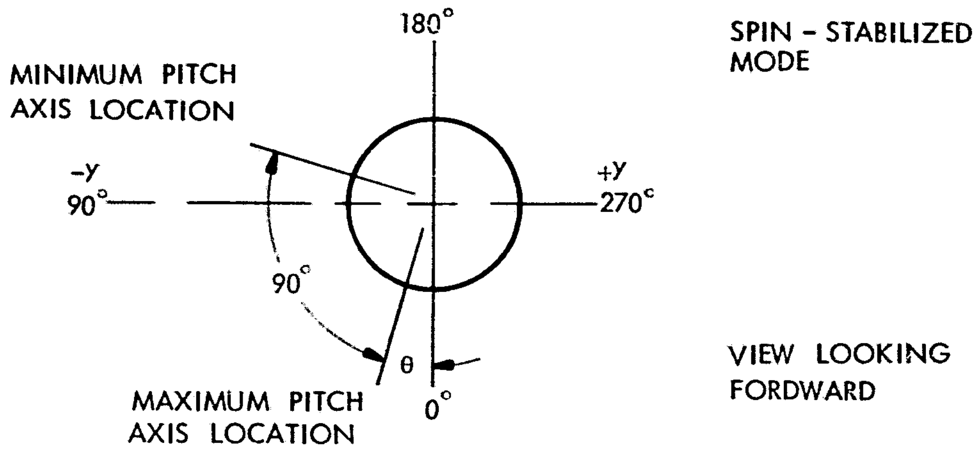
| Subsystem | ATS-E, pounds |
|---|---|
| Electronic System | (140.86) |
| Communication subsystem | 41.34 |
| Telemetry and command subsystem | 41.66 |
| Ancillary equipment | 57.86 |
| Wire Harness System | (80.56) |
| Main harness | 50.72 |
| Payload harnesses | 19.89 |
| Support installation | 4.20 |
| Coaxial cable installation | 5.05 |
| Resistojet harness | 0.70 |
| Power Supply System | (135.09) |
| Solar panel subsystem | 102.89 |
| Battery subsystem | 32.20 |
| Controls System | (99.45) |
| Free body spinup subsystem | 17.97 |
| $N_2H_4$ system AA | 22.09 |
| 180-degree inversion | 6.63 |
| East-west stationkeeping subsystem | 13.16 |
| Despin mechanism | 7.88 |
| Magnetic damping subsystem | 31.72 |
| Structure System | (139.57) |
| Basic subassembly | 70.27 |
| Secondary subassemblies | 52.00 |
| Miscellaneous structure | 1.76 |
| Payload supports | 15.54 |
| Miscellaneous Systems | (57.36) |
| Thermal control and paint installations | 38.28 |
| Sun sensor and nutation damper installation | 2.42 |
| Static and dynamic balance adjustment | 13.28 |
| Unaccountable weight | 0 |
| Solar pressure annulus installation | 2.53 |
| Air buoyancy | 0.85 |
| Payloads | (278.94) |
| Gravity-Gradient Experiment | (151.66) |
| Combination passive damper | 34.36 |
| X-boom mechanism (two) | 86.24 |

Table 3-2 (continued)

| Subsystem | ATS-E, pounds |
|---|---|
| TV camera | 3.77 |
| TV electronics | 5.03 |
| TV cable | 0.52 |
| IR earth sensor | 7.83 |
| Solar aspect sensor electronics | 2.70 |
| Power control unit | 8.91 |
| Solar aspect sensors (five) | 2.30 |
| Millimeter-Wave Experiment | (36.00) |
| Environmental Measurement Experiment | (48.31) |
| Primary package | 40.00 |
| Remote package | 7.00 |
| Antenna couple unit | 0.40 |
| CMVM | 0.91 |
| Magnetometer Experiment | (7.57) |
| Sensor | 0.74 |
| Electronics | 6.83 |
| Ion Propulsion Experiment | (20.40) |
| Thruster assembly (two) | 16.00 |
| Common mode voltage monitor | 0 |
| Cable assembly | 0 |
| Filter assembly (two) | 4.40 |
| Solar Panel Experiment | (7.65) |
| Large panel | 1.39 |
| Small panel | 0.26 |
| Electronics | 6.00 |
| Magnetic Compensation | (0.35) |
| Millimeter Transmitter Experiment | (7.00) |
| Final Orbital Condition | (931.83) |
| Propellant - 180-degree inversion | 2.04 |
| Propellant - east-west stationkeeping | 3.00 |
| Total at station after despin | (936.87) |

