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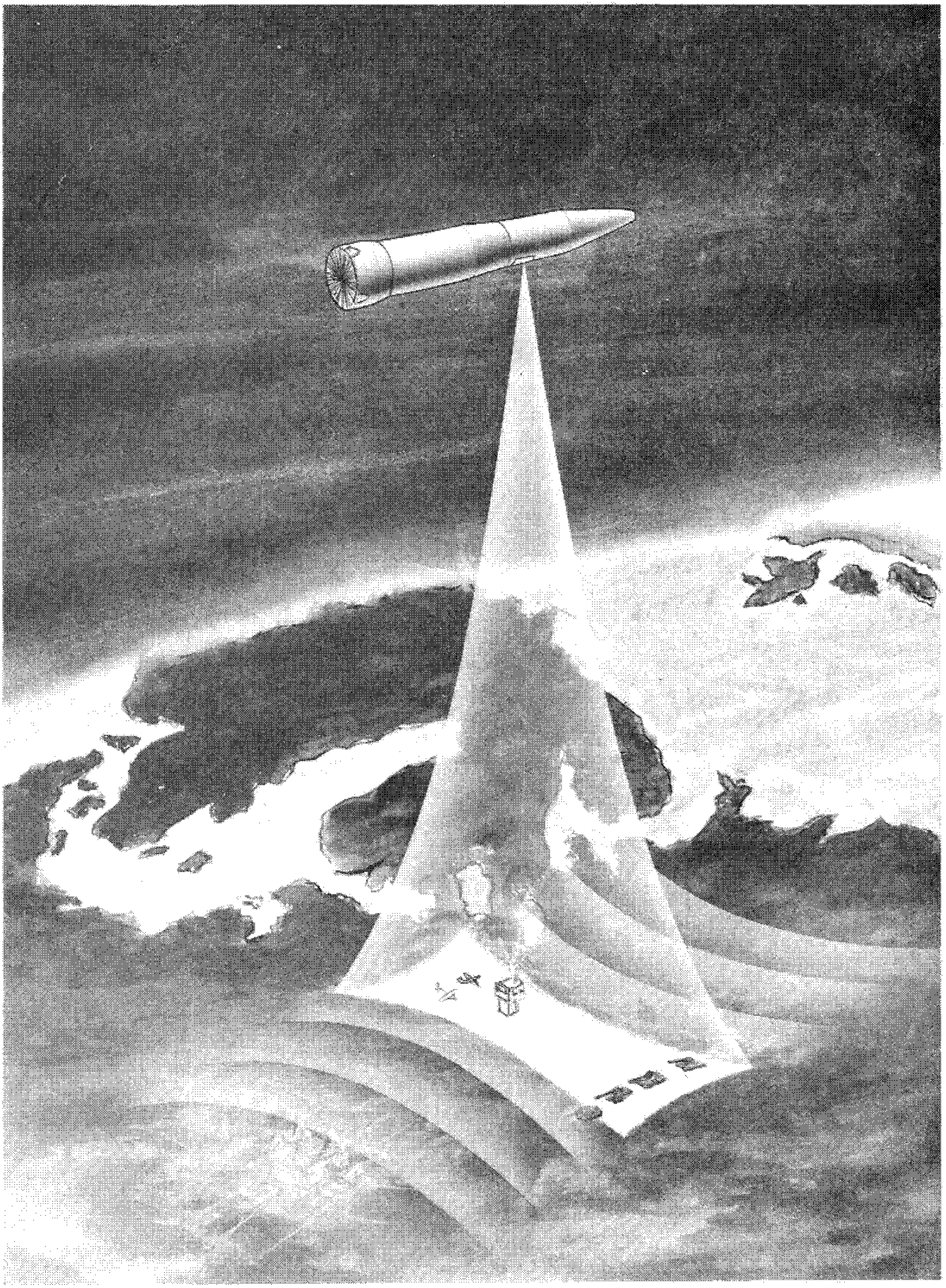

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FASTBACK

A FAST RESPONSE
PHOTOGRAPHIC RECONNAISSANCE
SYSTEM

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November 1970

MARTIN MARIETTA DENVER DIVISION

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FOREWORD

We believe that a requirement exists for an orbital photographic reconnaissance system which will economically provide high quality data to the user in less than 24 hours after the request is made. We also believe that such a system can be readily acquired and operated because of the availability of existing technology for recovery and reuse, the availability of economical launch vehicles, and the availability of well developed camera technology.

The following report summarizes the studies we have conducted to establish the characteristics of such a system, the engineering that establishes the feasibility, the cost studies to establish a budgetary estimate of the funding requirements, and the schedule studies to estimate program timing.

The result of these studies is a system which we call FASTBACK. We believe FASTBACK offers an attractive option for economically fulfilling the fast response intelligence requirement.

L. J. Adams
Laurence J. Adams
Vice President
Special Projects

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I. INTRODUCTION

FASTBACK is designed to give the U. S. intelligence community a flexible satellite reconnaissance capability that meets the following needs:

Rapid Response - It can be targeted against a target or an area of concern and return high quality imagery to the Washington area in less than 24 hours after the need for imagery is recognized.

On the basis of an analysis of the various factors involved, the system should have the imagery processed and in the hands of the intelligence community in a nominal average of eighteen hours.

By design of the launch complex, a daily launch rate can be maintained for at least four days.

Access - In one day it can provide overflight of a considerable portion of the globe. Depending upon launch azimuth selection, selected targets can be covered more than once in the same flight. Launch azimuth selection also allows the direction of overflight to fit the major dimension of the primary target complex or area.

Image Quality - The sensor that we have designed into FASTBACK will provide stereo imagery with 3.3 feet ground resolution at nadir, adequate to provide high assurance of getting a large volume of information on ground order-of-battle problems.

The capability of the camera system is, of course, a trade-off of the use of available weight and volume in the spacecraft between the optics, stereo system, and the film load.

Our trade studies show that by using a higher energy orbit injection motor an alternate sensor system which will provide a ground resolution of approximately 2.5 feet of monographic imagery can be used.

Target Coverage - For the stereo system the maximum film load will allow a total of 10,000 miles along track and the camera dynamics create a swath width of 130 nm. The launch azimuth will determine the number of orbits during which targets will be overflown. Of the four to nine revolutions per mission, targets can be overflown on three to seven revolutions. Within these constraints, there is extreme flexibility of camera operation.

If the camera is activated in the region of maximum declination, then the coverage will be primarily along an east-west axis and will allow double coverage of 10 to 15 target groups.

If the camera is activated in the regions of minimum declination, single pass coverage largely in a north-south direction is obtained.

Combinations of targets and areas located in regions of low and high declination are to be expected and a large degree of flexibility is inherent in the system's ability to launch essentially unrestricted in azimuth and with a wide variation in launch time.

Thus, the film usage, coverage shape, and area to be covered is under the control of the system tasker to a very large degree.

Weather Avoidance - The coverage pattern and timing of FASTBACK can be adjusted to take advantage of the best weather opportunity available over the target areas of interest. Since coverage can be concentrated in a reasonably compact area, maximum advantage can be taken of the good weather patterns associated with high pressure areas. On the other hand, if it is essential to obtain coverage in bad weather, the stereo coverage plus double access capability can provide four separate looks through the cloud pattern during a single day, thus increasing the probability of obtaining useful coverage in spite of the bad weather.

Low Cost - Based on a maximum use of already proven technology, recovery and re-use of the payload, and a low cost booster, every effort has been made to reduce cost as a factor affecting the decision for a given launch.

Low Vulnerability - Since each FASTBACK mission would have its unique orbit and there would normally be only four or five revolutions over Soviet territory, it would be much more difficult to attack this system than it would be to attack a satellite on polar orbit - especially if the polar system stayed on orbit longer than one day. Of course, some FASTBACK missions would not cross Soviet territory at all.

It is contemplated that FASTBACK'S flexible capabilities would make it an important source of intelligence information in problems like the following:

Soviet Bloc Military Posture During Political Tension - for example - a Berlin or Czechoslovakian crisis.

Soviet Involvement In Military Activity Outside Of The Bloc - e.g. Introduction of SAMs into North Vietnam, introduction of sophisticated weapons into Egypt, Soviet Naval activity in the Mediterranean.

Development Of Critical Military Technology And Doctrine - For example - FASTBACK could provide broad coverage of both Kapustin Yar and Sary Shagan on the same day to help assess a new ABM exercise. Coverage of a large portion of the ICBM belt could also be obtained in one day to help assess deployment trends. The Chinese Missile Test Range and nuclear test area could both be covered on the same day.

Tactical Situations Of Great Interest To The U.S. - At various times in the past we might well have made use of FASTBACK over Suez, Yemen, the Sino-Indian Border, the India-Pakistan area, the Taiwan Straits, Czechoslovakia, and Berlin.

Support Of U.S. Tactical Operations - FASTBACK could acquire high resolution coverage of large areas over enemy lines or at sea in a timely manner at no risk to U.S. lives and aircraft. If a U.S. field commander could depend on getting basic coverage of a 130 mile swath over enemy territory every good-weather day, he could employ his other reconnaissance assets with much greater efficiency.

Supplement Other U.S. Reconnaissance - On many occasions other U.S. reconnaissance assets cannot provide all of the coverage needed to satisfy an intelligence problem because of schedule and orbital constraints, weather, or competition with other intelligence problems of higher priority. FASTBACK would provide a low cost means of acquiring additional coverage over areas where insufficient imagery had been acquired by other means.

To Support U.S. Military Contingency Preparations - In many situations U.S. strategic forces must be postured to support U.S. policy. To achieve maximum effectiveness, these forces often need up-to-date information on the forces and target systems of a possible opponent, yet these forces and target systems might be far removed from the area on which U.S. intelligence attention was focused. For example, at the time of the Cuba Missile crisis, the posture of Soviet strategic forces was of great importance. In such a situation, FASTBACK could be used to get current information for U.S. contingency forces without interfering with other intelligence problems .

Other Needs - From time to time situations arise in which reconnaissance can contribute to the solution of important problems, even though these are not typical crisis situations. For example, coverage of the Alaska and Peru earthquakes might well have been obtained by a FASTBACK capability even though these did not involve any hostile action. Such uses would not necessarily preclude the system from acquiring useful imagery of intelligence targets on the same mission.

SYSTEM DESCRIPTION

Launch Site - A launch from the Pacific is desirable in order to obtain broad coverage of denied territory and to minimize system response time. Of the several islands considered (Jarvis, Hawaii, Howland, Baker, Kwajalein and Johnston), Johnston looks the most promising because facilities already located on Johnston are adequate to meet the demands of this program.

In the event that Johnston Island for some reason should not be available to this program, facilities at ETR and WTR could be used. A small degradation in multiple coverage areas will result.

Booster Selection - The Minuteman I was selected as the booster for FASTBACK primarily because of the large number (368)* of surplus ICBM vehicles that are available at minimal cost ($75 K per booster).* In addition to economic advantages, the selected booster is relatively simple in design, has a quick response time and has sufficient payload capability to meet the requirements of the program.

Spacecraft Description - An isometric cut-away of the spacecraft configuration is shown on Figure I-1. The spacecraft, including kick motor, is 16 feet long and weighs approximately 2600 pounds. Its base diameter is 37-1/2 inches to be compatible with the third stage of the Minuteman booster.

Subsystem selections were based on the use of existing space qualified hardware, whenever possible, to minimize development time and program costs. This led to the following subsystems:

*This information was obtained from the USAF Minuteman Booster Office and includes the cost of shipment to port of embarkation plus refurbishment.

The solid rocket motor that is used to inject the 900 pound on-orbit spacecraft into the 65 by 300 nm. orbit is the Thiokol TE-M-364-2 motor.

A hydrazine mono-propellant attitude control system (ACS) is used to maintain vehicle stabilization during the kick motor burn and for orbit adjust. The system is pressurized by gaseous nitrogen which is contained in the small sphere. In addition to the high thrust level hydrazine system, there is a low thrust level nitrogen system that also uses gases from the nitrogen pressurant system. The nitrogen system is used to provide 3-axis stabilization during orbit operations. This technology is well developed and components developed for other programs will be used in FASTBACK.

A small solid rocket motor, the Thiokol TE-236, is used for de-orbiting the spacecraft.

The parachute recovery system is packaged in a toroidal configuration around the de-orbit motor. This recovery system contains both the drogue-ballute and main parachute used for air recovery. The recovery system developed for the Martin Marietta Corporation PRIME vehicle will be adapted to this use.

A single guidance system is used for both boost and orbital operations. It consists of The Hamilton Standard DIGS inertial strapdown system and the Teledyne 8K, 24 bit word computer located in the forward portion of the spacecraft. The guidance system requires no command link or update.

The remaining equipment in the forward section of the spacecraft includes a 230 ampere-hour silver zinc battery, an RF beacon, ordnance firing circuits and subsystem support equipment.

Temperature control of the subsystems and payload is provided by a semi-passive thermal control unit using paints with selective thermal characteristics, insulation, and evaporative coolers. The external monocoque structure is covered with ablative materials to maintain the skin within temperature limits during boost and re-entry. The Martin Marietta Corporation has successfully used a very similar semi-passive thermal control unit and ablative materials on the PRIME lifting re-entry body program.

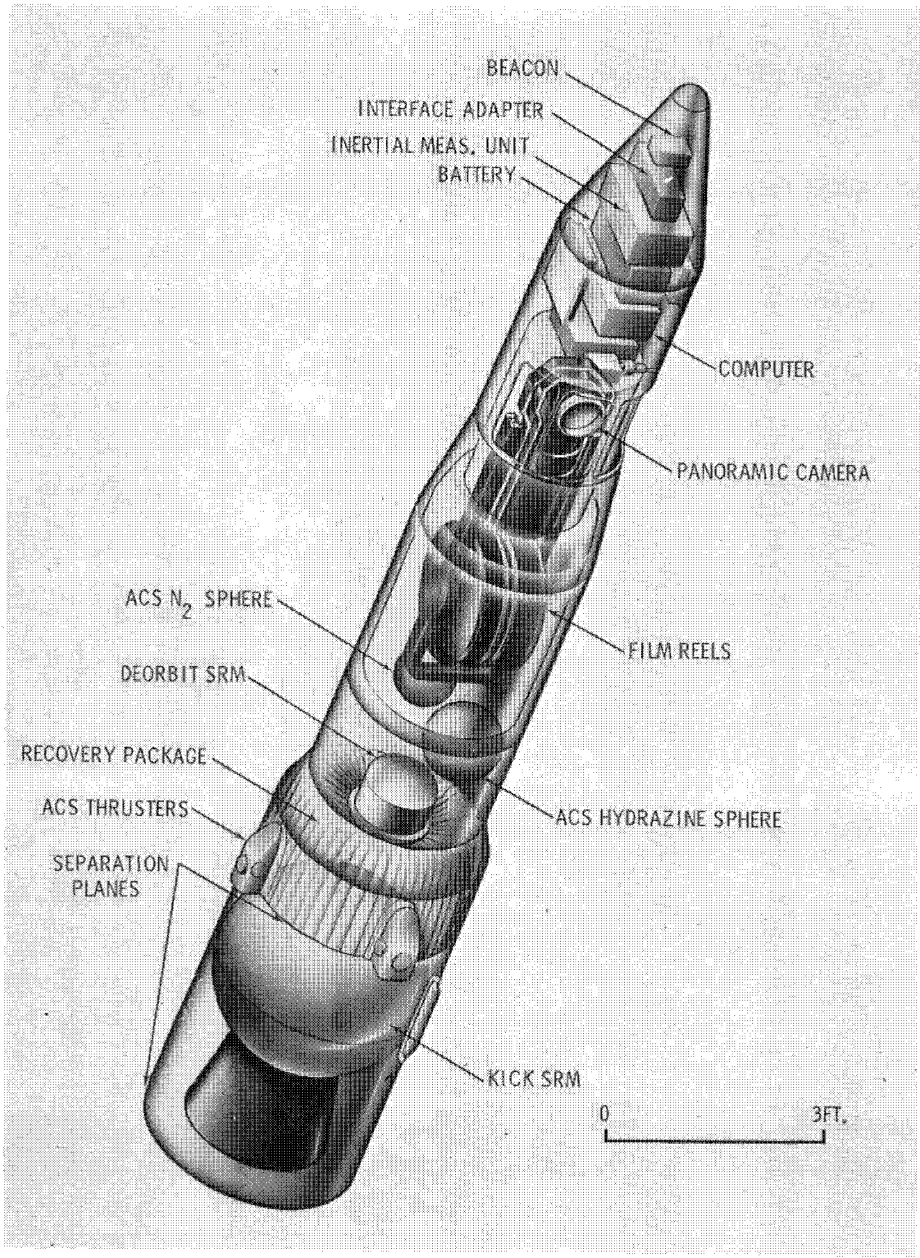

FIGURE I-1 SPACECRAFT CONFIGURATION

Payload (Optical System) - A panoramic optical bar camera is proposed. The front portion of the camera rotates about its longitudinal axis at a rate of 45 rpm and has a scan angle of ± 45 degrees. This scan angle equates to a ground swath of 130 nm. Stereo coverage is obtained by tilting the camera in pitch ± 10 degrees. The optical package has been sized to provide a ground resolution of 3.3 feet at nadir.
Three camera manufacturers have been contacted in regard to optical systems for this installation. Their results show that no new "state-of-the-art" assemblies or techniques are required for their designs. One of the manufacturers (ITEK) has built and flown an aircraft camera which is very similar to the unit proposed for this spacecraft installation. This camera will require slight modification to meet the space environment and a lens change to obtain increased ground resolution.

Recovery Considerations - The system has been designed for an air recovery off the east coast of the United States . Recovery is accomplished by using existing C-130 air recovery aircraft. It is anticipated that sufficient aircraft are available in the current program to support this assignment.

Reusability - The complete orbital spacecraft is recovered and the majority of the subsystems are refurbished and reused. Refurbishing and reusing the space vehicle reduces the recurring launch costs by about a factor of three.

Higher Launch Rates - Capability could be provided for significantly higher launch rates and longer sustained launch capability by adding additional spacecraft and support equipment to the program.

MISSION OPERATION SEQUENCE

Figure I-2 shows the overall flow of the major program components, indicating the geographical locations involved and the operations performed at each location.

The Minuteman boosters will be refurbished and modified at Hill AFB to interface with the spacecraft at a point immediately forward of Stage III (Guidance Section removed). They will be transported by rail and ship in their environmentally controlled transport vans to Johnston Island.

The spacecraft will be fabricated, assembled and tested at Martin Denver and mated with the payload. Following environmental and combined systems testing, the assembled spacecraft (loaded with film essentially ready for launch) will be airlifted in its environmentally controlled shipping container to Johnston Island.

The booster and spacecraft will be mated and checked out in a horizontal position on a transporter erector similar to that employed in the Thor Program. Activities at Johnston Island will be limited to spacecraft - booster combined systems tests, igniter installation, ordnance connection and propellant loading.

The assembled and checked out vehicle will be held in a horizontal protective enclosure on the launch pad, permitting rapid response to launch command. When erected for launch, a guidance system update will be supplied from ground equipment and the "go" status of the complete vehicle verified.

Following burnout of Stage III of MM-1, the spacecraft will coast approximately 180 seconds prior to firing the kick motor. The kick motor will be jettisoned upon completion of its burn. Tracking of the vehicle will be accomplished on the first and next-to-last orbits to refine the prediction of the point of recovery for positioning of recovery forces.

The retro motor will be jettisoned following firing and the recovery package deployed. The complete vehicle will be recovered by air and returned to an East Coast facility suitably equipped to enable removal of the film package. The film will be processed and supplied to the user on the East Coast. The spacecraft with the camera system will be returned to Martin - Denver for refurbishment, test, reassembly and reuse.

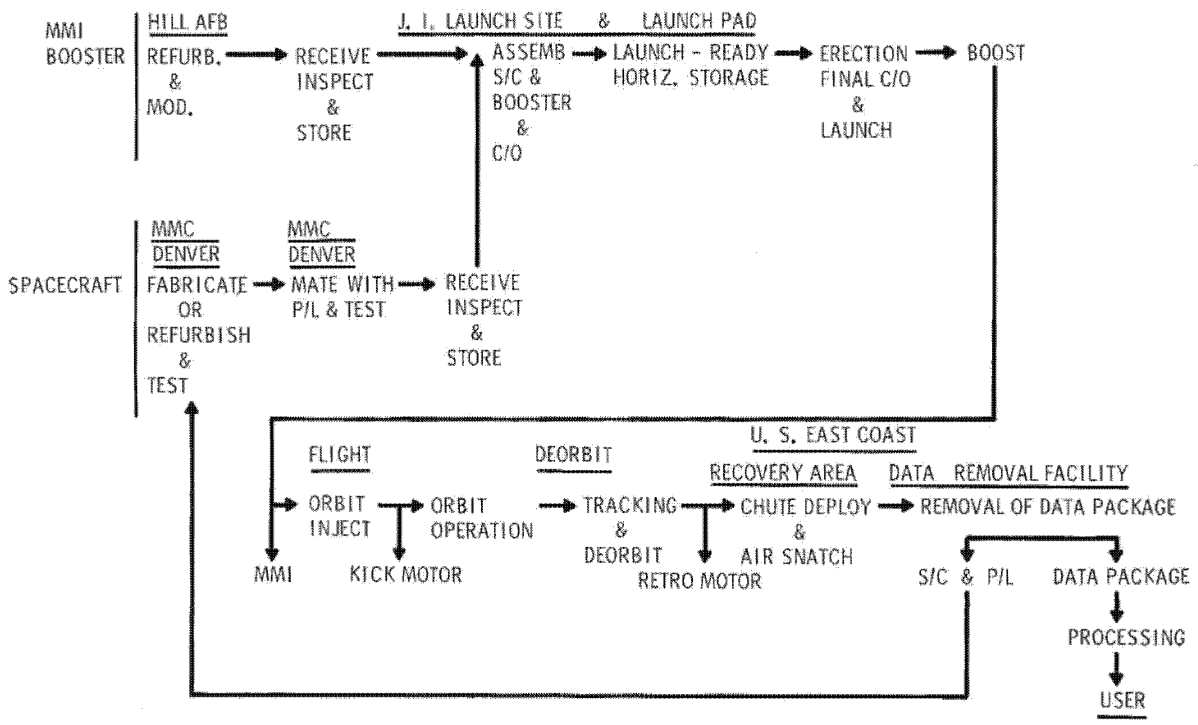

FIGURE I-2 MISSION OPERATION FLOW

II. MISSION CAPABILITY

FASTBACK can be employed with great flexibility. Each mission is affected by many factors including location of the primary target, the coverage mode desired, the pattern of secondary targets and the speed of recovery desired. Fortunately, these factors can be manipulated by the mission planners to place the coverage and shape the coverage pattern to maximize the likelihood of obtaining useful intelligence.

The ground trace for a mission launched on an inclination of 55° with an elliptical orbit (65 x 300 nm.) is shown in Figure II-1.

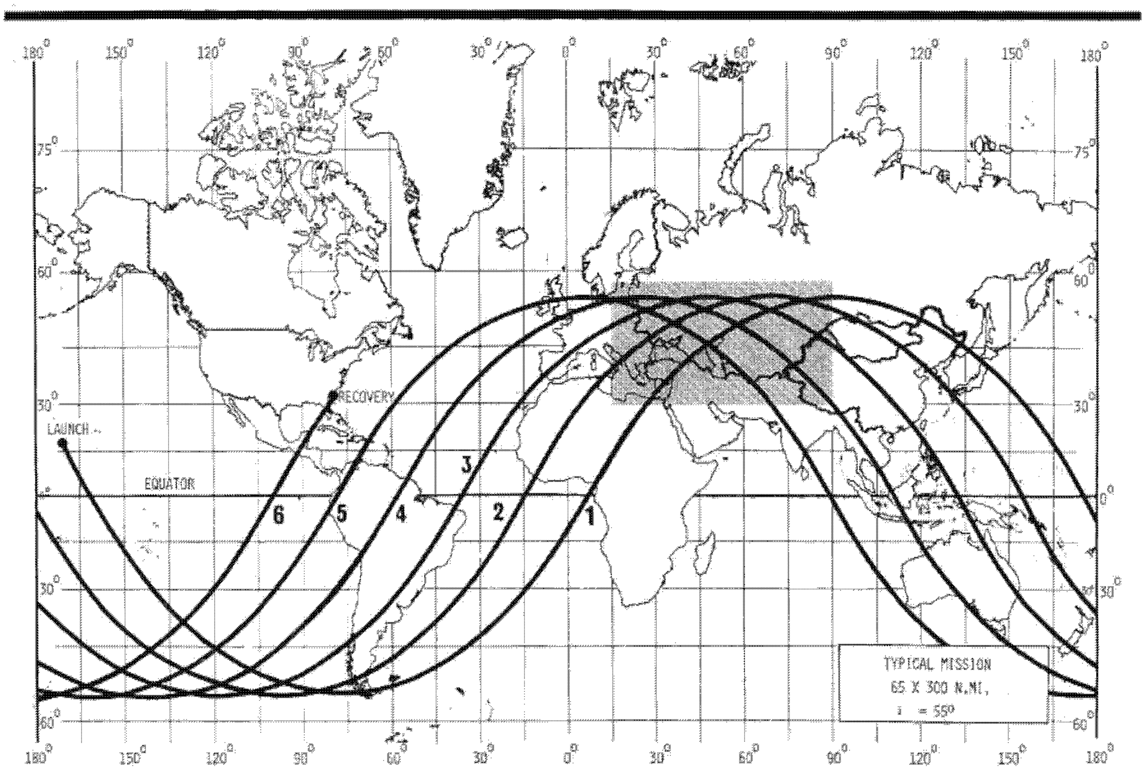

FIGURE II-1 MISSION GROUND TRACE

It shows that the spacecraft was launched from Johnston Island in the Pacific and recovered off the east coast of the United States after six revolutions. FASTBACK is capable of providing stereoscopic photography of a swath 130 nm. wide for a distance 10,000 nm. along track. Figure II-2 shows that a concentration of this coverage could have been obtained in the vicinity of maximum declination where the ground traces of successive revolutions cross.

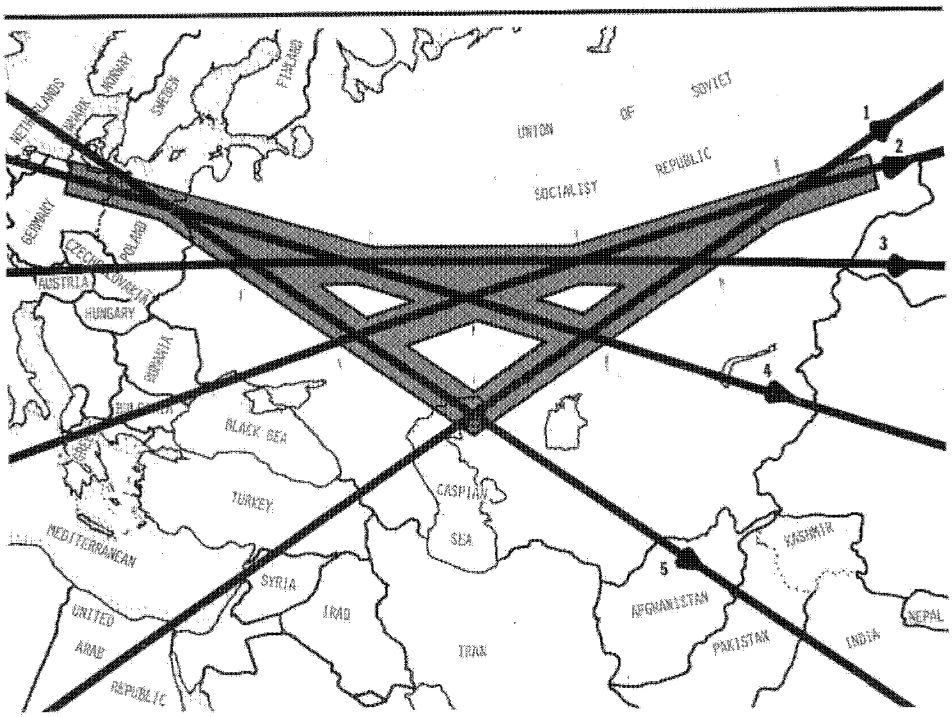

FIGURE II-2 AREA COVERAGE CAPABILITY

On four occasions, successive revolutions crossed at one and one-half hour intervals; on three occasions, alternate resolutions crossed at three hour intervals; on two occasions, revolutions crossed at four and one-half hour intervals and on one occasion, revolutions crossed with an interval of six hours. Each crossing provided the opportunity to get double coverage of an area 130 x 130 nm. or approximately 1700 sq. nm. each.

The figure also shows that in addition to the area of concentrated coverage, the ground traces provide the opportunity for single coverage of long portions of each revolution. The length of useful coverage on each revolution would be limited to that portion occurring in suitable daylight.

The selection of specific camera on and off points would depend on the disposition of the reconnaissance targets falling within access swath along each ground trace.

To cover problems involving differing target arrays, the area of concentrated coverage and the location and direction of individual ground traces can be changed from mission to mission by selecting an appropriate inclination of the orbit. The area of concentrated coverage can be placed anywhere in either northern or southern hemisphere between 17^{o} and 70^{o} and west of the ground trace of the first revolution out of Johnston Island, as shown in Figure II-3.

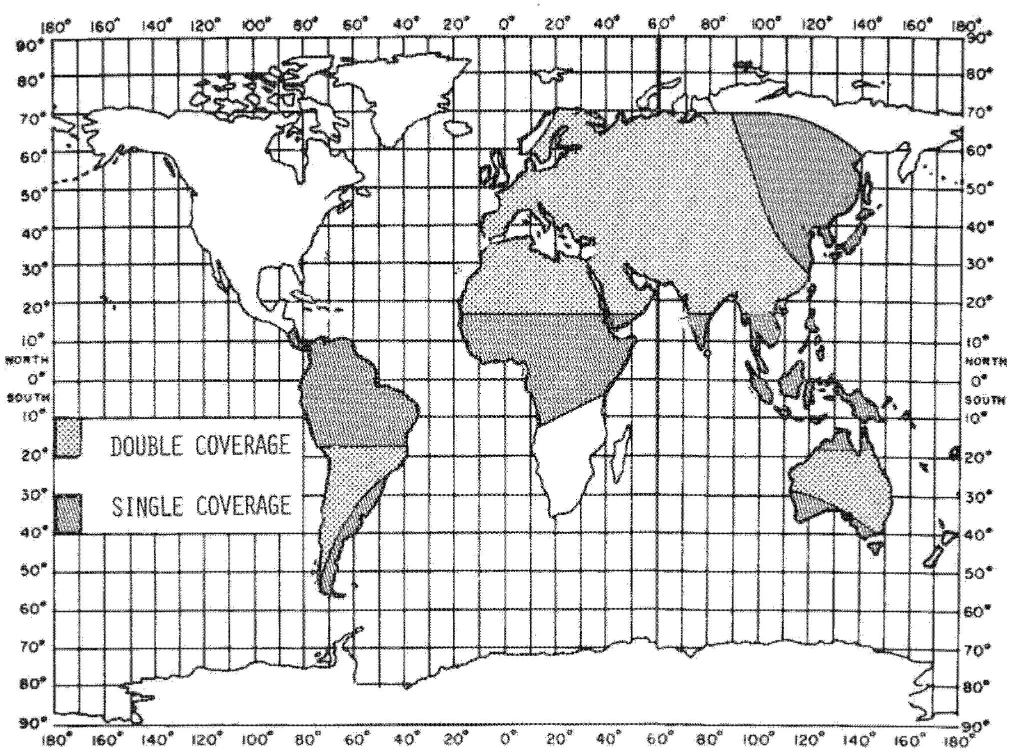

FIGURE II-3 GLOBAL COVERAGE

The area of concentrated coverage can be extended to the west of the first revolution for as many active revolutions as can be used prior to recovery. Orbits at high inclinations (approaching 70°) use only four revolutions before they return to the recovery area. Those at inclinations approaching 17° use nine. Between 17°N and 17°S the system can have access to any point for which coverage is desired but coverage will be limited to the single access swath of individual revolutions. Between 17° and 70° double coverage at one and one-half hour intervals can be obtained for nearly all locations. Double coverage at longer intervals is confined to a north-south zone depending on the inclination of the orbit where alternate (or more widely separated) revolutions cross. The coverage capability actually employed against specific intelligence problems would be the choice of those operating the system.

With the coverage capability described above the FASTBACK system might have been employed in the recent Jordanian situation according to the following scenario:

The President has decided that the hijacking of the four planes with subsequent international blackmail by the Palestine Commandos and the increased threat to King Hussein's Regime, may place upon the U.S. an obligation to act to rescue the hostages being held in Jordan. In advance of a decision to act, certain up-to-date intelligence about the area was required.

Specific targets in Jordan included the following:

Drop zone or landing sites at Zerqa as close as possible to the hostage planes. Area selected to be large enough for an airborne company drop or helicopter landed company operation.

Drop zone or landing site at or near Amman adequate for company size operation.

Photographs of other airfields or landing zones within 50 miles of Amman which could support company size operations.

At or near frontiers of Jordan,location, identification, strength and where possible,state of readiness of Jordanian, Syrian, Iraqi and Israel units.

Identification and strength of guerrilla units in Jordan or Syria paying particular attention to Irbid and Mafraq (also Iradis) in Jordan and Dera in Syria.

Tank headquarters in Qatana airfields at Mezze and Dumayr and any deployment southward and status of PT (OSA Class) Boats in Latakia or Banias in Syria.

In the UAR, cover the Suez deployment plus any major units, with identification, deployed toward Jordan or Israel and status of all military airfields, and naval bases at Alexandria and Port Said.

Photography of the Coast of Israel and Lebanon and interior road net is required to permit selection of landing beaches and routes of communication to the objective area. Sufficient detail to provide terrain analysis is desired.

This scenario would have employed one launch for direct coverage of the area of crisis to provide intelligence for the planning stages of the intervention. Figure II-4 shows the coverage and planning that would have satisfied this requirement.

If the decision had been made to intervene, then from inception of the actual operation additional in-area coverage required would have been furnished by conventional reconnaissance assets. FASTBACK would have been shifted to monitoring the response of the Soviet Union in marshalling and staging areas and in the discernable readiness condition of strategic forces. A possible coverage pattern is shown in Figure II-5. This coverage might have been required for more than one day, depending on developments in the Jordan-Israeli-Syrian area.

If the decision to intervene had not been implemented,

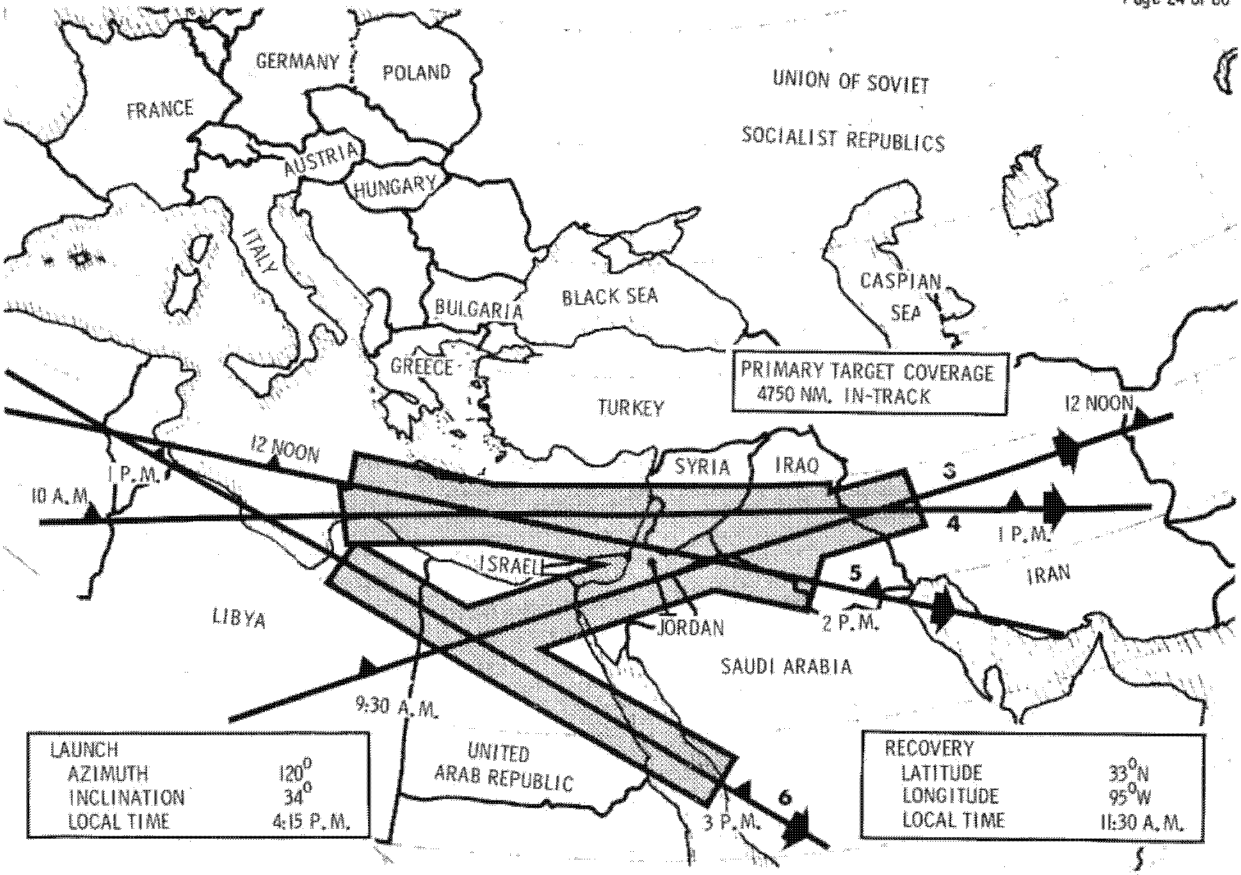

FIGURE II-4 JORDAN-ISRAEL-SYRIA COVERAGE

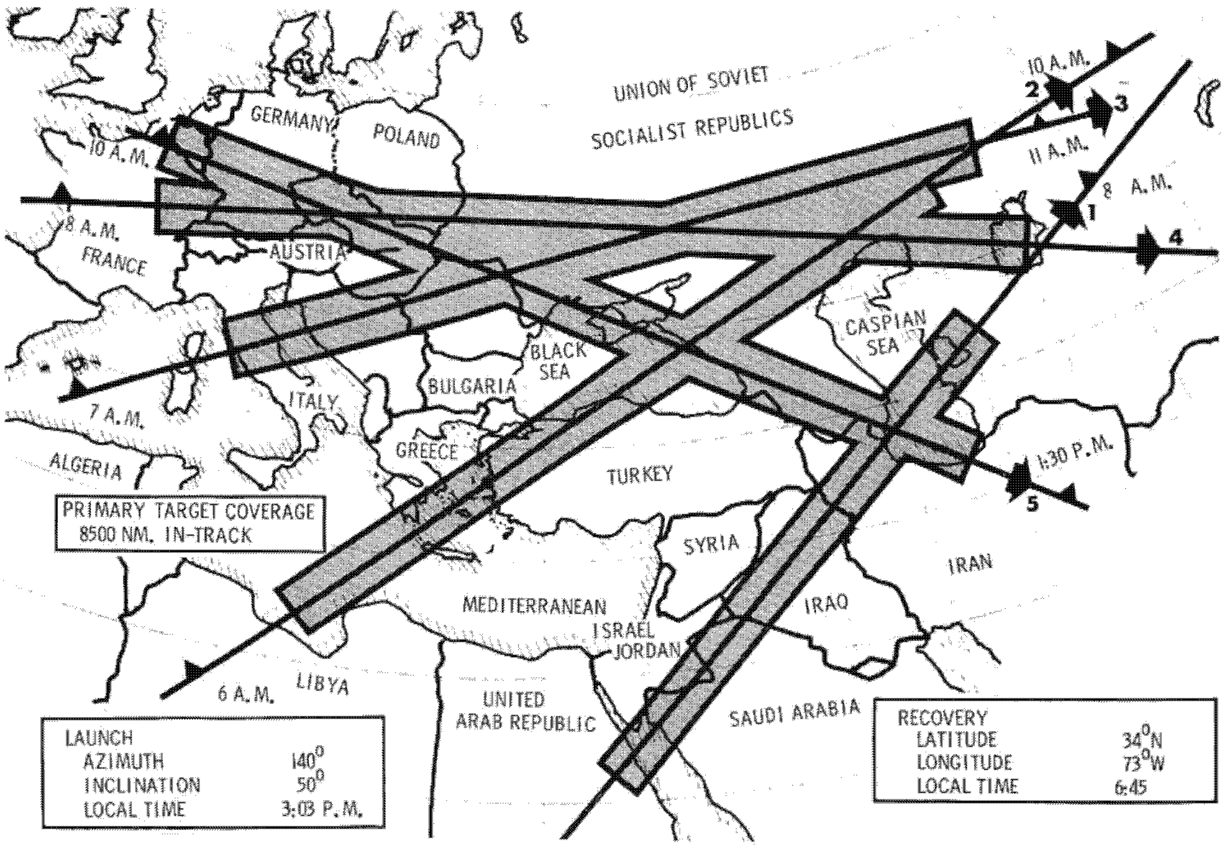

FIGURE II-5 SOUTHWESTERN RUSSIA COVERAGE then FASTBACK could have been used to continue coverage over the confrontation area. The same mission could sample the readiness posture of the Soviet Union and maintain surveillance of the confrontation area. Figure II-6 illustrates a possible coverage pattern to accomplish this.