# TABLE 3-3. ATS-E MASS PROPERTY DATA

| Condition | Current Weight, pounds | CG* Station, inches | $I_{xx}$, slug-feet$^2$ | $I_{yy}$, slug-feet$^2$ | $I_{pitch}$, slug-feet$^2$ | $\frac{I_{roll}}{I_{pitch}} = R/P$ | Maximum Pitch Axis Location, θ, degrees |
|---|---|---|---|---|---|---|---|
| Centaur payload at liftoff | 1962.05 | 2.88 | | | | | |
| Interstage | 82.59 | | | | | | |
| Spacecraft at separation | 1879.46 | 4.22 | 178.68 | 176.70 | 178.81 maximum | $\frac{120.10}{176.57_{min}} = 0.68$ | At 14.25 |
| Propellant - $N_2$ spinup | 8.44 | | | | | | |
| Total after spinup; prior to apogee motor ignition | 1871.02 | 4.22 | 178.93 | 176.81 | 178.98 maximum | $\frac{119.74}{176.75_{min}} = 0.68$ | At 9.41 |
| Propellant - apogee motor | 760.00 | | | | | | |
| Expendables - igniter | 1.00 | | | | | | |
| Expendables - apogee motor | 5.00 | | | | | | |
| Total after apogee motor burnout | 1105.02 | 13.47 | 112.55 | 110.44 | 112.61 maximum | $\frac{103.35}{110.38_{min}} = 0.94$ | At 9.41 |
| Outgassing - apogee motor | 3.00 | | | | | | |
| Total prior to apogee motor ejection | 1102.02 | 13.56 | 111.67 | 109.56 | 111.73 maximum | $\frac{103.31}{109.50_{min}} = 0.94$ | At 9.41 |
| Apogee motor installation | 73.81 | | | | | | |
| Motor adapter | 43.24 | | | | | | |
| Total after apogee motor ejection | 984.97 | 16.63 | 85.84 | 83.66 | 83.61 minimum | $\frac{99.10}{85.89_{max}} = 1.15$ | At 8.32 |
| Propellant - $N_2H_4$ AA subsystem | 35.85 | | | | | | |
| Pressurant - $N_2$ AA subsystem | 0.29 | | | | | | |
| Total at station; prior to despin | 948.83 | 16.40 | 83.69 | 81.36 | 80.55 minimum | $\frac{95.23}{84.50_{max}} = 1.13$ | At 153.00 |
| Yo-yo despin assemblies | 11.96 | | | | | | |
| Total at station; after despin | 936.87 | 16.50 | 81.56 | 81.04 | 79.45 minimum | $\frac{93.09}{83.15_{max}} = 1.12$ | At 139.03 |



## STRUCTURE, GENERAL ARRANGEMENT, AND INTEGRATION

The major effort for this report period was the engineering of the package of 20 ECR changes to the ATS-E vehicle. The change package consists of the following major items:

- Additional millimeter-wave transmitter
- Magnetic control system
- High-gain omni antenna
- Common mode voltage monitor
- Switching converter
- Third harmonic generator
- Solar cell experiment
- L-band repeater

In total, 46 new units, 18 thermistors, and approximately 1800 terminations will be added to the spacecraft by the changes. Eighty-seven of the 141 new drawings and changes are released to date.

Preliminary systems spacecraft testing was completed during this report period, using the experiments and equipment currently available. The ATS-E spacecraft has had a number of units, the aft bulkhead, and the thrust tube back removed to provide access for rework. Retrofit is progressing as changes and new units are available.

The latest ICD on the millimeter-wave transmitter added two protrusions to the mounting surface of the unit that interfere with the existing structural attachment. An additional change will be required to add a spacer block to the spacecraft mounting surface and modify the horn attachment bracket to accommodate the revised unit.