Examples of FASTBACK coverage of three other intelligence problems are shown in Figures II-7, 8 and 9. Figure II-7 shows the coverage pattern that might have been employed during the Czech crisis. Double coverage was placed over Czechoslovakia with large amounts of film left for coverage of targets in other areas.

Figure II-8 shows the coverage that might have been employed against the Arab-Israeli War of 1967. Again, double coverage was employed.

The final example shows how coverage might have been employed over the Sino-Soviet border dispute. Only single coverage of the area of primary interest is possible because the location is east of the ground trace of the first revolution. Furthermore, coverage is limited to 1300 hours or later local time because of the need to have daylight in the recovery area. On the other hand, much useful supplemental coverage could be placed west of the area of primary interest.

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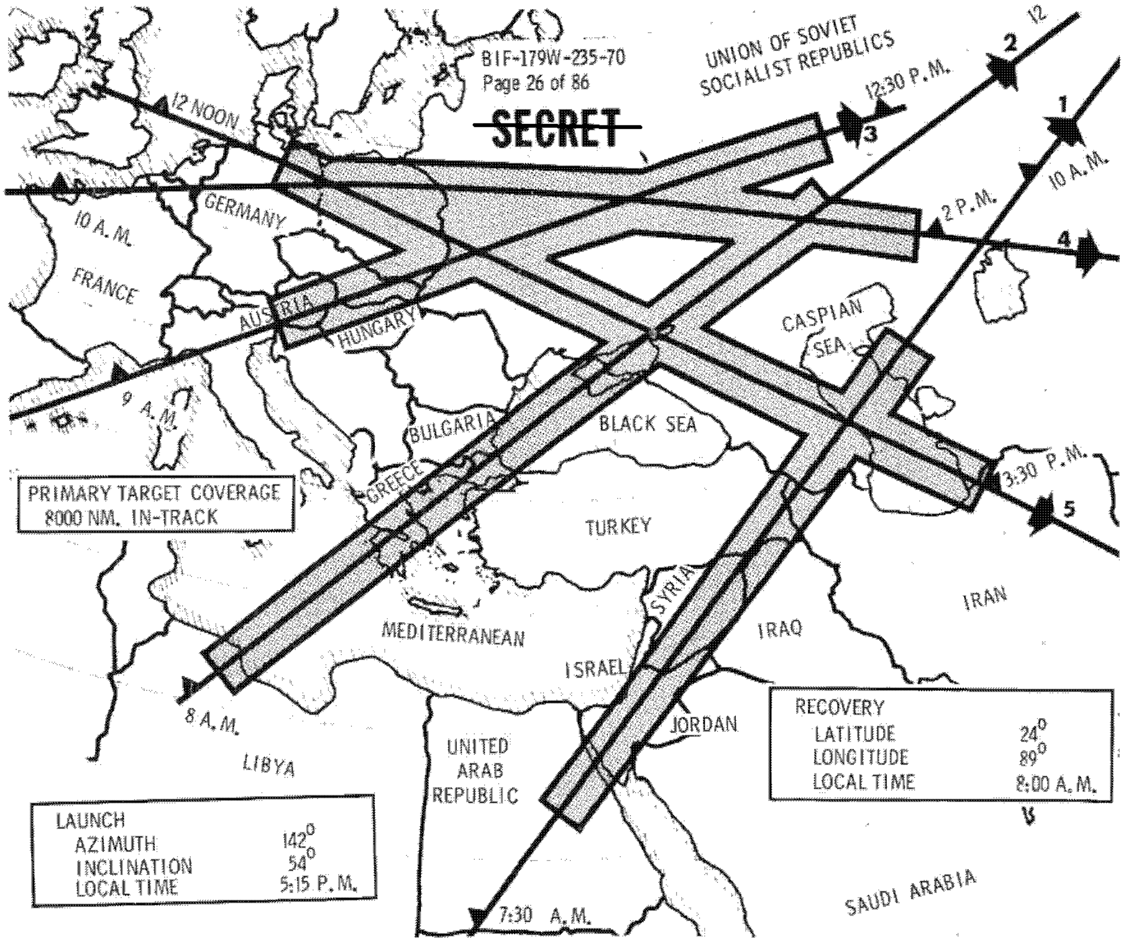

FIGURE II-6 COMBINED JORDAN AND SOUTHWESTERN RUSSIA COVERAGE

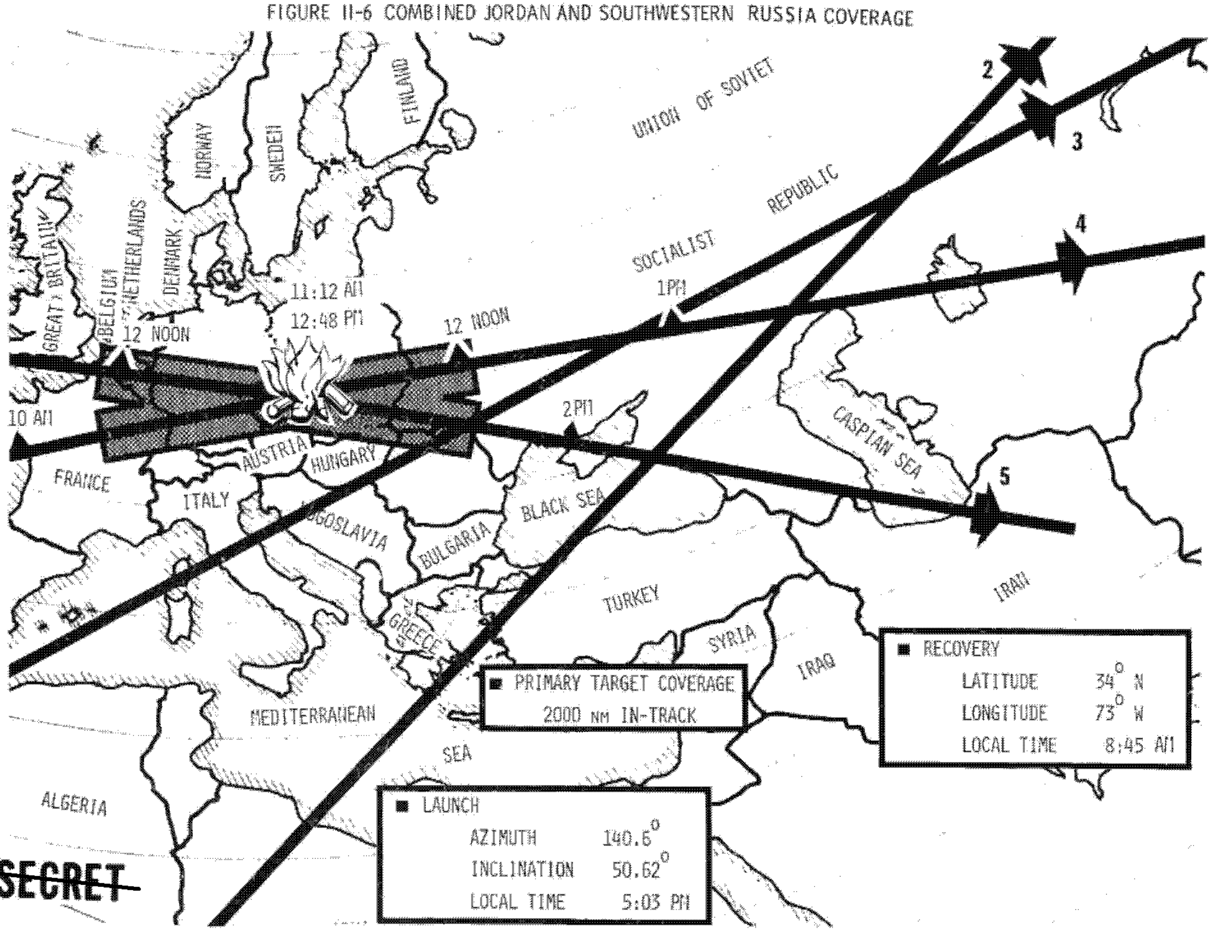

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FIGURE II-7 CZECHOSLOVAKIA CRISIS

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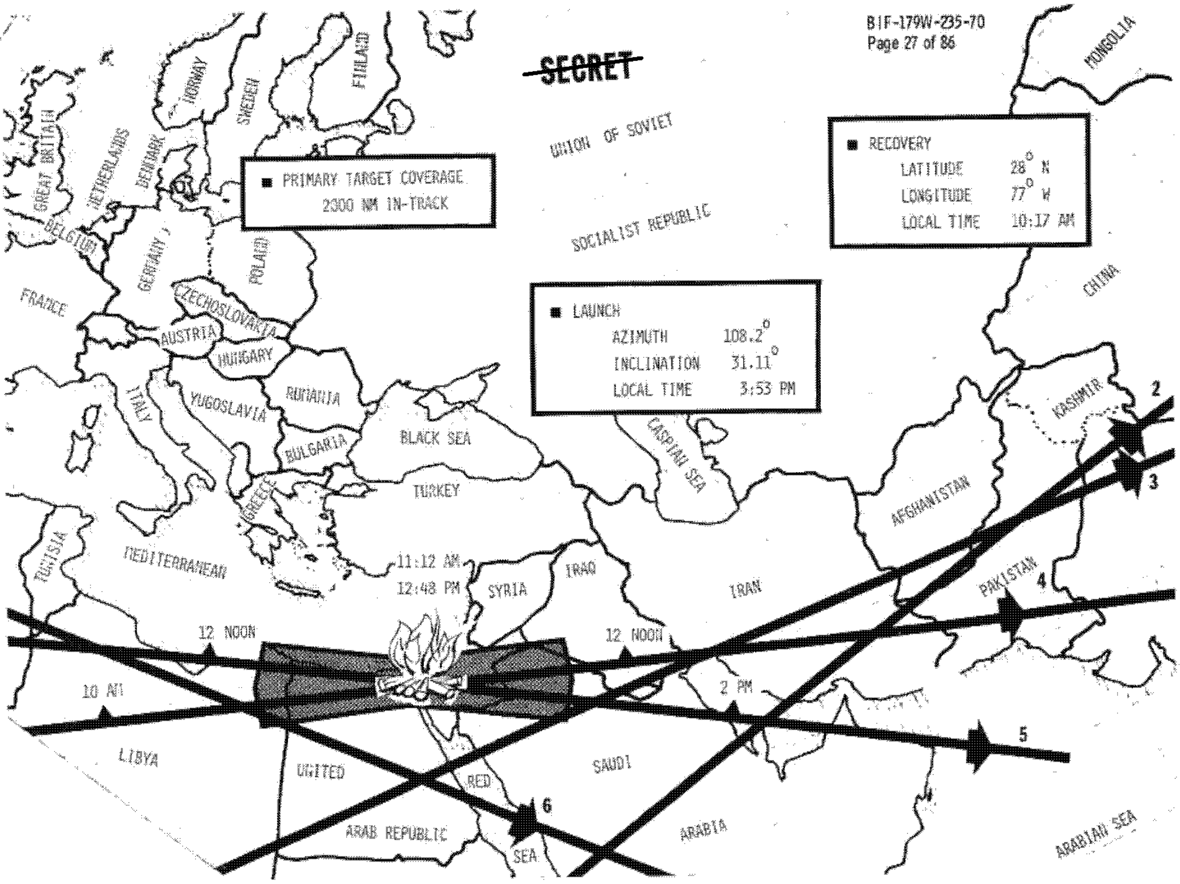

FIGURE II-8 ARAB - ISRAEL CONFRONTATION

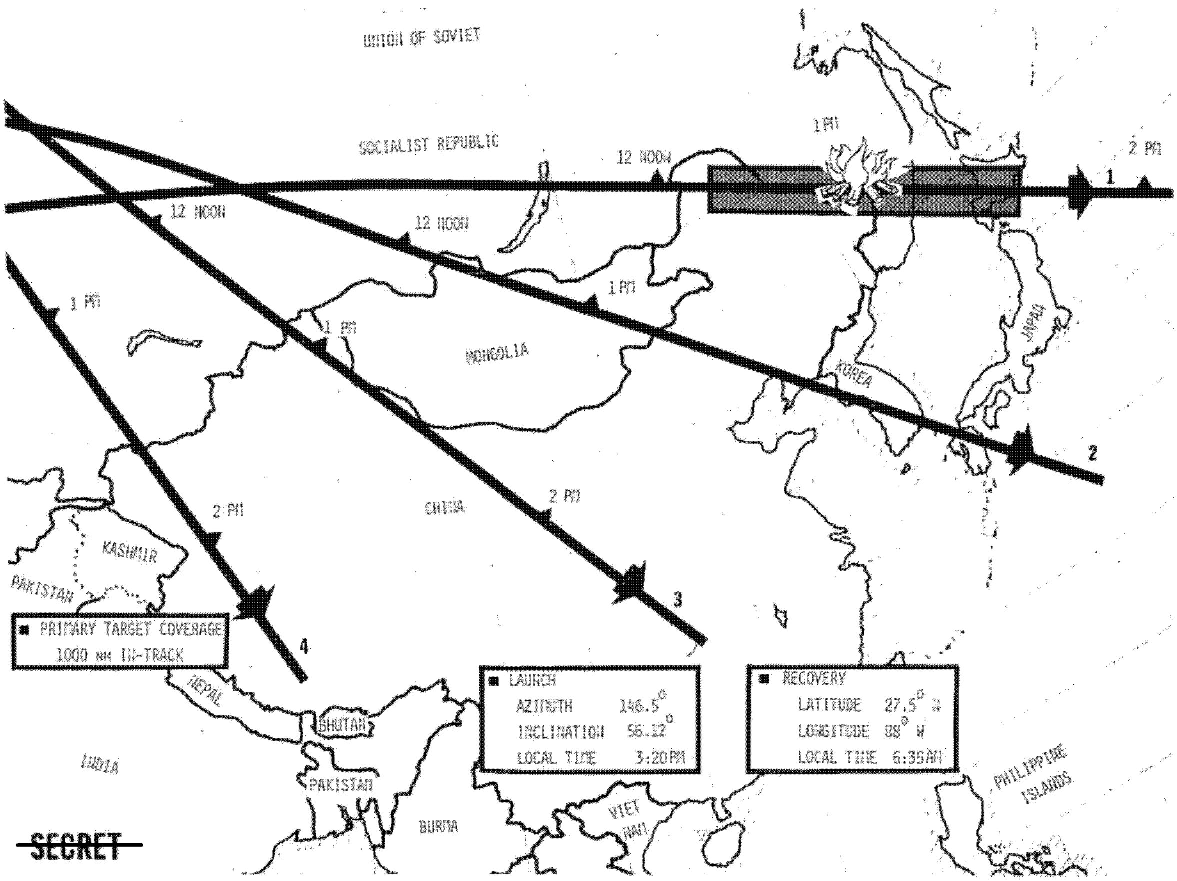

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FIGURE II-9 SINO-SOVIET DISPUTE

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III. SYSTEM RESPONSE

The response of the FASTBACK system is a function of the number of spacecraft in the system, the number of boosters at the launch complex, the launch crew size, the number of launch pads, the recycle time of the spacecraft from recovery to return to the launch complex, and the required refurbishing of the launch pads after a number of launches.

Normally, boosters mated to the spacecraft and checked out will be located on both launch pads. These vehicles will be complete except for attitude control system propellant loading, ordnance igniter installation and target insertion. Additional spacecraft and boosters, when available, are mated, checked out and stored in the Missile Assembly Building.

The minimum time to launch from this status is four and one-half hours. Final target selection can be delayed until three hours before launch.

The targets to be covered will be selected by the user. By using a previously prepared set of indexed coverage maps each representing one degree of inclination, rapid determination of the desired inclination can be made. Each map will show single and double coverage areas, local times along track, retro sequence times, tracking times, and recovery areas and times.

After the user has selected the map that best satisfies his coverage requirements, he will inform the FASTBACK mission control of his selection with specific targets and times. Mission control will then interpolate from the selected coverage map to the exact orbit using precomputed guidance constants for launch and recovery operations to select all constants, sequences and operation requirements.

These mission requirements will then be transmitted to the launch operations at the launch site, to the recovery force and to the tracking organization for immediate mission planning and execution.

With the receipt of the guidance constants and mission timing from mission control, launch operations will select the appropriate tape for the flight computer from the library of tapes that correspond to the 50 coverage maps. A new tape, to vernier the main tape using the interpolated values received, will be punched and verified. Both tapes will be loaded and verified (See Figure III-1).

The actual launch time is a function of the target geography, time-over-target and recovery time desired. However, in general, these factors will ordain a late afternoon launch from Johnston Island. Therefore, for an 1800 hour launch, a launch warning from Washington would be required by 1930 hours Washington, D. C. time, and the target selection by 2100 hours.

The recovery would take place on the Atlantic coast in daylight nine hours (six revolutions) after launch or 0900 in Washington. The film would then be available for viewing by the user five hours later or 18 hours from original launch alert. Figure III-1 portrays this sequence.

In the 48 hours following launch, the launch pad is inspected, refurbished and the stored spacecraft and booster is mated to the erector.

In the meantime the second ready spacecraft and booster are ready for the next launch decision.

This sequence can be continued for two launches from each pad when the capability of the system will be essentially depleted. Then a period of recovery will have to be allowed during which the spacecraft will be recycled and the launch crew rested.