## ATS-E SPACECRAFT FABRICATION

The spacecraft was completed during September for start of systems testing. At that time, a plan was established for the incorporation of several new experiments on the spacecraft, the largest impact being the addition of the magnetometer. According to plan, starting 1 November 1968, the spacecraft would be returned to Manufacturing for making the mechanical (bulkheads, removable structure, stringer, etc.), electrical (harnesses) and thermal changes. A two-shift, 60-hour week was planned with completion for continuation of systems testing just prior to 25 December.

In October, Hughes was notified that two additional experiments, the solar cell degradation and L-band repeater, were to be incorporated, resulting in a significant spacecraft delivery slippage. All overtime and second shift effort was removed from the plan. At the close of the report period, a new plan for incorporating the L-band repeater had not been finalized. It is anticipated, however, that all experiments can be incorporated in sufficient time for completing systems testing and shipment to NASA by 18 July 1969.

## ATS-E SPACECRAFT TEST

During this reporting period, the initial spacecraft assembly was completed except for the following items. These items are not on or connected to the spacecraft.

1) Millimeter-wave
2) Squib drivers
3) Gravity-gradient TV
4) Resistojet
5) Solar panels
6) Omni antenna
7) Planar arrays
8) Sun sensors

The tests shown in Table 3-4 were completed prior to returning the spacecraft to manufacturing for the rework required by the additions listed in Table 3-5.

TABLE 3-4. TESTS COMPLETED PRIOR TO REWORK OF ATS-E

| Number | Name |
|---|---|
| Miscellaneous Test | |
| 11 | TM modulation index |
| Spacecraft Long-Form System Performance Tests | |
| 1 | Command receiver sensitivity |
| 2 | Command receiver bandwidth |
| 3 | Command system sensitivity and receiver/decoder cross-strapping |
| 4 | Decoder operation, count mode |
| 5 | Telemetry transmitter frequency |
| 6 | Telemetry transmitter power output |
| 7 | Encoder analog-to-digital conversion accuracy |
| 8 | Pulse rate measurement |

Table 3-4 (continued)

| Number | Name |
|---|---|
| 9 | Real-time data |
| 10 | Telemetry cross-strap check |
| 11 | Telemetry channel check |
| 12 | Telemetry dwell |
| 13 | Timer oscillator frequency and stability |
| 14 | Spacecraft clock accumulator operation |
| 15 | Master oscillator frequency stability |
| 16 | VCO and master oscillator frequency |
| 28 | Subliming solid drivers |
| 30 | Solenoid driver and jet operation |
| 31 | Subsystem power consumption |
| 32 | Battery discharge control |
| 33 | Voltage limiter check |
| Telemetry Calibration | |
| 1 | Unregulated bus voltage sensors |
| 2 | Solar bus current sensors |
| 3 | Battery bus voltage sensors |
| 4 | Battery discharge current sensors |
| 5 | Battery charge current sensors |
| 7 | Telemetry transmitter output power sensors |
| 13 | Communication transmitter output power sensors |

TABLE 3-5. NEW EXPERIMENTS AND HUGHES UNITS
TO BE ADDED TO ATS-E

| Number | Description |
|---|---|
| 1 | Magnetometer |
| 2 | Magnetic damper |
| 3 | Solar cell degradation |
| 4 | Third harmonic generator |
| 5 | Voltage monitor |
| 6 | L-band repeater |
| 7 | Millimeter-wave beacon transmitter |
| 8 | Switching converter |

## BIBLIOGRAPHY

M. H. Magie, "Strain Gauges on Heat Pipe Solar Panels, ATS," Hughes Aircraft Company, 6 September 1968.

R. A. Pizarro, "H. P. /S. P. Procurement Specification," 12 September 1968.

R. A. Pizarro, "Upgrade of ATS T-6 S/C to ATS-E (F5) Configuration for Qualification Vibration Test," 15 October 1968.

R. A. Pizarro, "Upgrade of T-6 Test Vehicle for Use in ATS Heat Pipe Solar Panel Qual. Tests," 18 October 1968.

M. H. Magie, "Centaur Adapter Margin or Safety; (ATS)," 21 October 1968.

R. V. Swanson, "ATS-E (F-5) Station 0. 0 Separation Clearance Study," 23 October 1968.

C. D. Knauer, "ATS-E Loads for Transmitter and Transmitter Horn Installation," 4 November 1968.

R. A. Pizarro, "ATS-E Aft Heat Pipe/Solar Panel Insert Failure Analysis," 21 November 1968.

T. A. Savo from W. R. Clay, "ATS Monthly Progress Report, 1 through 30 September," 2 October 1968.

T. A. Savo from W. R. Clay, "Revised Schedule for ECR Package on ATS-E," 7 October 1968.

BIBLIOGRAPHY (Continued)

Distribution from W. R. Clay, "Design Evaluation ECR Package ATS-E," 9 October 1968.

T. A. Savo from W. R. Clay, "Incorporation of Solar Cell Experiment at Incremental Time Periods," 9 October 1968.

D. A. Reed from W. R. Clay, "Hardware ABM for ECR Change Package ATS-E," 11 October 1968.

Distribution from W. R. Clay, "Design Evaluation ECR Package ATS-E," 21 October 1968.

T. A. Savo from W. R. Clay, "ATS Monthly Progress Report, 1 through 31 October," 4 November 1968.

J. P. O'Donnell from E. S. Rice, "Loosened Screws Attaching ATS-E Forward Substrate to Bulkhead Fittings," 8 November 1968.

J. V. Mullin from Clay, Pizarro/O'Donnell, "Install Reworked Aft Heat Pipe Substrate on T-6," 19 November 1968.

Reed/Mullin from W. R. Clay, "Rework Plan for ATS-E Centaur Adapter Assembly," 25 November 1968.

# 4. GOVERNMENT-FURNISHED EQUIPMENT INTERFACES

## ATS-E ACTIVITY

### Gravity-Gradient Experiment

The gravity-gradient experiment flight hardware, less booms, was received on 20 September. The General Electric booms were received on 21 October. All hardware was EPC-tested and installed on the spacecraft. The TV camera cable G298053 - was made to the ATS-A configuration and was too short to reach the TV electronics unit. A new cable is to be supplied by General Electric.

The Westinghouse outline and mounting drawing was carefully reviewed by Hughes. The Westinghouse boom appears to be a direct replacement for the General Electric boom. The major difference between the booms is a weight increase of 4 pounds in each of the Westinghouse X-booms. This weight increase is critical.

At present, a physical interference problem exists between the General Electric X-boom squib covers (removal) and the resistojet thruster brackets. This interference does not exist with the Westinghouse booms. If it becomes necessary to fly the General Electric booms, the resistojet thruster brackets will be trimmed.

### Environmental Measurements Experiment

The EME prototype hardware completed EPC testing in September and was installed on the spacecraft. The following problems arose during spacecraft installation:

1) Spacecraft wiring did not follow prescribed routing to J-2 and J-3. Westinghouse agreed to cut a hole in their thermal housing to permit one cable to enter on each side of the unit to J-2 and J-3.

2) There was insufficient clearance between J-2 and J-3 and the unit thermal insulation to permit a normal connector boot

assembly. Hughes will remove the connector boot and pot the connectors.

3) Hughes thermal shield clearance holes on one side of the unit did not line up with the prescribed mounting bolts. Hughes will redrill clearance holes to the thermal shield mounting bolts.

The addition of a voltage monitor to the spacecraft necessitated the removal of the gravity-gradient coupling adapter at $\theta = 90$ degrees and a rewiring of the adapter at $\theta = 270$ degrees. Rewiring of the coupling adapter has been completed.

Millimeter-Wave Experiment

During this report period, NASA made the decision to add a millimeter-wave backup transmitter onto the spacecraft. The addition of this unit generated several problems which have now been resolved:

1) The unit required 5 ma at +6.8 volts which will be supplied from the Hughes encoder regulator.

2) The unit must dissipate 22 watts - mostly by conduction. This requirement will be satisfied by mounting the unit on a thermal doubler of approximately 170 square inches of 0.071-inch aluminum.

3) The unit mounting surface has two 0.300-inch protrusions which interfere with the mounting bracket. Hughes will insert a 0.300-inch plate match cut to the transmitter mounting surface between the transmitter and the mounting bracket.