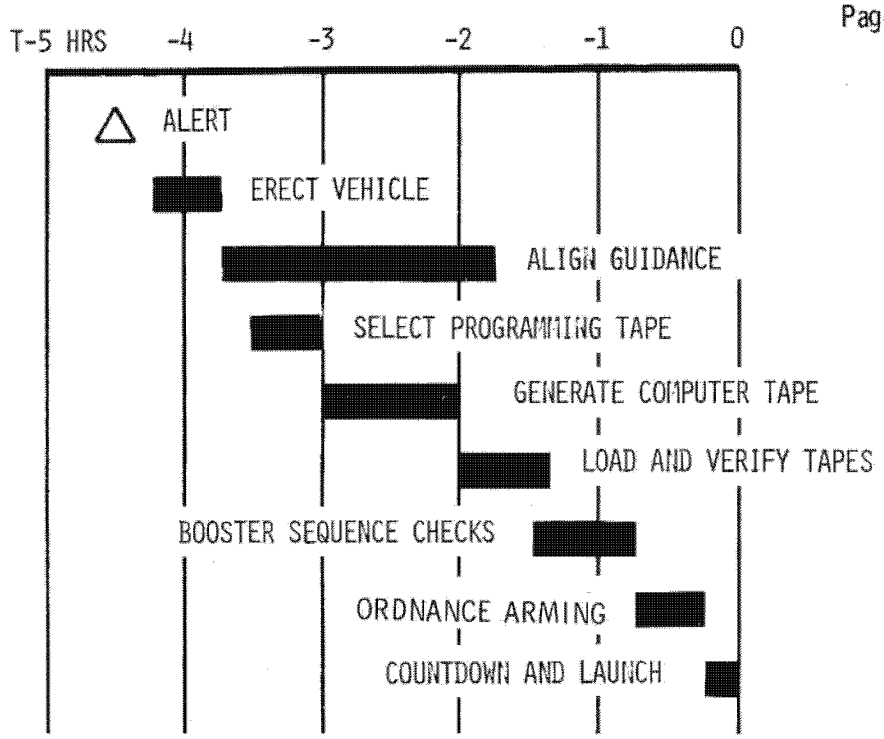

FIGURE III-1 LAUNCH PREPARATION TIMELINE

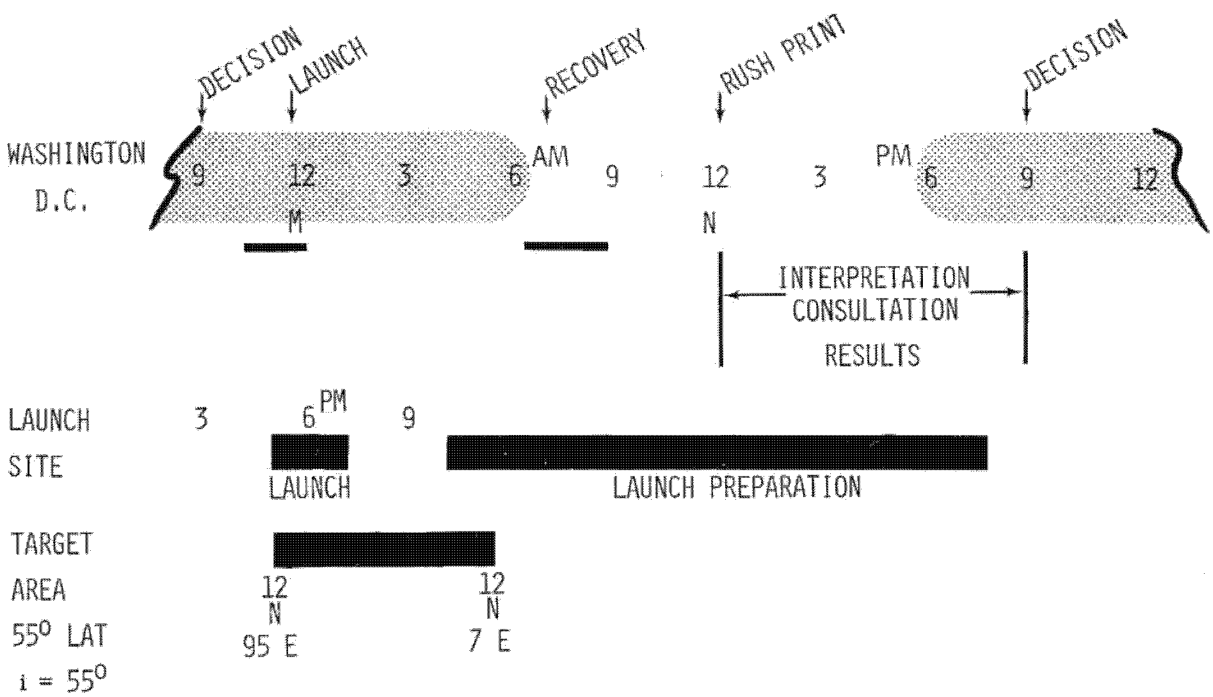

FIGURE III-2 TYPICAL TIMING FROM DECISION TO RESULTS

IV. SYSTEM DESCRIPTION

The spacecraft consists of three basic elements: the operational satellite: the kick motor for injection into a nominal 65 x 300 nm. orbit: and the deorbit/recovery system. The physical interface with the booster occurs at the upper end of the third stage where the guidance bay normally attaches. The overall spacecraft is approximately 16 feet long with a maximum diameter of 37½ in. at the Minuteman interface. This size is compatible with previous loads analyses for possible Minuteman space payloads which show a high launch probability for payloads in this size and weight class. The external shape of the spacecraft (refer to Figure IV-1) was selected considering the respective design criteria for aerodynamic, static, and dynamic stability, minimum drag, internal packaging, external geometric boundary constraints, and low cost fabrication. This shape results in a W/C_DA of about 400 lb/ft^2.

The system weighs approximately 2600 lbs. at launch and 900 lbs. initially on-orbit after kick motor ejection.

A. OPERATIONAL SATELLITE

The operational satellite consists of the optical camera payload system and the vehicle which provides a stabilized platform for camera mounting and control of its pointing. In addition, the vehicle provides electrical power and thermal control for the payload. On the first one or two flights, only a partial load of film will be carried so that an instrumentation system can be included. Each of the subsystems will be discussed in more detail in the following paragraphs.

1. Structural/Mechanical Subsystem - The operational satellite load carrying structure will be essentially monocoque cones and cylinders. Trade-off studies indicate that minimum weight will result from use of magnesium alloys. Thicknesses range from .06 inches to .12 inches.

   An opening is provided in the forward cylindrical section

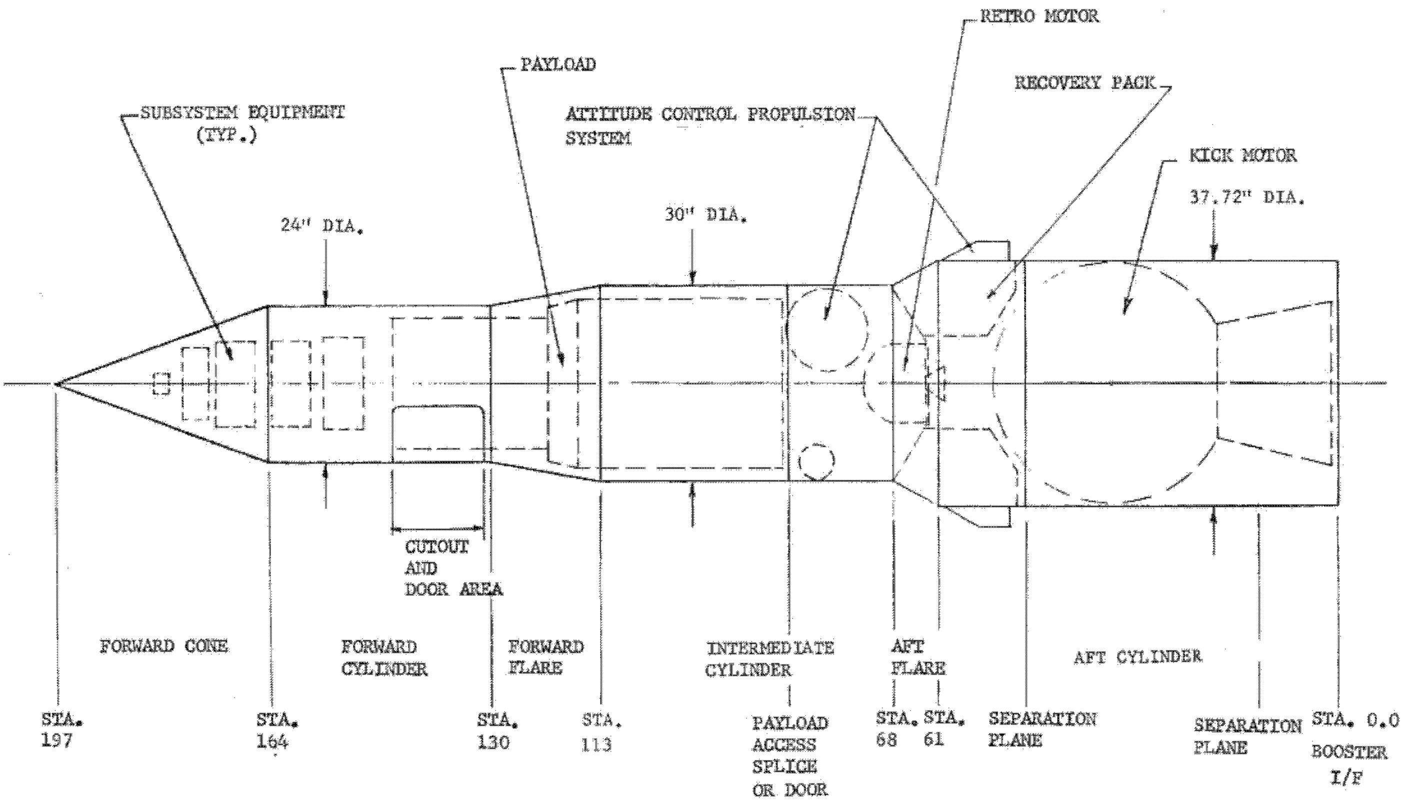

FIGURE IV-1 SPACECRAFT INBOARD PROFILE

to permit light to enter the camera aperture. An opening 14 inches wide by 145° is required. Three permanent struts, approximately 3/8 inch wide will carry loads across the opening. The opening will be covered by a door during launch and reentry.

The reentry heat shield utilizes ablative materials. The spherical nose tip requires a carbon phenolic material. This material has been used frequently on other programs. The remainder of the forward cone and the forward flare conical section utilize a silica based material (3560 HF). This material was previously used successfully on the Martin Marietta Corporation developed PRIME reentry test vehicle. The reentry stagnation heat input to the PRIME vehicle was 161,000 BTU/FT^2 compared to 75,000 BTU/FT^2 for the FASTBACK system. Thickness is approximately 1 to 1.5 inches. The cylindrical sections are protected by another silica based material (SLA 561). This is a low density ablater being developed by Martin Marietta Corporation for the Viking Mars Lander vehicle. Its thickness is approximately one inch. The aft conical section is protected by a third silica based ablator (ESA 5500). This material was also successfully demonstrated on the PRIME vehicle.

Access to internally mounted equipment is provided at circumferential splice locations.

Internal arrangement of equipment places the electrical power and guidance components in the forward cone and half of the forward cylinder. The payload extends through the rear half of the forward cylinder, the forward flare, and two-thirds the length of the intermediate cylinder. That portion of the intermediate cylinder aft of the access splice, the aft flair, and the portion of the aft cylinder forward of the kick motor separation plane contain the recovery system and attitude control/vernier propulsion system. The reaction jets and vernier engines are externally mounted on the aft cylinder just forward of the kick motor separation plane.

2. Guidance and Control Subsystem - The Guidance and Control Subsystem, shown in Figure IV-2, will satisfy the following requirements:

- Provide for launching at any desired azimuth in the launch sector.
- Provide guidance steering and attitude stabilization command signals during boost.
- Provide spacecraft attitude control and stabilization during coast, ΔV maneuver, orbital operations and deorbit.
- Provide sequencing discrete signals during boost, camera operations and deorbit.

G&C hardware consist of a Strapdown Inertial Measurement Unit (IMU), a digital Guidance Computer (GC), Input/Output (I/O) Adapter, and Valve Drive Amplifier (VDA) Unit. To support and check out this airborne system, associated Auxiliary Ground Equipment (AGE) and supporting flight and laboratory software will be provided as described in Section VIII.

a. Inertial Measurement Unit - The IMU will sense incremental angular displacements about the vehicle axes and will sense velocity increments along the vehicle's orthogonal axes. These data will be supplied to the digital computer in the form of discrete pulse trains which indicate changes of angle and/or velocity.

The IMU consists of three gyros, three accelerometers, six pulse torquing servoamplifiers, frequency countdown unit, warmup and fine temperature control amplifiers, crystal oscillator, Porro prism, power supply and housing assembly. Thermal control will be passive with the

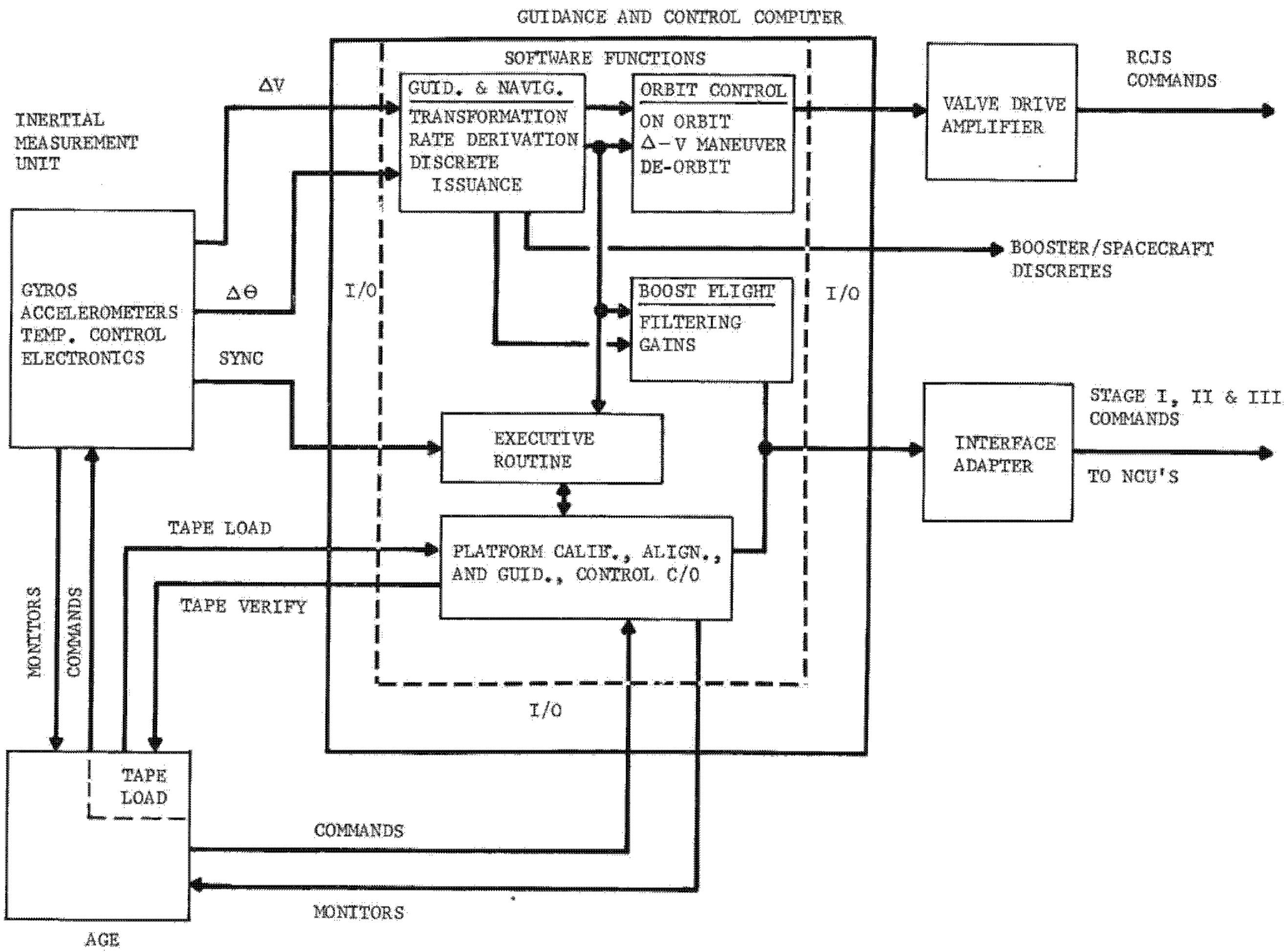

FIGURE IV-2 G & C FUNCTIONAL BLOCK DIAGRAM

IMU mounted on the spacecraft cold plate. The IMU was developed initially for the Apollo Lunar Module. It is currently being adapted to the Delta Launch vehicle. Qualification Tests will be completed in June 1971.

b. The Guidance Computer (GC) will be a general purpose, stored program machine designed spacifically for space and boost vehicle environments. The computer subassemblies consist of a memory assembly, power support assembly, and logic section assembly. The computer memory size is 4,096 words (expandable to 16,384 in increments of 4,096 words) with a word length of 24 bits including sign. Current estimates indicate that FASTBACK will require approximately 8,000 words of memory.

Thermal control will be passive with the GC mounted on the spacecraft cold plate. The GC will be qualified for the Delta application by April 1971.

c. Input/Output Adapter - The I/O Adapter will accept the Guidance Computer analog pitch, yaw and roll thrust vector commands, and process these electrical signals and provide the appropriate commands to the Minuteman Stage I, Stage II, and Stage III nozzle control units. This adapter will be an analog unit and will be mounted on Stage III of the Minuteman vehicle since it will be used only during the boost phase.

d. Attitude Control System Valve Drive Amplifiers These units will accept the Guidance Computer on-off command signals and provide the power amplification to drive (turn on-turn off) the ACS jet nozzles. This

Guidance System I/O equipment will be mounted in the spacecraft.

During the boost phase, the G&C will provide the guidance steering and attitude stabilization command signals to the Stage I, II, and III Nozzle Control Units (NCU). The NCU's respond to provide the booster thrust vector control. The G&C will also provide the sequencing discrete command signals during the boost phase.

Attitude control and stabilization of the spacecraft during coast phase following separation from the booster and during the ΔV maneuver (kick motor and vernier propulsion operation) will be provided. Electrical on-off commands to the Attitude Control System (ACS) reaction jet nozzles (solenoids) will provide the pulsed thrust for attitude control. The ΔV maneuver will attain the required tangential orbital velocity within $\pm$ 12.0 ft/sec.

The G&C will also provide electrical on-off command signals to the ACS reaction jets to provide attitude and rate control of the spacecraft during the on-orbit operation. In addition, it will issue discrete electrical signals to provide sequencing that includes on-off commands to the mission equipment. During the mission on-time, the spacecraft attitude will be maintained to within $\pm 0.5^{\circ}$ (2σ) of the reference attitude, and the limit cycle rates will be limited to less than $\pm$ 0.1 deg/sec (2σ).

The G&C will provide spacecraft attitude control and sequencing discretes for deorbit activities up to the time of reentry. Barometric signals will be used for parachute deployment.

3. <u>Attitude Control/Vernier Propulsion Subsystem</u> - The Attitude Control/Vernier Propulsion Subsystem will provide control torques for the vehicle prior to orbit insertion through orientation for reentry. The specific phases of the mission and control requirements are:

. Initial stabilization after release from booster. Low level thrust in roll, pitch, and yaw to offset tip-off rates.

. Solid rocket kick motor firing. Low level thrust for roll and high level thrust in pitch and yaw to offset solid rocket motor thrust/C.G. misalignment. Vernier velocity for orbit trim. Low level thrust for roll control, high level thrust for velocity makeup.

. On-orbit vehicle attitude control. Low level thrust in roll, pitch and yaw to counteract external disturbances.

. Reorientation for deorbit and entry. Low level thrust in roll, pitch and yaw for reorientation for deorbit motor firing and entry attitude.

. Deorbit solid motor firing. Low level thrust for roll control and high level thrust in pitch and yaw to offset deorbit solid rocket motor thrust/C.G. misalignment.

A schematic of the proposed system is shown in Figure IV-3. Low level thrust to satisfy the roll, pitch and yaw requirements is provided by eight 1 lb_f thrusters using gaseous nitrogen. Bi-directional roll control is provided by two thrusters per direction acting in couples. Pitch and yaw control is provided by one thruster per direction per control mode. The thrusters are operated in a pulse mode with control provided by a direct acting normally closed solenoid valve for each thruster. Nitrogen is provided to thrusters at a constant regulated pressure so the thrust level remains nearly constant throughout the mission.

High level thrust is provided by four monopropellant

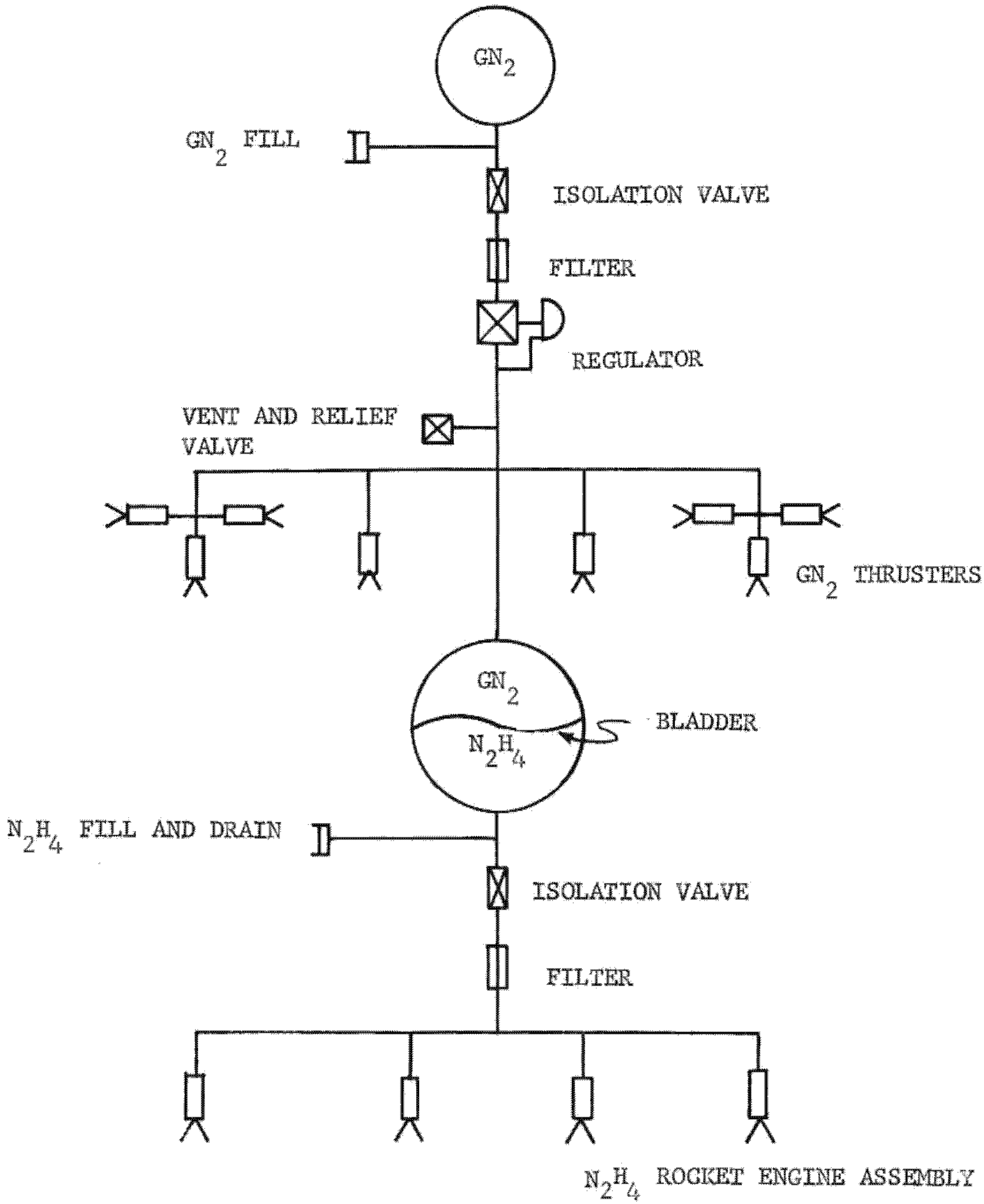

FIGURE IV-3 ATTITUDE CONTROL VERNIER PROPULSION SYSTEM hydrazine (N_2H_4) rocket engine assemblies (REA) each delivering 150 lb_f thrust. A single REA is used for each direction in pitch and yaw. The REA incorporates a valve for propellant flow control. The REA can be operated in pulse mode or continuous burn. Pressure regulated nitrogen pressurizes the hydrazine tank and a bladder provides positive expulsion of N_2H_4 during low-g coast conditions.