An experiment configuration consisting of the transmitter, a 2-1/2 inch waveguide with a 90-degree twist, a directional coupler, an 8-degree waveguide bend, and an antenna has been agreed to. ICD G298083 has been assigned to this experiment and requires revision and signoff.

The experimenter has encountered space and power problems with the millimeter-wave primary unit. Hughes has submitted an increased space envelope to the experimenter and is awaiting a revised ICD G298088. Hughes is costing a change to the experiment payload regulator to increase the current capacity from 1.5 to 1.7 amperes.

Ion Engine Experiment

Both flight ion engines with their filter units were received on 22 September, successfully completed EPC testing, and were trial-fitted on the spacecraft. The units are presently in GFE stores with a local NASA lien against the neutralizer filaments.

During this report period, the decision was made by NASA to add a voltage monitor to the spacecraft similar to the one flown on ATS-D. Electrical interface and physical characteristics have been agreed to.

Spacecraft redesign to add the unit has been completed and hardware rework is in progress.

## Resistojet Experiment

Nonflight prototype hardware was received in September, successfully completed EPC testing, and was installed on the spacecraft. Minor problems were encountered and resolved when fitting the thruster brackets to the spacecraft structure.

## Magnetometer Experiment

During this report period, the decision was made by NASA to add a magnetometer to the spacecraft. The physical characteristics will be identical to the experiment flown on ATS-D with the exception of wiring. Additional wiring between the sensor and electronics unit and between the electronics unit and the EME will be required. The installation arrangement will be identical to that of ATS-D. Engineering to install the new units is complete, and spacecraft rework is progressing.

## Solar Cell Radiation Damage Experiment

During this report period, a decision was made by NASA to add a solar cell radiation damage experiment to the spacecraft. An interface meeting was held at Hughes on 7 November with NASA, JPL, and Hughes in attendance to clarify interface data. Unit outline and mounting drawings, interface specification, and test plan requirements were reviewed and action items assigned.

## Open Interface Items

1) Flight hardware for environmental measurements, gravity-gradient, millimeter-wave, resistojet, magnetometer, voltage monitor, and solar cell experiments.
2) Interface control drawings (new or revised) for gravity-gradient, millimeter-wave, voltage monitor, and solar cell experiments.
3) Test plans (new or revised) for gravity-gradient, millimeter-wave, magnetometer, voltage monitor, and solar cell experiments.
4) Interface specifications (new or revised) for gravity-gradient, millimeter-wave, voltage monitor, and solar cell experiments.

Table 4-1 gives the status of the ATS-E experiments as of 30 November 1968.

## TABLE 4-1. ATS-E EXPERIMENT STATUS

| Experiment | ICD | Interface Specification | Test Plan | Drill and Alignment Fixture | Installation Fixture | Flight Hardware |
|---|---|---|---|---|---|---|
| Gravity gradient | Approved | Received | Received | Received | Received | Received |
| Westinghouse booms | Signoff required | Not applicable | Required | Not applicable | Not applicable | Due 2-7-69 |
| Ion engine | Approved | Received | Received | Received | Not required | Received |
| Ion engine filters | Approved | Received | Not required | Not required | Not required | Received |
| Voltage monitor | Signoff required | Received | Revision required | Not required | Not required | Due 2-7-69 |
| Resistojet | Approved | Received | Received | Not required | Not required | Prototype rcvd<br>Flight due 2-14-69 |
| Resistojet clock box | Approved | Received | Received | Not required | Not required | Prototype rcvd<br>Flight due 2-14-69 |
| EME primary unit | Approved | Received | Received | Not required | Hughes | Prototype rcvd<br>Flight due 3-1-69 |
| EME remote unit | Approved | Received | Received | Not required | Not required | Prototype rcvd<br>Flight due 3-1-69 |
| Gravity-gradient coupling adapter | Approved | Received | Received | Not required | Not required | Due 2-7-69 |
| Millimeter wave | Revision required | Revision required | Received | Not required | Hughes | Due 2-21-69 |
| MMW backup transmitter | Revision required | Revision required | Revision required | Not applicable | Not applicable | Due 2-21-69 |
| Magnetometer | Signoff required | Received | Revision required | Not required | Not required | Due 2-7-69 |
| Magnetometer electronics | Signoff required | Received | Revision required | Not required | Not required | Due 2-7-69 |
| Solar cell radiation damage | Signoff required | In work | In work | Not applicable | Not applicable | Due 2-21-69 |

## APOGEE MOTORS

The apogee motors have been designated for the ATS-E spacecraft. They are motor Z-6 for flight and Z-4 as the flight spare. No support activities were required during this quarter.