4. Thermal Control Subsystem - The Thermal Control Subsystem must maintain all on-board equipment within acceptable temperatures. Although specific temperature limits have not been established for all equipment at this point, it is expected that control of most equipment between 0 to 100°F will be required. Specific items, such as the battery, rocket motors and the parachute may have more definitive requirements. The payload optics will have extremely restrictive requirements. A $\pm$ 5°F range about nominal mean must be maintained. In addition, the temperature gradient across the payload optical elements must be limited to 2° max.

The relatively large quantities of internal heat dissipation require a semi-active system. An evaporative cooler system was selected on the basis of simplicity, weight savings, cost, and demonstrated reliability. Individual coolers are placed on the guidance computer and reference unit, guidance adapter electronics, the battery, the power supply, and the payload. Water, contained in the cooler packages, is vaporized under heat loads, thus very effectively removing heat energy from the equipment packages. The evaporators are manifolded together, dumping through a common exit at the aft end of the vehicle. An orifice at the exit will control back pressure in the system to about 0.5 psi. This corresponds to a saturation temperature of 80°F.

Each evaporative cooler will be similar in design, although some size variations are expected in order to accommodate the several equipment items. A cooler will consist of a lightweight structural shell containing a porous wick material. The wick holds the water under zero gravity conditions and will pump the water to the heated surface to replenish that water vaporized during operation.

The cooler performance will be enhanced by the use of insulation on the vehicle inner skin. This material will minimize the soak-back and electromagnetic heat loads. The cooled equipment will be mounted with low conductance hardware to minimize conduction from the heated structure.

In addition to the cooler, the payload will require local electrical heaters. These items will be controlled by temperature sensors on or near the optical elements to achieve the $\pm$ 5°F tolerance.

Thermal control during ground and pre-flight checkouts will be achieved by an active purge system. Cool, dry nitrogen will be injected in the forward end of the vehicle by a duct system. Resultant flow will be back around the equipment, carrying dissipated heat aft, where it is dumped overboard.

During reentry the vehicle surface becomes warm and the ablator and vehicle inner skin absorb heat. Consequently, heat soak back into the vehicle could cause equipment overheating.

The results of a transient thermal analysis show that 1/2 hour after recovery, the ablator surface temperature is less than 100°F and the inner skin of the vehicle is less than 150°F. Heat transfer into the vehicle is inhibited by the insulation on the inside of the vehicle which provides a nearly adiabatic surface.

The primary development item in the thermal control system is the evaporative cooler. Previous Martin Marietta Corporation experience with similar units on the PRIME reentry test vehicle provides a development background which will be applied to FASTBACK.

5. Electrical Power Subsystem - The electrical power subsystem, Figure IV-4, consists of a primary Silver-Zinc battery, an Ordnance and Electronics Unit (OEU), and a wiring harness with associated clamps and connectors.

The silver-Zinc battery will be fabricated from low rate, manually activated, primary AgZn sealed cells. Based on the present load profile for an 864 minute (14.4 hour) mission, the battery will be rated 230 AH for operation at 80°F. This provides approximately 13% margin. Under such operating conditions the battery terminal voltage will be 29 ± 4 vdc for the entire mission.

The OEU houses ordnance firing circuits, capacitor banks, and other electronic and switching functions as required. The OEU is a new design and build item but technology from previously developed designs is applicable. Eleven ordnance functions have been identified to date, as shown in Figure IV-4. A capacitor bank is used in conjunction with an SCR to fire each ordnance bridgewire. No single ordnance event requires more than two pair of bridgewires; each capacitor bank can be recharged in less than three seconds and no two ordnance events are less than three seconds apart. Consequently, four capacitor banks are required for all ordnance events. The capacitor banks are charged by a 40-volt square wave signal which is formed by an inverter in the OEU.

6. Electronics Subsystem - The following electronic equipment will be utilized in the booster and the satellite system.

- Flight Termination System (Booster)
- Tracking Beacon
- Instrumentation System (First Two Vehicles Only)

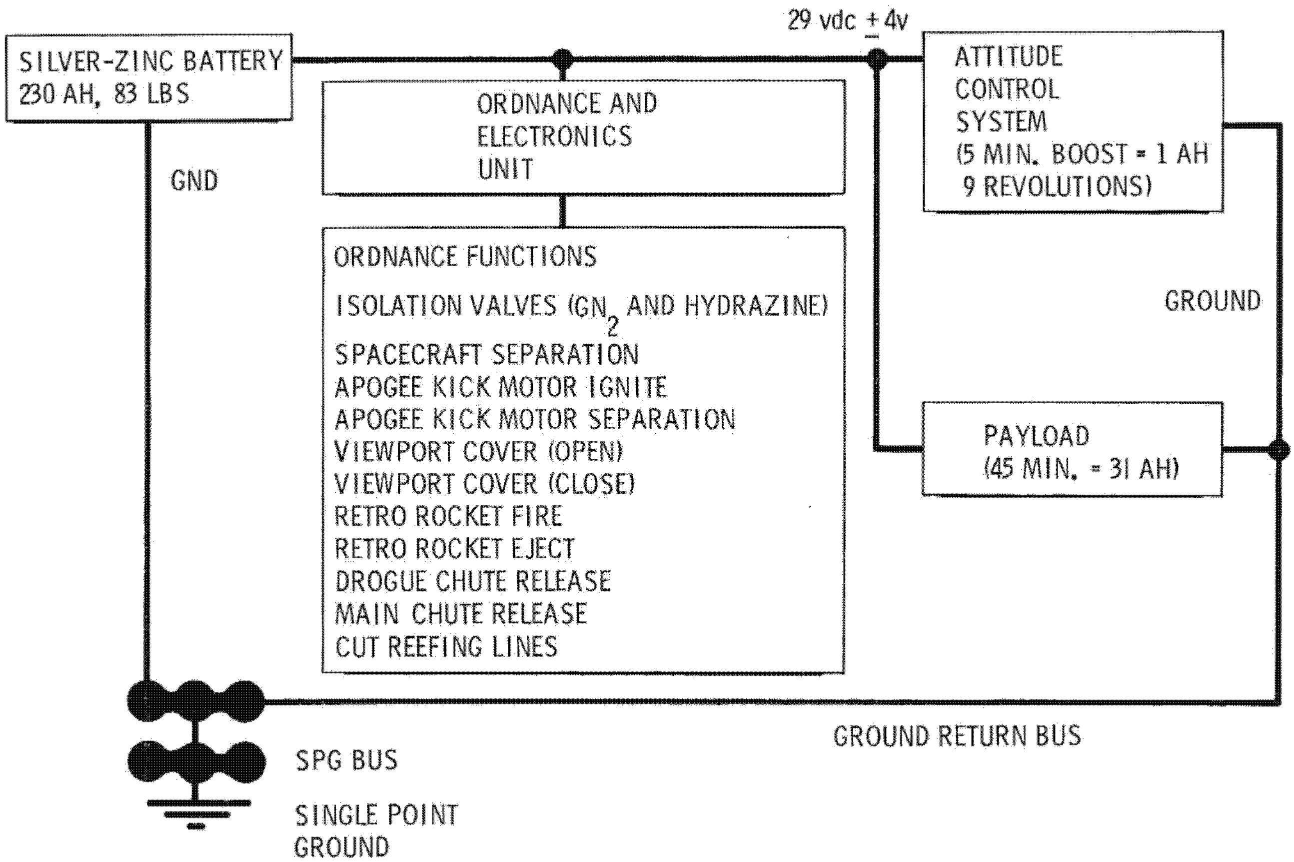

FIGURE IV-4 ELECTRICAL POWER SUBSYSTEM

Booster Flight Termination System - Two separate and independent radio command destruct receivers will be provided. The RF system (consisting of two antennas and a four port junction), the battery, and the flight termination ordnance will be common to both receivers. The antenna system will provide essentially omnidirectional coverage over 70% of the radiation sphere. The signal from the command receivers will go directly to the flight termination ordnance with no intervening switching device. The flight termination system will be located in the booster and will be used solely for range safety purposes.

Tracking Beacon - A tracking beacon, compatible with ETR downrange tracking radars, will be added to the orbiting vehicle as an aid in orbital tracking for accurate determination of the recovery footprint. The beacon, which is a C-band transponder, will operate through a hemispherical antenna system.

Instrumentation System - An instrumentation system will be carried on the first two flights. Transducers (primarily temperature and vibration sensors) will collect all necessary design verification measurements. The transducer outputs, along with subsystem monitoring measurements, will be multiplexed for relay to the ground by a telemetry transmitter via an omnidirectional antenna system. The data multiplexing system and telemetry transmitter will be selected to be compatible with existing ground instrumentation facilities.

B. KICK MOTOR

A velocity increment is required after Stage III burnout to place the satellite in the nominal operational orbit. The required 410,000 lb_f-sec. total impulse will be provided by an existing solid rocket motor. The Thiokol Chemical

Corporation TE-M-364 motor fulfills the total impulse requirements as well as those for minimum physical envelope (no greater than 37.5 inch diameter) and low acceleration (less than 10 g's). Design data for vacuum conditions at 70^{o}F are:

Total Impulse, lb_f-sec	418,500
Specific Impulse, lb_f-sec/lbm	290.5
Burn Time Average Thrust, lb_f	9400
Burn Time/Action Time, sec	42.6/44.3
Loaded Weight, lbm	1574
Burnout Weight, lbm	126

The motor is spherical in shape and features one fixed, submerged nozzle. Modifications to the motor mounts will be required. Total impulse uncertainty is such that a velocity vernier will be required. This will be provided by the ACS propulsion system described in Section IV.A.3.

Although the motor was originally designed for Surveyor, it has subsequently been qualified in an uprated version for the Burner II and Delta upper stages. The motor is currently in production and has been fired 88 times and flown 25 times. A growth version of this motor (total impulse = 650,000 lb_f-sec) is currently being developed by Thiokol for NASA.

C. DEORBIT/RECOVERY SYSTEM

The maneuver to deorbit the vehicle will require a total impulse of approximately 7500 lb_f-sec. The existing Thiokol TE-M-236 motor fulfills this requirement. The following table summarizes the motor design data for vacuum conditions at 60^{o}F.

Total Impulse, lb_f-sec	10,500
Specific Impulse, lb_f-sec/lbm	262
Burn Time Average Thrust, lb_f	1,250
Burn Time/Action Time, sec	7.5/9.6
Loaded Weight, lbm	60.6
Burnout Weight, lbm	20.7

This motor was developed and qualified for use on the Discoverer Program. There have been 372 production motors delivered and nearly all of these have been used in flight articles.

An aerodynamically stabilized entry of the spacecraft is proposed. The aeroshell shape and materials selection have previously been described.

After reentry, deceleration of the spacecraft is accomplished by a ballute and parachute deployed in three stages. Deployment of the first stage ballute will be accomplished by ejection of the aft thermal cover at minimum q ($\approx$ 100,000 ft). Aerodynamic drag on the aft cover will extract the ballute pack from the vehicle. The deployed ballute will provide initial atmospheric deceleration of the vehicle. The second and third stages of deceleration are accomplished by a single main parachute which is reefed during the second stage of deceleration and fully opened during the third stage. Deployment of the main chute will be accomplished in a manner similar to that used for the ballute. Ballute drag forces will be utilized to extract the main chute pack from the vehicle at 50,000 ft. After a time interval of approximately 4 sec. the main chute will be unreefed permitting full inflation of the chute. The fully inflated canopy configuration will provide sufficient stability ($\pm 3^{o}$) and drag to achieve the descent rate (30 fps at 10,000 ft.) for air snatch retrieval of the operational satellite.

The recovery subsystem will include a VHF radio homing beacon which aids in acquisition and retrieval if the C-Band beacon is not compatible with the recovery A/C. Power to the recovery beacon will be switched on immediately prior to the entry sequence. This system will work in conjunction with a receiver system located in the recovery aircraft to facilitate a mid-air recovery. Also, alternate gores of the main parachute are colored international orange to assist in visual tracking during final descent.

This type of recovery subsystem has successfully flown on the PRIME recovery vehicle and other existing programs. Little modification is necessary to adapt these systems to our proposed system.

The parachute recovery system is packaged in a torroidal configuration around the deorbit motor. The beacon is located in the nose of the satellite.

V. SENSOR DESCRIPTION

A panoramic optical bar camera is required for the FASTBACK system in order to obtain broad stereo coverage. The optical bar camera should weigh less than 275 pounds and provide a ground resolution of three to four feet at nadir. Final camera selection will also be based upon the use of existing space qualified hardware and proven technology.

Three camera manufacturers were contacted regarding installations for this spacecraft and they have shown that these objectives can be met. Their results are summarized on Figure V-1. Note that all three camera manufacturers are basically recommending the same optical design. The major difference in the weight of the Fairchild system resulted from their recommendation that the spacecraft structure be used as a light-tight container and as a reaction member. This eliminates the need for a 40 pound camera case and support structure. The predicted ground resolution at the 50% probability level is shown to be 3.5 feet or less for all three systems. This performance is based upon the use of a 5. inch wide SO-349 thin base film.

MANUFACTURER	FAIRCHILD	HYCON	ITEK
TYPE	PANORAMIC (OPTICAL BAR)	PANORAMIC (OPTICAL BAR)	PANORAMIC (OPTICAL BAR)
FOCAL LENGTH	27"	24"	24"
RELATIVE APERTURE	F/4.0	F/3	F/3.5
WEIGHT	204#	250#	265#
GROUND RESOLUTION (50%) @ 65 NM.	3.5 FT	3.2 FT	3.3 FT

FIGURE V-1 SUMMARY OF OPTICAL SUBSYSTEM

The three camera manufacturers have stated that no new "state-of-the-art" assemblies or techniques are required for their designs and they already have directly related experience in the design of the various elements being proposed. In all cases the single long lead item is glass. Therefore, this item is included in our schedule as a long lead item which would be ordered shortly after program go-ahead.

Optical systems similar to the ones proposed have been built and flown in reconnaissance aircraft. One option is the ITEK aircraft camera (KA-80A) which is very similar to the one they are proposing for this installation. Their system will require slight modifications to meet the space environment and a lens change to obtain increased ground resolution. Other optical manufacturers' options are also currently being considered.

Overall system performance is shown on Figure V-2 for the three proposed camera systems. The cross hatched area bounds the predictions of all three manufacturers. These predictions are based upon 65 nm. at nadir and spacecraft attitude and attitude rates of $.5^{\circ}$ and $.1^{\circ}$/second (2σ) respectively. The small focal length of the FASTBACK system coupled with the low orbit altitude minimizes the atmospheric effects on imaging, when compared to larger focal length systems.

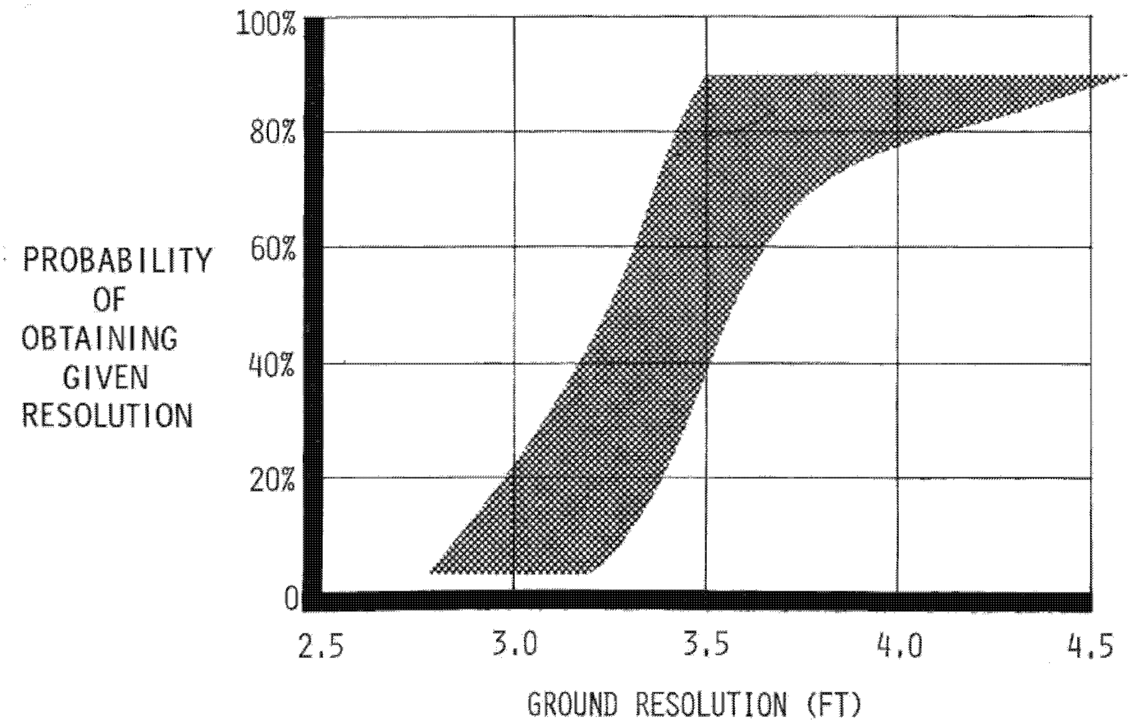

FIGURE V-2 FLIGHT PERFORMANCE PREDICTION

VI. REFURBISH AND REUSE

BASIS FOR SELECTION

The FASTBACK Program is based on maximum recovery, refurbishment and reuse of spacecraft hardware. Since air recovery capability exists for vehicles of the size of the FASTBACK spacecraft, it is practical to recover, refurbish (as required), and reuse the complete orbiting vehicle.

The incremental testing and refurbishment labor beyond that required to test and assemble a new vehicle is not expected to be significant because the initial design is carried out with reuse in mind. Refurbishment at the spacecraft level is performed by replacement at the black box or component level. Refurbishment turn-around times, costs for refurbishment, and reuse economic cross-over points which have been determined for FASTBACK are compatible with results of studies performed by others.*

REFURBISHMENT AND REUSE PHILOSOPHY

The recovered vehicle, following film removal, will be flown to Denver for refurbishing. Upon receiving the spacecraft, activities will be performed to observe and document obvious degradation due to mission environment. A Go-No-Go test to determine status of recovered subsystems will be run prior to refurbishment operations. Subsystem tests will be performed on all functional equipment to establish internal and interface integrity, parametric performance and/or malfunctions to the black box level. Malfunctioning items will be replaced from spare stock and bench maintenance on removed components will be performed on a noninterference basis with spacecraft recycle operations.

During the subsystem checkout process, selected parameters will be monitored and integrated into existing data that has been compiled

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* Recovery, Refurbishment and Reuse of Spacecraft, SAMSO-TR-67-22 (North American Aviation Report SD67-959)

from previous testing to establish trends, so that the equipment future operating capacity can be predicted based on quantitative data.

Providing all subsystem performances are within specifications, including trend limits that have been established, the spacecraft will pass on to the subsequent testing to establish flight readiness; i.e., Production Environmental Test (PET), Combined Systems Test (CST), alignment (including weight and balance), Hardware Acceptance Test (HAT), and Pack and Ship activities.

TEST AND CHECKOUT

Test and checkout operational flow for both initial development/ manufacture and recycle/refurbishment is shown in Figure VI-1. Test activities at Johnston Island are described in Section IX.

An operational mockup will be used as a test bed to determine subsystems interface compatibility of spare components, both new and recycled, prior to installation in the operational article. Tests can be conducted to establish trend data parameters and to analyze those already established.

METHODS OF ESTABLISHING RELIABILITY

Of significant impact on the effective reuse of components is the flight environment in which the equipment is operated and the levels to which it is qualified. Reuse requires that sufficient analysis and testing be conducted to accurately predict the expected operational environment. Once the operational levels are established, adequate margins will be applied for qualification testing.

Using the vehicle flight profile from pre-launch through recovery, the dynamic and thermal environments encountered by the vehicle have been identified. One set of components and the complete vehicle will be subjected to the environmental requirements specified for qualification. If the basic reliability, provided by the methods of specifying environmental requirements, is to be maintained, reuse requires that the design be qualified for either an increased test duration or increased test level. One method is to extend the test

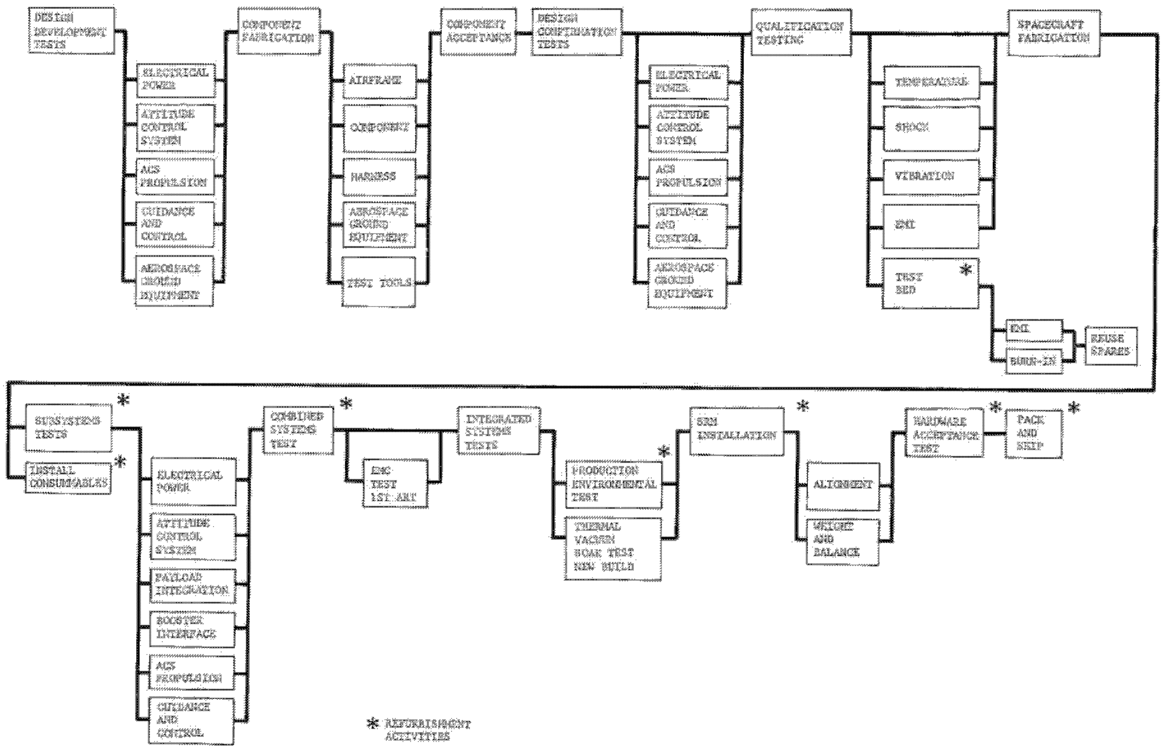

FIGURE VI-1 DEVELOPMENT & RECYCLE TEST FLOW

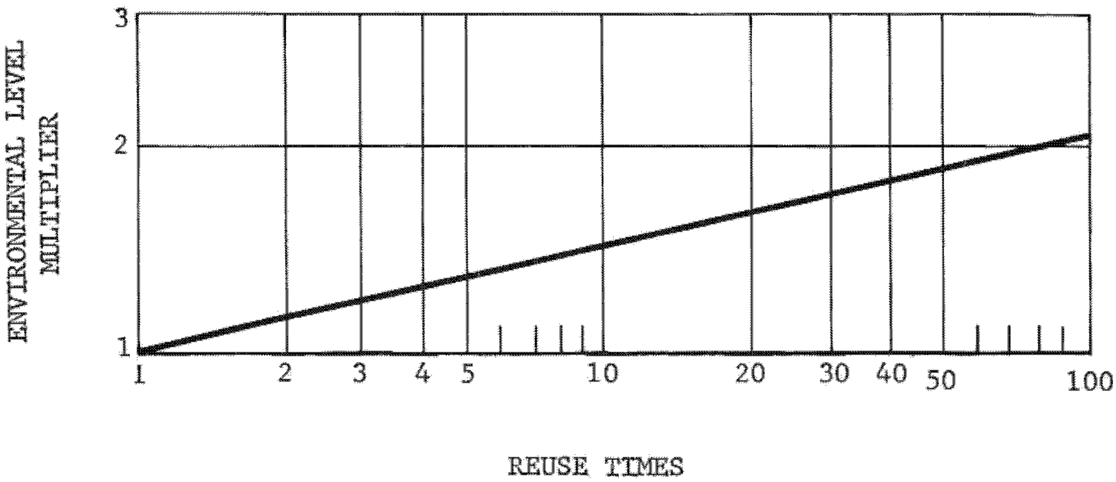

FIGURE VI-2 ENVIRONMENTAL LEVEL MULTIPLIER VS. REUSE TIMES duration to equal the total flight duration, including all reuses. A criterion for determining the increase in test levels to account for longer duration is shown in Figure VI-2, which has been normalized to the original requirement. The curve indicates that the requirement of ten (10) flights would be reflected in a 42% higher shock and vibration requirement than if the vehicle were required to perform a single flight.

Environmental acceptance tests will be performed on all electronic components to weed out in-process manufacturing defects not readily detectable by inspection methods. The tests consist of temperature cycling at the predicted flight temperature levels and random vibration at a level less than the specified flight levels.

Reliability controls and programs will be imposed to maintain the defined level of reliability to include:

- Hi-rel Piece Parts Control
- Limited Shelf Life Use Time or Cycle Controls
- Flight Critical Component Program
- Vendor Control Approval and Surveillance
- Failure Mode and Effects Analysis
- Corrective Action Program

Trend evidence established on a test-to-test basis beginning at development test and continuing through all tests, flight, and post-flight combined systems test on a chronological ship-to-ship basis will establish the predicted limit of constant failure rates. Based upon PRIME trend experience, limited flight environmental instrumentation on a minimum of two operational vehicles provides sufficient trend data from flight at minimum cost.

Performance trend data from recycle tests correlated with the predicted life, as determined in qualification tests, will be used to determine a components retirement from use.

The controls and programs imposed are the same as those successfully developed and used on Gemini and Titan III. The trend analysis and qualification test levels and duration for reuse are the same as those developed on the PRIME Program. The retirement from use based on predicted life and correlated trend data is similar to techniques used by commercial airlines and military aircraft.

BASIS OF COST ESTIMATE

Refurbish costs are estimated at ten percent of the initial cost of the reused hardware. This number compares favorably with the results of other studies made on refurbishing similar spacecraft,* particularly if the advantage of design for reuse is factored into the comparison. PRIME experience showed that the 21 hardware items satisfactory for reuse included the major cost items. The eight items requiring refurbishment included electrical disconnects, environmental control valves, cold plates, ACS thrusters, and temperature and pressure sensors. Replaced hardware, such as SRM's (kick and deorbit), battery, recovery package and expended ordnance is priced separately for each flight.

Based upon a launch rate of 12 per year, five FASTBACK spacecraft are required. Four of them are required to meet the high launch rate (one per day for four consecutive days) and an additional vehicle is required for the contingency of a loss of a vehicle during the three year launch period.

Our reliability analysis indicates an overall system reliability value of 90%. Half of the failures were assumed to result in a complete loss of the spacecraft and the remaining 5% resulted from a spacecraft subsystem failure, allowing the spacecraft to be recovered. A launch booster failure would be an example that would result in a complete loss of the spacecraft, and the camera is an example of a subsystem failure that would allow the spacecraft to be recovered.

One vehicle is assumed to be sufficient from a reliability viewpoint since the probability of requiring the maximum launch rate early in the program in combination with a complete spacecraft failure is low.

* "Recovery, Refurbishment and Reuse of Spacecraft", SAMSO TR-67-22 (North American Report SD67-959)

RELATED EXPERIENCE

Though not exactly comparable, there is considerable related experience to draw upon for confirmation of the feasibility of the FASTBACK concept.

The X-15 Program contains parallels to FASTBACK in that repeated flights have been made of vehicles recovered from near space (having flown at altitudes of 67 miles at 4000 mph). By 1964, 120 flights had been made using three vehicles, one of which was damaged for part of that period. Refurbish and reuse was accomplished on the heavily instrumented, man-rated X-15, which is considerably more complex than the FASTBACK spacecraft. These operations were completed in time spans comparable to those projected for this program and with hardware and techniques that have since advanced significantly in reliability and automation.

The PRIME Program provided a thorough examination and evaluation of the condition of equipment comparable to that employed in FASTBACK after return from sub-orbital flight. Aside from those components expected to be non-operative as a result of reentry, the major subsystems (electrical, guidance and control, structure, instrumentation, ACS) were found to be flight-ready.

Commercial and military aircraft, the latter employing camera systems similar to those of FASTBACK, fly many missions comparable in length to this program, separated by only routine maintenance. The severity of the environments experienced by FASTBACK (which are predictable, measurable, and can be designed for) is more than offset by the fact that aircraft must be safe for manned flight.

Gemini and Apollo recovered vehicles, though not recycled for manned flight, were capable of reuse with minimal maintenance. The heat shield qualification program for the MOL Program used a recycled Gemini vehicle successfully.

CONCLUSION

The refurbish-reuse concept is the logical application of existing technology and experience to a program designed to satisfy national requirements in a manner compatible with the current economic climate. It is a flexible approach which allows repeated reassessment of risk and confidence factors during the program, with alternate options available, depending upon conditions prevailing at the time.

VII. BOOSTER DESCRIPTION

A. CONFIGURATION

The booster selected for the FASTBACK mission is the surplus Minuteman I LGM30B vehicle. The use of this vehicle with an orbit injection motor provides at least a 1000 pound payload capability into all orbit inclinations desired for the FASTBACK mission.

The LGM30B vehicle propulsion and control systems are to be used as is, but the guidance wafer and guidance system will not be used because of weight, power and gimbal limitations. The boost phase guidance and autopilot functions will be provided by the payload guidance and computer system. A range safety tracking and receiver system will be installed in Stage III to provide tracking and thrust termination capability during the first three stages of flight.

The Minuteman I vehicle is 45.2 feet long up to the payload interface. The Stage I motor provides 12.4×10^6 lb-sec of impulse through four gimballed nozzles. Stage I has a diameter of 5.5 feet and is 24.9 feet in length. The Stage II motor has four gimballed nozzles and delivers 2.81×10^6 lb-sec of impulse. This motor is 3.69 feet in diameter. The stage III motor has four gimballed nozzles that can be thrust terminated. Rated impulse is 1.02×10^6 lb-sec and the motor is 37.6 inches in diameter.

B. PAYLOAD CAPABILITY

The payload capability of the LGM30B vehicle with the orbit injection motor is shown in Figure VII-1 for FASTBACK orbit inclinations with launch southeast from Johnston Island. A payload of 1000 pounds can be injected into the desired orbits with inclinations between 17 and 70 degrees with 60 pounds of propellant margin reserved in the Stage III motor.

The nominal FASTBACK orbit with perigee of 65 nm. over the target and apogee of 300 nm. was selected to maximize photo resolution while minimizing guidance injection dispersions, orbit lifetime limitations and recovery area dispersions.

A coast phase of approximately 180 seconds between Stage III burnout and injection motor ignition was found to greatly increase payload capability over that of a four-consecutive-burn sequence. Dynamic pressure (q), dynamic pressure times angle of attach ($q\alpha$) and aerodynamic heating indicator (AHI = $\int qVdt$) values of 4400 lb/ft^2, 17,000 $\frac{\text{lb deg.}}{\text{ft}^2}$ and 706 $\times 10^6$ lb/ft, respectively, were used as constraints for trajectory design.

Injection motor burnout will occur 930 nm. from launch at a latitude of 3.9 degrees North for a 55 degree inclined mission orbit.

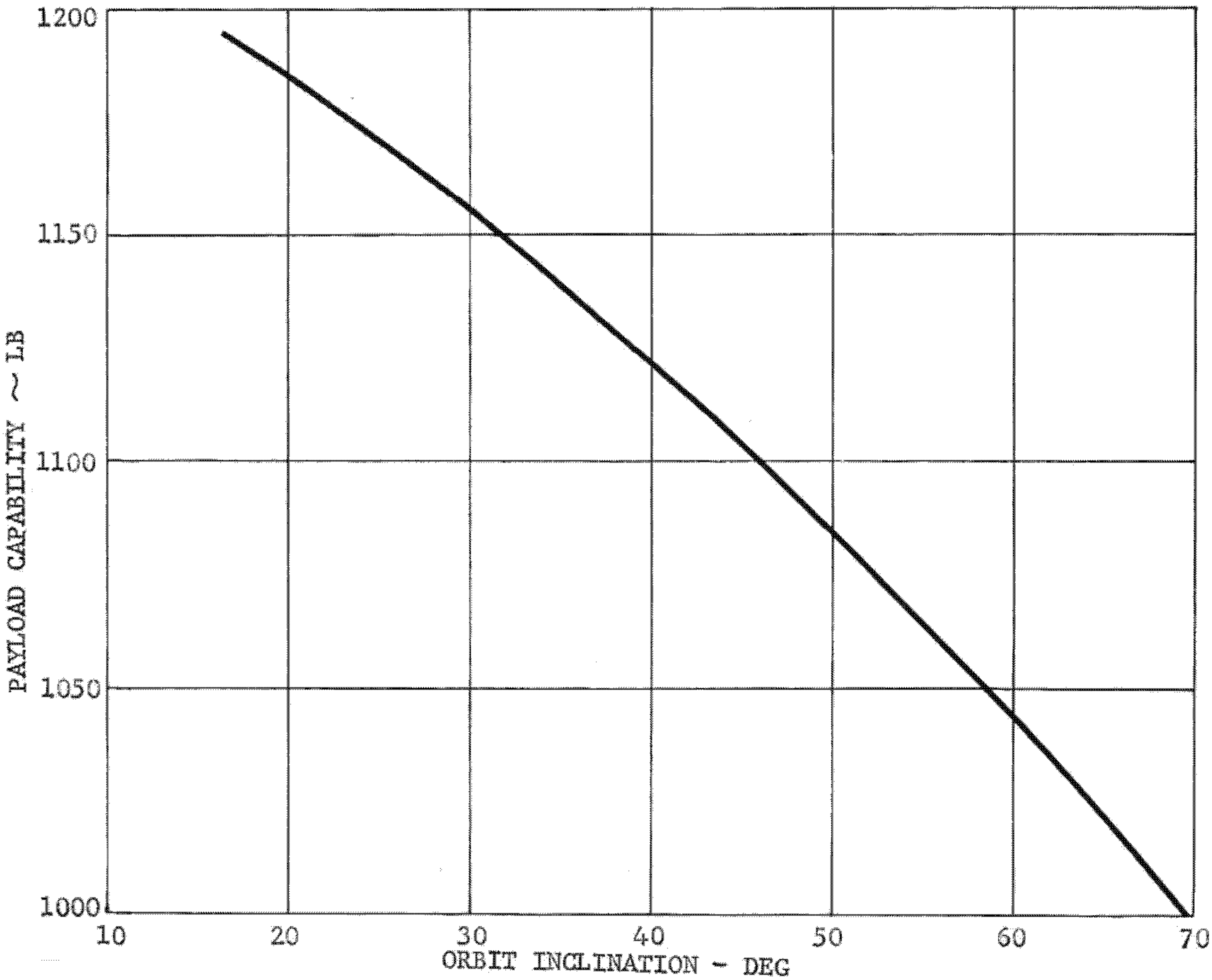

FIGURE VII-1 PAYLOAD CAPABILITY

C. RANGE SAFETY

An analysis was performed to determine the extent of the range safety problem at Johnston Island. A probability study for the proposed flight plans was performed and casualty expectancy (kill probabilities) determined. This analysis was based upon the same requirements that are used for WTR and ETR as described in "Range Safety Manual", AFETRM 127-1. The data shows that a Minuteman I launch from Johnston Island does not present a range safety problem since the kill probability for FASTBACK missions is considerably less than those of other programs.

VIII. AGE

A. ELECTRICAL AGE

The electrical AGE will be mounted in a van with provisions for peripheral equipment storage. The core of the AGE will be a small computer, its purpose being to automatically control vehicle functions and responses in a time ordered fashion. A completely automated subsystems and combined systems test approach is proposed in order to comply with the program's quick reaction requirements.

Peripherals will include interconnecting cables between the van and the test article and necessary hand carried test equipment.

The AGE will perform the following functions:

1. Generate stimuli to the vehicle to simulate in-flight conditions.
2. Receive, condition, and evaluate responses from the vehicle systems and format this data on magnetic tape for trend analysis.
3. Generate the program for the airborne computer, load and verify that program in the computer and update as required.
4. Provide prelaunch vehicle verification.
5. Provide complete launch control for the vehicle including safing and shutdown capability.

Three sets of AGE will be required. One set will be located at Denver for checkout and acceptance of the vehicle after refurbishment is complete. Two sets will be located at the launch site, one set to perform marriage tests and compatibility verification between the space vehicle and the booster and the other set to perform launch activities. Each launch site set will have a switch rack to enable use of one set for two checkout lines and the other set for two launch pads.

The three sets will be designed to be identical to minimize costs in software and to provide trend data for each vehicle.

Electrical AGE Software - The software package that is proposed for this program will support the following tasks:

1. Guidance and control system tests including airborne computer loading and verification operations.
2. Vehicle verification and marriage tests.
3. Vehicle acceptance tests.
4. Mission integration and control.
5. Launch operations including shutdown and safing functions.

The baseline software package will be generated by a general purpose computer such as the IBM 360-30, and verified by a land-line interface between the FASTBACK system computer and the general purpose computer.

Modifications to the software will be accomplished with the computer interfacing with a teletypewriter.

B. MECHANICAL AGE

Mechanical AGE is defined as all mechanical and structural equipment required to handle, transport, ship, assemble, service, checkout, erect and launch the booster plus satellite, starting with the satellite final assembly and carrying through refurbish and recycle operations, but not including airplane peculiar recovery operations. Equipment required is based on the following ground rules: The satellite will be delivered to the launch site, checked out and ready to launch except for ACS and ordnance servicing. Minuteman stages will be delivered to the launch site in the Minuteman Transporter Erector or the Minuteman Shipping and Storage Container, checked out and ready to launch except for ACS, hydraulic and ordnance servicing. Mechanical AGE required for Minuteman field servicing will be furnished G.F.E.

The structural type equipment is that required to handle the spacecraft and its components starting at final assembly and carrying through the various in-plant test operations, the launch site assembly and the return and refurbish cycle. It consists of a handling dolly and slings, test adapters, payload and equipment handling tools and a shipping container.

Mechanical equipment is that required to service the spacecraft in-plant and at the launch site. It consists of an ordnance carrying case and installation kit and an ACS propellant servicing unit.

Launch site equipment is that required to assemble and service the spacecraft/booster, to checkout the combination, transport it to launch pad, erect it and launch. It consists of a vehicle alignment kit, a launch mount and a transporter erector. The transporter erector will incorporate necessary launch umbilical provisions to preserve intact necessary launch umbilical connections.