## CONFIGURATION MANAGEMENT

### Interface Change Request (ICR) Activity

The following ICRs for the ATS-E spacecraft were received at Hughes during the reporting period:

| ICR | Subject |
|---|---|
| 00984 | GG — revise wiring for sun shutter operation. |
| 00985 | Heat pipe — add 19 thermistors and telemetry. |
| 00986 | EME — black conductive paint - forward solar panel, bulkhead, and thermal barrier. |
| 00987 | GG — add telemetry to indicate release of primary boom tip masses. |
| 00988 | GG — add primary boom alignment mirror. |
| 00989 | CMVM — add common mode voltage monitor to spacecraft. |
| 00990 | EME — particle sensor field-of-view interference. |
| 00991 | IRS — relocate IR sensor. |
| 00992 | Ion — reassign telemetry from ion engine to magnetic control system. |
| 00993 | Mag — relocate compensation magnets. |
| 00994 | S/U — provide firing delay for spinup system. |

# 5. TECHNICAL SUPPORT ITEMS

## COMPONENTS AND MATERIALS

Process specification HP 4-112 for applying thermal control finishes was revised to greatly broaden the application techniques and improve the quality assurance provisions. The specification is in the process of being released.

Particulate counting and hardware cleaning has been provided on an as-required basis.

A Hughes process specification and a Hughes material specification are being prepared for a black antistatic thermal control coating that was evaluated and recommended by the Materials Technology Department.

Adhesive bonding procedures were established for bonding ballast weights to the heat-pipe substrate. A failure analysis was conducted on a damaged substrate and determined not to be an adhesive failure between the weights and the substrate.

HS-6018 was revised to include a second type nickel grid for MICAM.

A detailed process specification was prepared for assembly of the MICAM stick in support of the electronic hardware.

## QUALITY CONTROL

### Quality Engineering

Detailed inspection plans (DIP) 475262-801 and 475263-801 covering the assembly of the aft and forward solar panels have been written, approved, and distributed.

Planning screeners have reviewed 193 sets of planning containing 3810 operations during this report period; 129 defects were observed and the plans returned for correction.

## Drawing and Change Control

The change control point has processed 76 changes to the inspection area during this report period and received 107 verified changes in return.

## Test and Test Verification

System test was completed on the $N_2$ spinup system (475072-101). Telemetry calibration tests were performed on all sensors.

Major modification continues on the ATS-E spacecraft with major modification to the wire harness about complete.

The McDonnell Douglas aft solar panel substrate ID 11401-1 has been vibrated on the T6 spacecraft in all three axes random and sine.

A new solar cell degradation experiment is to be installed in the thermal band area of the ATS-E spacecraft.

Experiment package control tests were performed on the following units:

| | |
|---|---|
| Ion engines | G298033 S/N-07 and S/N-08 |
| Gravity-gradient experiment | |
| Boom assembly A | G298041 S/N-5962335 |
| Boom assembly B | G298041 S/N-5962336 |
| Power control unit | G298045 S/N-5962034 |
| Libration damper | G298042 S/N-5962031 |
| IR earth sensor | G298044 |
| TV camera | G298043-1 |
| TV electronics | G298046-1 |
| Five solar aspect sensors | G298047 |
| Solar aspect sensor electronics | G298048 |

Long-form system performance tests 1 through 16 were conducted on the ATS-E spacecraft. The only failure was against the command receiver 2 (475210-100) F-17, which indicated a narrow bandwidth of 40 kHz. Trouble and failure report 12255 was written covering the discrepancy.

The resistojet clock G298712 was rotated 180 degrees on the assembly per EO 33849 against unit installation drawing 3041994.