IX. LAUNCH COMPLEX AND LAUNCH OPERATIONS

A. FACILITIES AT JOHNSTON ISLAND

From a study of the present layout and building complement of Johnston Island, it appears that sufficient facilities exist now to handle the FASTBACK program with no additional construction. (Ref. Holmes and Narver drawings 91-002-C33.21 and C 32.20, dated September 1969, and related "as-built" drawings) (Figure IX-1)

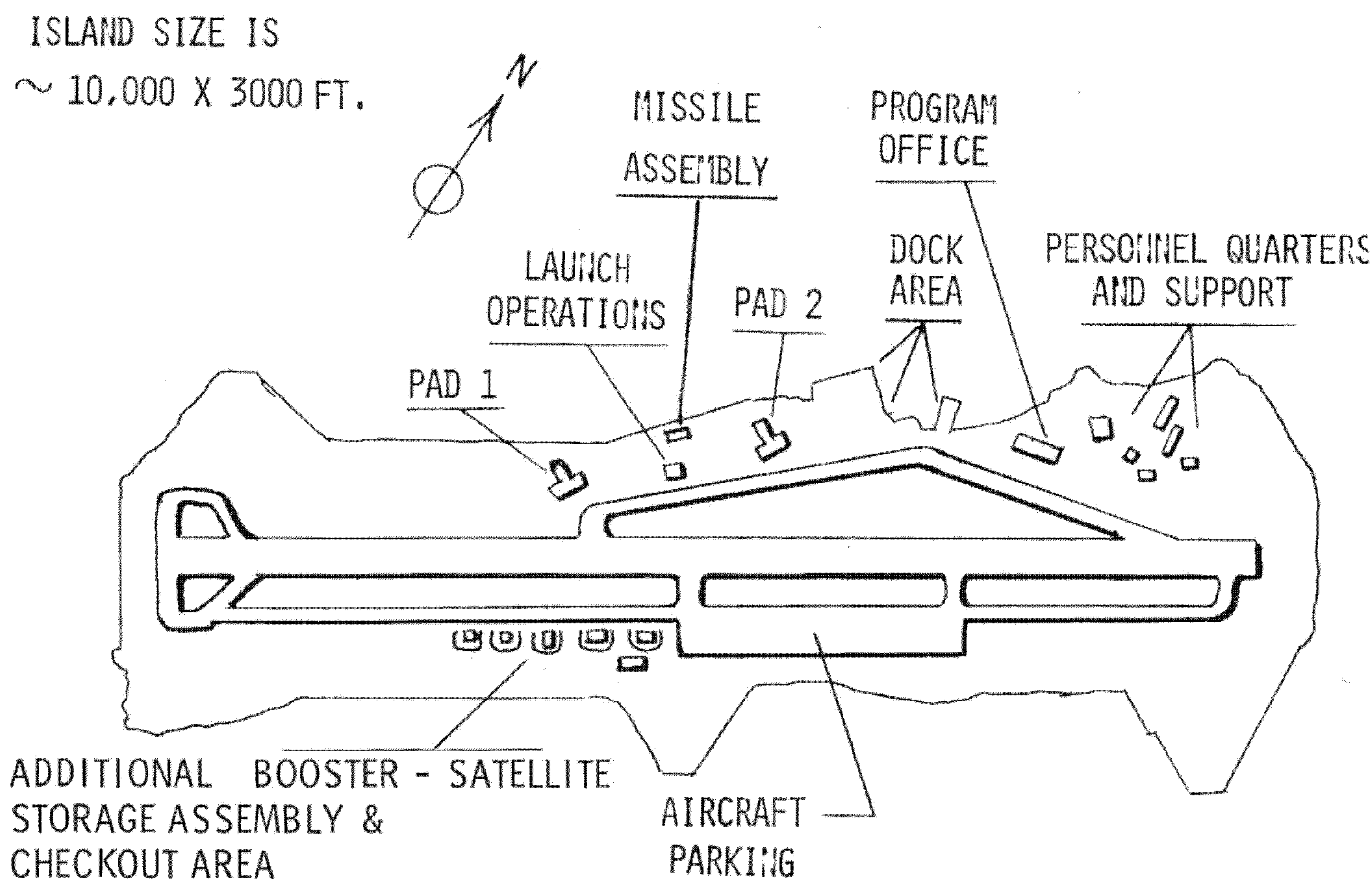

FIGURE IX-1 JOHNSTON ISLAND LAUNCH COMPLEX

It is planned that the FASTBACK program would take over the existing 437 program launch complexes, launch operations building, missile assembly building, and some other peripheral support facilities. Modifications will be needed in the MAB and LOB.

More precise definition of facility needs are identified below and for clarification are identified as (1) Operational Facilities and (2) Base Support.

1. Operational Facilities - These resources are needed:
   - The airstrip with its accompanying control tower and the wharf with its capability to unload from sea-going barges or small ships.
   - Buildings meeting quantity - distance criteria and with provisions for a) environmental controls, b) electrical explosion proofing, c) overhead crane operation, and d) security protection to:

     Receive, store and checkout a maximum of six boosters.

     Receive, store and checkout a maximum of five satellites.

     Assemble, checkout and ready the vehicle for pad installation.
   - Bunkers for ordnance storage and checkout.
   - An area for hydrazine propellant storage.
   - The launch complex consisting of two launch pads and the launch operations building to accept, erect, checkout and launch the vehicle.
   - The range safety network to provide flight termination capability for erratic flight, unknown position or violation of established impact limits.
   - A shop dedicated to the maintenance and calibration of FASTBACK oriented hardware.
2. Base Support - These facilities are required:
   - An administrative building, living quarters and

messing facilities for about 85 personnel.

- Warehouses, vehicle shops, maintenance shops, laundry facilities and other related support functions.
- The range of utilities necessary to support operational and living requirements.
- Weather and communication support.

B. FLOW OF SATELLITE AND BOOSTER

The launch crew on Johnston Island will have the capability of handling all operational activities in support of the receipt to launch schedule.

Upon receipt of the booster (or the satellite) at Johnston Island it will be moved to a storage area where a receiving inspection will be performed. As-received tests will be made to verify that no in transit problems were incurred.

A booster and a satellite will be moved to the assembly and checkout building where they will be mated and marriage compatibility tests performed. At this time the computer will be base line programmed, ordnance mechanically installed and safe arm plugs installed. The vehicle will then be transported to and installed in the erector shelter on the launch pad.

Prelaunch operations consisting of battery trickle charging, ordnance verification and launch enable checks will be made. Upon command the vehicle will be erected, ACS propellant loaded, a final program read into the computer, the platform aligned, ordnance armed and the vehicle launched.

Post launch activities consist of refurbishing the launch pad. A readied standby vehicle will be available in the Assembly and Checkout building for each launch pad. Immediately following a launch the capability will exist to move a vehicle out to the pad and ready it for flight such that four launches can be accomplished in four days.

LAUNCH OPERATIONS FLOW

BOOSTER
- Receipt
  - Storage
    - Receiving Inspection
      - Test

SPACECRAFT
- Receipt
  - Storage
    - Receiving Inspection
      - Health Checks

SPACECRAFT/BOOSTER VEHICLE
- Assembly
  - Ordnance Installation
    - Movement to Launch Pad Shelter
      - Prelaunch Operations
        - Erection
          - Propellant Loading
            - Computer Final Program
              - Platform Alignment
                - Ordnance Armed
                  - Launch
                    - Refurbish Pad

It is obvious that we have assumed that Program 437 will be discontinued, but that the island will continue to be operated, at least in a standby status. If this assumption is not supported by pending Government decisions, alternative courses of action are possible.

1. <u>If 437 is Placed On Standby</u> - FASTBACK could use the 437 facilities with provision made in the necessary modifications, so that joint use could be possible.

2. <u>All The Present Activities Will Be Removed From The Island</u> - The FASTBACK programs would operate the island, maintaining only those program facilities and base support functions necessary for FASTBACK. This would mean opening the airfield and port for just FASTBACK traffic and with little or no general island maintenance. By proper personnel rotational policies and employing technicians with a range of skills, numbers can still be kept in the region of 100 people and thus the cost of operating on Johnston Island still kept down.

X. SEQUENCE OF EVENTS

The following sequence describes the major events from liftoff at the launch pad of Johnston Island to the return of that spacecraft to Johnston Island for reuse.

A. LIFTOFF TO ORBITAL INJECTION

After the booster has been erected and the spacecraft computer reprogrammed for a specific mission, the launch is initiated at that time which will give optimum lighting over the target area(s) of interest. The vehicle is launched along a southeast azimuth to give an orbit inclination that will provide the optimum target coverage.

After the burn and separation of the MMI third stage, a coast phase of approximately 180 sec. is begun. This coast phase allows the vehicle to attain the desired orbit altitude while using the earth's gravity to turn the velocity vector until it is parallel to the desired orbit. The orbit injection motor is then fired to attain an orbit with nearly sufficient energy to provide the nominal mission orbit (65 nm. perigee and 300 nm. apogee) with perigee location over the target area. The orbit injection motor is then jettisoned and a final velocity increment is added by the ACS thruster to adjust the orbit. This delta velocity also aids the separation of the spent injection motor from the spacecraft.

The use of the 180 sec. coast phase greatly increases the payload capability of the launch system over that which can be attained with four consecutive burns.

B. ORBITAL INJECTION TO DEORBIT

After the spacecraft achieves its operating orbit, the data collection phase begins. The viewing port is opened, the spacecraft goes into a belly down, forward-pointing attitude, and the optical sensor is sequenced for exposures over the target area. During this period, the attitude control system pitches the spacecraft at a rate which causes the spacecraft to maintain a belly down attitude. All signals for spacecraft operation are controlled by the guidance computer.

Since the deorbit sequence of events is preprogrammed, injection errors can result in appreciable impact dispersions. Therefore, the C-Band beacon is used during the first half of the first ascending orbit pass to permit orbit determination with FPS-16 trackers on Ascension Island.

This orbit determination then permits the recovery aircraft to be deployed to the initial predicted recovery point. The beacon is used a second time during the last ascending pass to permit final recovery point prediction by tracking from Antigua.

Prior to the deorbit motor firing, the spacecraft is reoriented from its forward pointing attitude to the attitude required for the deorbit motor thrusting. This action is controlled by a programmed command to the attitude control system. Thrusters provide the force required and gyros sense the attitude changes.

C. DEORBIT TO RECOVERY

After the deorbit motor has been fired, the spacecraft is reoriented to a forward-pointing attitude and the deorbit motor is jettisoned. Spin-up rockets then impart 5 to 10 revolutions per minute to the spacecraft about the longitudinal axis. After the maximum reentry deceleration has been sensed, a recovery package sequence will be initiated. Approximately 15 minutes after deorbit motor firing the beacon will be turned on to aid in the location of the spacecraft as it descends. The recovery package begins its operation by separating a thermal shield (cover) and deploying a drogue ballute for braking. The drogue ballute is released at 100,000 feet and it deploys the main chute in a reefed configuration. After further deceleration, this chute is unreefed (50,000 feet) and the spacecraft descends at its slowest sink rate. The main chute employs a large protruding bag which is located at the top of the main chute. As the parachute descends, this protruding part is engaged by the boom and cable recovery apparatus on the retrieval aircraft. After intercept of the spacecraft, the apparatus is retracted and the spacecraft hoisted aboard.

D. RECOVERY TO REUSE

After the spacecraft has been recovered, the recovery aircraft is then taken to a light-tight facility for disassembling the spacecraft and unloading the film. The film is then delivered to a photo processing agency and the spacecraft is placed in a container for air shipment to the Denver facility for refurbishment. At Denver, circuits are tested, camera action is checked, ablative surfaces replaced and the film is reloaded. Upon completion of the refurbishment and system checkout, the spacecraft is packed and shipped to Johnston Island. At Johnston, the spacecraft is unpacked, inspected for damage from shipment and then mated with a tested Minuteman booster. The combined vehicle is then tested and placed in ready storage at the assembly building or launch pad. These vehicles are ready for launch except for reprogramming of the computer, erection of the booster and various arming functions.

XI. SYSTEM ERROR ANALYSIS

A. ORBIT DISPERSIONS

The 3-sigma orbit injection errors associated with the proposed Hamilton Standard guidance system are shown in Table XI-1. The significant effects of these injection errors on the orbit elements are to change the orbit period by eight seconds and the perigee/apogee altitudes by 2.2 and 4.6 nm. respectively. The resulting errors in orbit inclination and vehicle flight path angle are less than 0.03°.

TABLE XI-1 3-SIGMA ORBIT INJECTION ERRORS

Tangential Velocity Error	± 12.0 fps
Radial Velocity Error	± 9.2 fps
Normal Velocity Error	± 11.9 fps
Tangential Position Error	± 1524 ft
Radial Position Error	± 2 ft
Normal Position Error	± 1781 ft

Without tracking, the accuracy with which the vehicle's position in orbit can be predicted is principally a function of injection accuracy, atmospheric density prediction accuracy and time in orbit. Figure XI-1 shows the expected 3-sigma downrange error resulting from the injection errors from Table XI-1 and a 16% error in atmospheric density, as a function of mission lifetime. An error in injection velocity contributes to error in predicted vehicle position by both a constant rate due to initial period difference and a rate that increases with time which results from perigee altitude error. Because of the expected error in predicting position in orbit, the optical system must be operating before and after the nominal overflight period to ensure complete coverage. Figure XI-1 shows that the downrange error increased approximately ± 50 nm. per revolution which is equal to 1% of the total coverage capability of 10,000 nm. Therefore, for example, on the seventh revolution, 7% of the film on board must be used to compensate for the error in orbit position.

B. IMPACT DISPERSIONS

The uncertainty in the reentry impact location is greater prior to launch because of the dominant effect of expected injection errors. Impact uncertainty can be significantly reduced by tracking the vehicle. The feasibility of first revolution tracking has been established via trajectory simulation. These data show that vehicles launched from Johnston Island with orbit inclinations between 17° and 75° can be acquired during the first revolution by the Ascension Island Tracking Station (Ref. Figure XI-2). A ten minute station pass utilizing a C-band tracker is expected to reduce orbit period uncertainty to less than 5.4 seconds (3-sigma) and reduce perigee altitude uncertainty to less than 400 feet (3-sigma). This permits the downrange impact uncertainty to be reduced as shown in Table XI-2.

TABLE XI-2 IMPACT DISPERSION (3-SIGMA)

Reentry Revolution	Downrange Uncertainty - Nautical Miles		
	No Tracking	First Rev. Tracking	Final Rev. Tracking
3	± 140	± 100	± 62
4	± 186	± 117	± 62
5	± 239	± 143	± 62
6	± 296	± 166	± 62
7	± 358	± 198	± 62
8	± 424	± 223	± 62

This table also shows the impact dispersion if a tracking station pass is obtained one revolution before reentry. Figure XI-2 shows that when the Antigua Tracking Station is utilized, tracking just prior to reentry is feasible for most missions originating from Johnston Island. The minimum impact dispersion value of ± 62 nm. results primarily from the deorbit maneuver

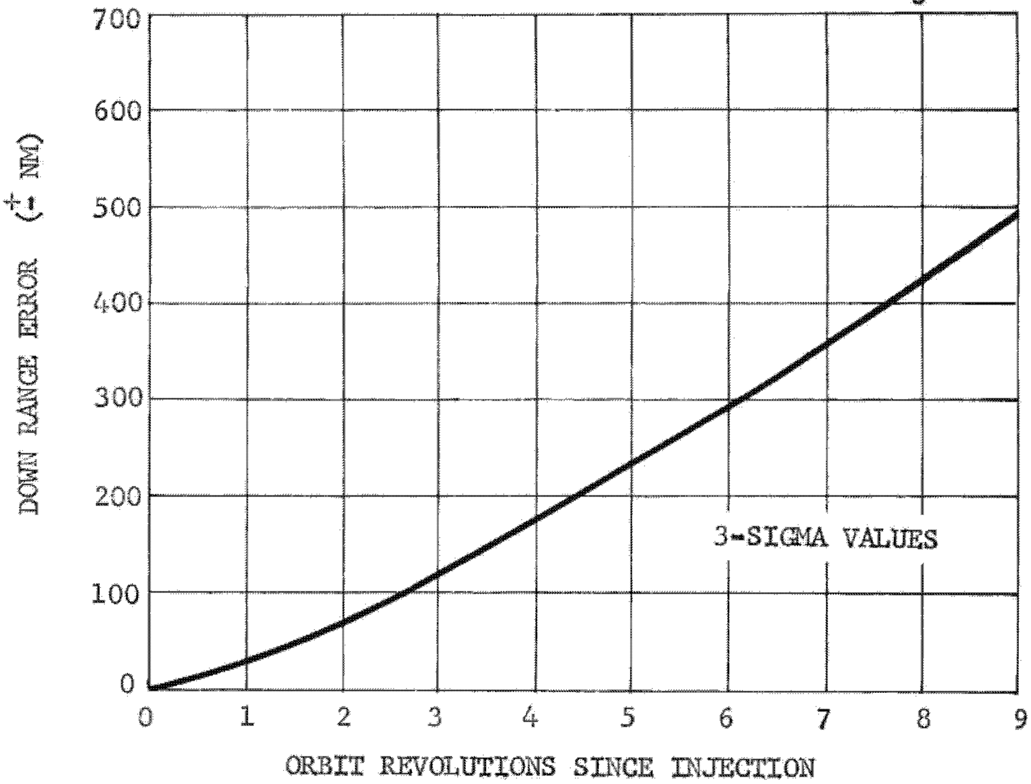

FIGURE XI-1 ORBIT ERRORS (NO TRACKING)

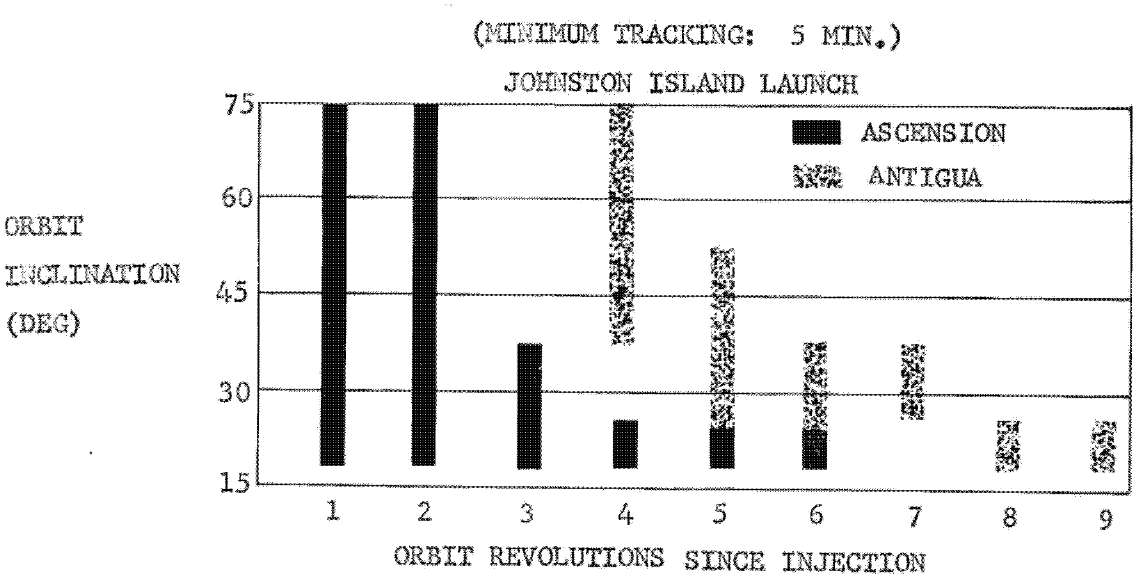

FIGURE XI-2 TRACKING STATION COVERAGE

errors as discussed in the following paragraph. Reducing the orbit prediction error to zero will only reduce the impact dispersion from ± 62 nm. to ± 60 nm.

Impact dispersion originating in the flight phase from retro thrust maneuver to impact results principally from errors in retro velocity magnitude, alignment and timing. Atmospheric variations and ballistic parameter errors are secondary sources of impact error. The reentry vehicle has a one degree angle of attack uncertainty due to c.g. offset which would result in an impact uncertainty of over 200 nm. if the spacecraft is not spun during reentry. A seven revolution/minute spin rate is thus used to reduce this dispersion. This rate is not large enough to require a despin maneuver before parachute deployment. Table XI-3 presents the 3-sigma error for each parameter associated with the reentry flight phase and shows an RSS combined value of 62 nm. In addition, this table presents the sensitivity of range from retro-thrust to impact for the significant parameters, assuming a retro-velocity of 300 feet per second is applied at apogee. By increasing the size of the retro motor, the 62 nm. dispersion could be reduced.

TABLE XI-3 DOWNRANGE IMPACT DISPERSION
(TRACKING ONE REVOLUTION BEFORE REENTRY)

Parameter	3-Sigma Error	Range Sensitivity	Range Error
Retro Magnitude	± 5 fps	10 nm./fps	± 50 nm.
Thrust Alignment	± 2 deg	13 nm./deg	± 26
Retro Altitude	± .02 nm.	48 nm./nm.	± 1
Retro Timing	± 5.4 sec	4.2 nm./sec	± 23
Ballistic Parameter	± 40 lb/ft^2	0.2 nm./(lb/ft^2)	± 8
Atmos. Density	± 8%		± 2
Winds	(250 knots)		± 3
Angle of Attack	(7 rpm roll rate)		± 2
		RSS TOTAL	± 62 nm.

It is evident from Table XI-2 that when tracking occurs one revolution before reentry the principle source of impact dispersion is deorbit velocity error. In order to further reduce the expected impact dispersion by a significant amount it would be necessary to reduce either the error magnitude or the range sensitivity to velocity error. The 5 fps error used in the analysis was obtained by combining the following components: 1) impulse error, 3.0 fps; 2) vehicle weight error due to non-nominal use of attitude control gas, 3.3 fps; and 3) tracking velocity error, 2.2 fps. Any significant reduction in velocity error would require each of the contributing sources to be reduced since they are approximately equal. However, increasing retro-velocity magnitude to 900fps reduces the range error sensitivity to 5 nm./fps and reduces the total range error to $\pm$ 44 nm. provided the current error budget can be maintained.

XII. SYSTEM ALTERNATIVES

The system design approach that was used to satisfy the major program objectives of FASTBACK is sufficiently flexible to allow for alternative approaches in several areas. Some of the more pertinent alternatives that are available are presented in this section of the report.

A. ALTERNATE LAUNCH SITES

If, for any reason, Johnston Island could not be made available for the FASTBACK program, the use of AFETR and AFWTR was investigated.

From ETR launch azimuths of 38° to 90° (inclinations of 28° to 55°) can be used. This results in coverage as shown in Figure XII-1. The higher inclinations would require some widening of the present launch restrictions imposed by range safety authorities.

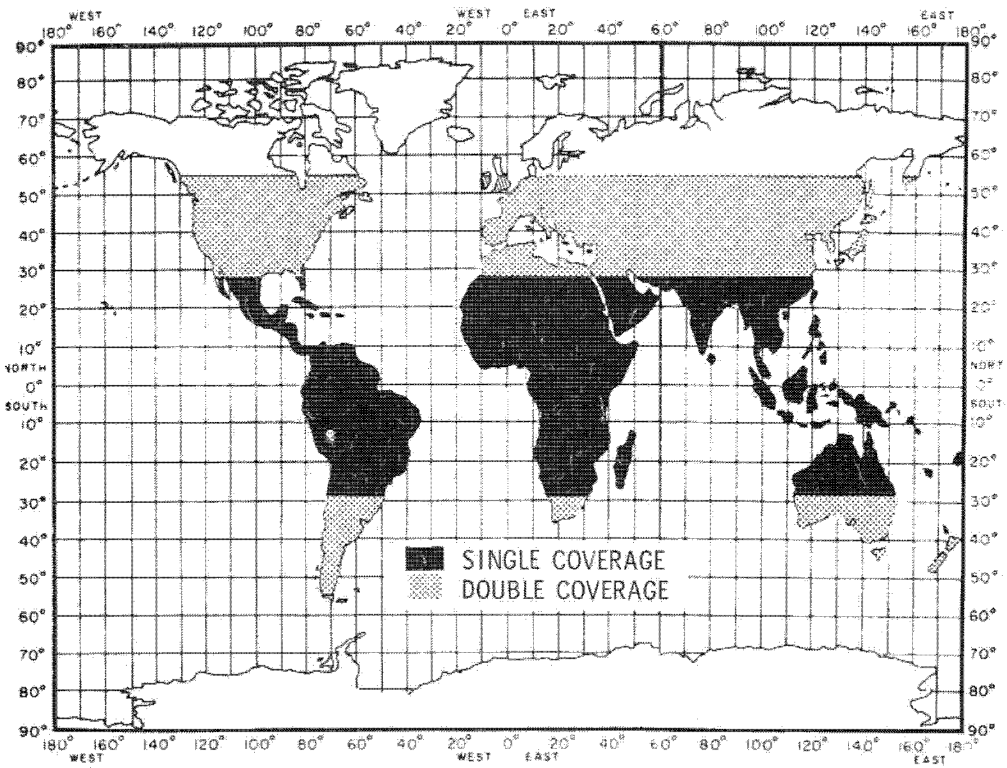

FIGURE XII-1 COVERAGE FROM ETR (LAUNCH AZIMUTH 38° - 90°)

From AFWTR launch azimuths of 149° to 180° (inclinations above 65°) can be used. This results in coverage as shown in Figure XII-2. Again, some widening of normal launch azimuth restrictions would be required but not beyond that which have been granted in the past for other programs.

The combined coverage from both sites is shown in Figure XII-3. The loss of ability to launch into the 55° to 65° inclination results in the inability to have double coverage in what might become a very important target area.

Launch from AFETR would not allow the meeting of a 24 hour response time. On-orbit time must be 24 hours and recovery and processing time would necessarily have to be added. So response time could extend to nearly 30 hours.

The nearly doubled orbit life will result in a significant increase in accumulated system errors. Because the camera operation on a mission launched from AFETR will take place generally in the later revolutions, these errors will cause greater cross track and along track errors as well as recovery uncertainties. These errors may force an addition to the system to update the operating program in the command and guidance package. Although the addition of a command receiver and some electronics does complicate the spacecraft, it is not viewed as a major problem either in engineering or cost. The increased mission life time would also have an impact on spacecraft heating, electrical power and attitude control propellant which would have to be investigated.

The support facilities and personnel available at Kennedy and Vandenberg and the above ground launch equipment in storage at Kennedy would make activation of these two sites a relatively minor task.

B. <u>ALTERNATIVE CAMERA APPROACH</u>

The illustrative camera that has been designed into FASTBACK and described earlier in this report was selected as a primary approach on the premise that stereo with 3.5 foot resolution was

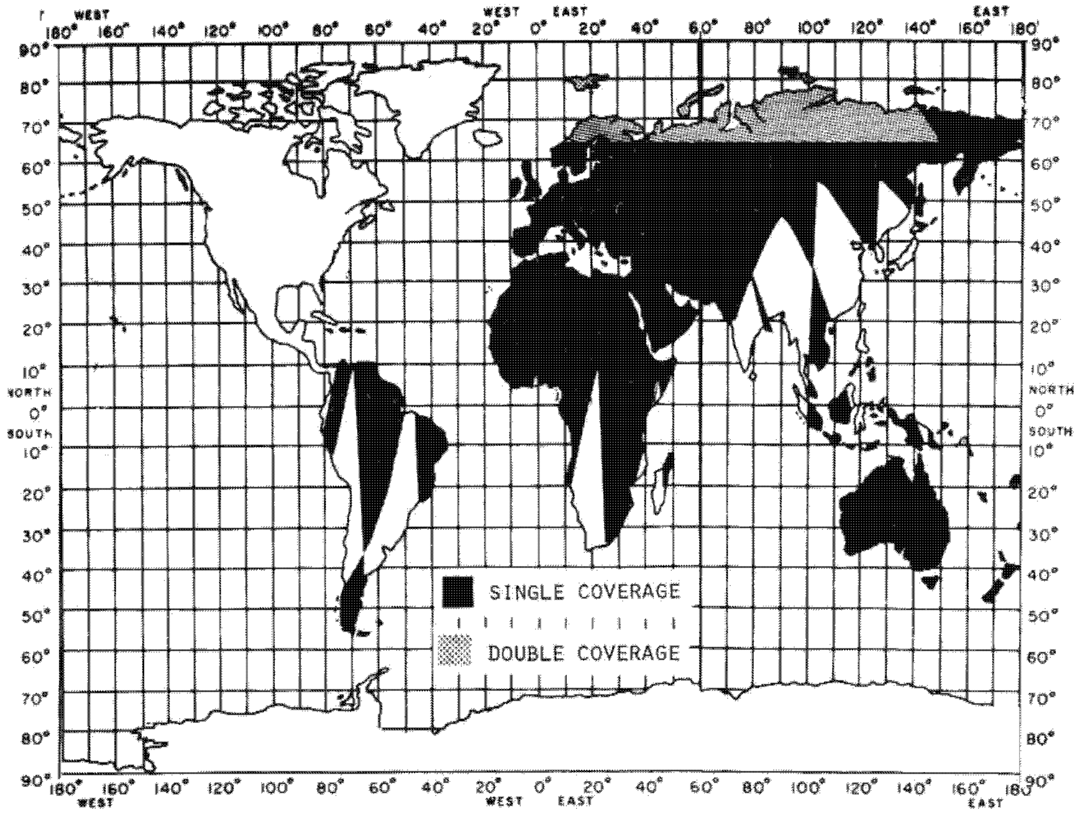

FIGURE XII-2 COVERAGE FROM WTR (LAUNCH AZIMUTH $149^0 - 180^0$)

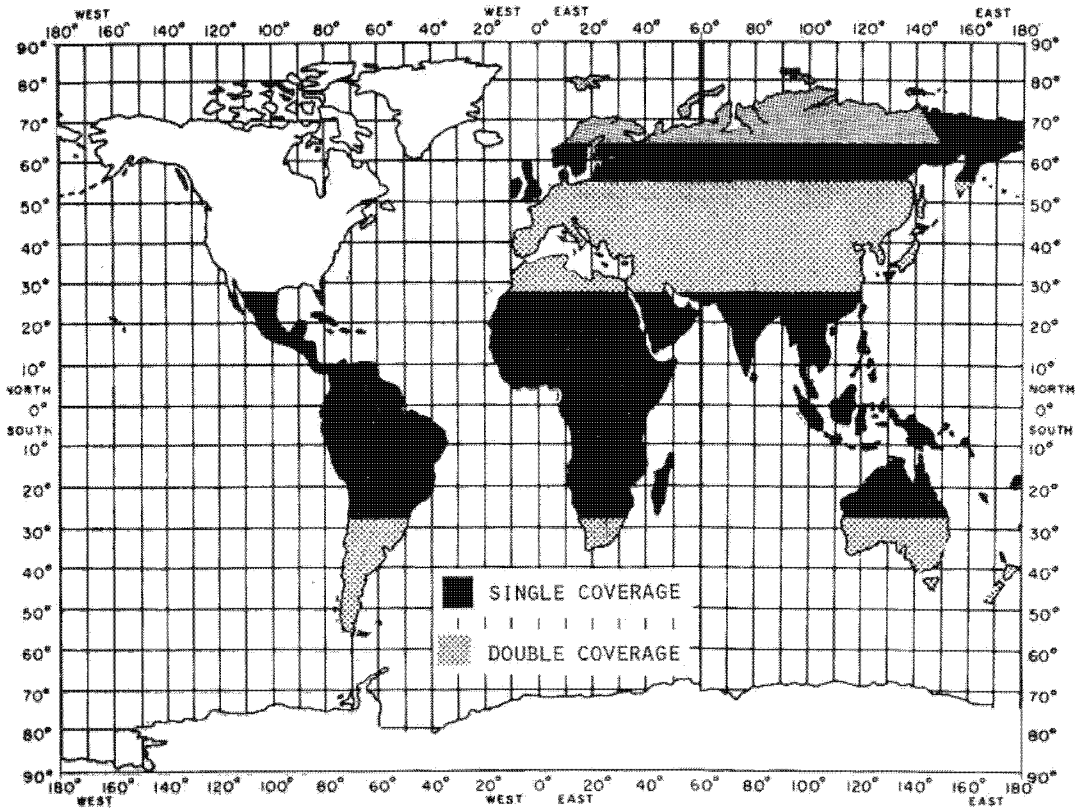

FIGURE XII-3 COMBINED COVERAGE FROM WTR AND ETR preferred over mono with a better ground resolution. We have discussed with the camera manufacturers the possibility of a monographic system. For an increase in weight of approximately 100 pounds and a slight increase in spacecraft dimensions, a camera with a 36 inch focal length and an f/3 relative aperture to obtain a ground resolution of 2-1/2 feet at nadir can be furnished. By reducing the film load as a result of not having stereo and employing a larger orbit inject motor, this camera could be accommodated.

C. ALTERNATE RECOVERY AREA

If, for any reason, an east coast recovery was disallowed, recovery over Hawaiian waters could be made at the cost of about 15 hours in response time. This added time results from the three additional revolutions to reach Hawaii and the flying time from Hawaii back to Washington, D.C.

It would have the advantage of keeping the fleet of recovery aircraft at the same base but would force the maintenance of courier aircraft in a similar alert status as the recovery aircraft.

XIII. PROGRAM SCHEDULE

The program milestone summary schedule, Figure XIII-1, displays an energetic approach which we feel is achievable for a first launch fifteen months from go-ahead of the development program. Our approach assumes a system using off-the-shelf hardware where possible and maximum utilization of existing technology. We plan three sets of van-mounted Ground Support Equipment, one of which will be used in Denver for test and acceptance of the spacecraft and spacecraft re-furbishments. The other two will be supplied to the Johnston Island launch site. Spacecraft to launch vehicle interface checks will be performed using simulators at the Contractor and OOAMA facilities. An item critical to this schedule is the availability of military air-lift for movement of the spacecraft between Denver, Johnston Island, the recovery site and back to Denver; in addition, the first two Minutemen are scheduled for airlift to Johnston Island. The balance of Minutemen will be stock piled at a port of embarkation for sea transport to the island in lots of from six to nine. The refurbish and launch programs assume an average 12 per year rate over the three year launch program for estimating purposes.

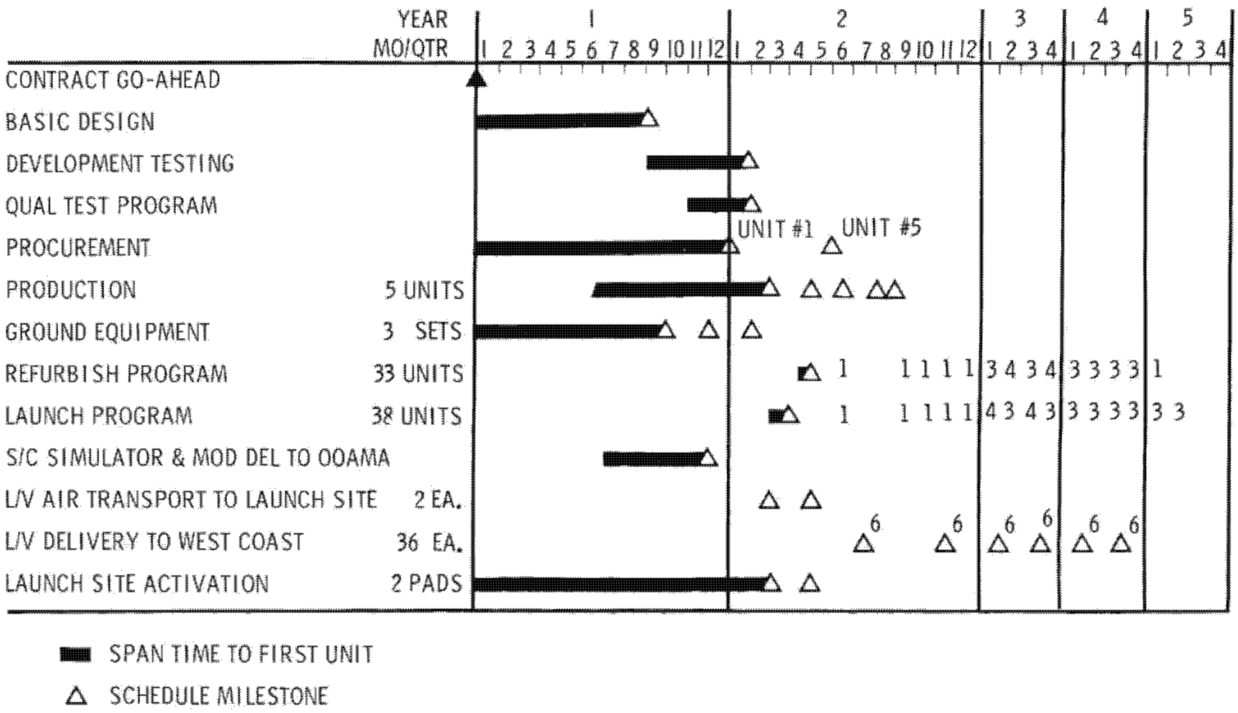

FIGURE XIII-1 PROGRAM SCHEDULE

XIV. PROGRAM COSTS

A. GROUND RULES AND ASSUMPTIONS

1. Minuteman I missiles will be available in desired quantities in early 1972 at an approximate cost to the Customer of $127,000.00 each. These dollars include the acquisition cost of the vehicle, modification of the vehicle, booster spares, and transportation to the launch site.
2. Assume the use of Johnston Island as the operational launch site, and:
   a) The island will be made available to begin activation activities in time to accomplish the first launch objective in mid 1972.
   b) Facilities presently on the island are largely useable and require minimum modification.
   c) Base support requirements for activation and operation, i.e., roads, island port, airfield, and other support normally available at a Government installation, will be supplied at no charge to the program.
3. Assume that recovery aircraft capability is existing for other uses and only specific recovery mission operating costs will be charged to this program.
4. Assume that the film development and distribution will be handled at facilities and by personnel presently in the Washington, D. C., area and would not be charged to this program.
5. Assume that no major modifications to an existing East Coast recovery base will be required.
6. Assume that on-orbit tracking of the satellite. will be furnished by existing tracking stations.

B. BASIS FOR PRICING

1. Price includes thirty-eight (38) operational launches, with the first two having a development instrumentation package installed.

2. Data from previous estimates for similar type spacecraft, payloads, launch facilities and overall operating costs was utilized to arrive at the estimated program cost.
3. Where use of off-the-shelf equipment and components has been identified, vendor estimates have been used.
4. Recurring spacecraft and payload costs are based on a launch rate of twelve (12) per year for a three (3) year period.
5. Five (5) spacecraft and payloads will support the twelve per year launch rate and will provide capability for quick successive launches of one (1) per day for four (4) days.
6. The spacecraft and payload will be reuseable and the refurbish cost will average 10% of the total spacecraft/payload recurring cost, less spares and less nonreuseable items.
7. Hardware required for necessary modifications to the Minuteman missile will be designed and supplied by the Contractor and installed by the Air Force (OOAMA) using Contractor supplied spacecraft simulator for checkout. Contractor supplied missile modification costs are included in the structure subsystem cost.
8. Operational spares are estimated at 25% of the total spacecraft subsystem and payload recurring costs.
9. Parachutes required for the recovery system will be used for a minimum of three (3) launches; therefore, the estimated cost for the recovery system is based on twelve (12) parachutes.
10. Two (2) existing Thor pads and erectors are available and can be modified to accommodate the FASTBACK system.
11. A launch crew of eighty-five (85) Contractor personnel will be permanently stationed at Johnston Island for duration of the program.
12. Existing Minuteman technical data will be used and minimum update will be required.
13. Recovery operations were calculated using $3,000 per hour of airplane operation.

C. COST SUMMARY (IN MILLIONS OF DOLLARS)

NONRECURRING COST				
• Spacecraft and Payload Development	17.04			
• Activation and AGE, GSE Fabrication	13.25			
Total Nonrecurring Cost		30.29		
RECURRING COST				
• Five Spacecraft and Payloads	13.60			
• Spares	2.40			
• Refurbish and Replacement	13.22			
• Launch Operations	8.28			
Total Recurring Cost		37.50		
Total Cost			67.79	
Profit @ 10%			6.78	
TOTAL PRICE				74.57
OTHER CUSTOMER COSTS				
• Missile Acquisition				4.83
• Recovery Operations				3.42
TOTAL PROGRAM PRICE				82.82

D. ESTIMATED FUNDING REQUIREMENTS

Funding requirements will be identified at the start of the development program. However, the requirement for any one fiscal year is not expected to exceed $25,000,000.

E. ADDITIONAL LAUNCH COSTS

The average recurring launch cost for the program described is $1.3 million per launch. Using the same hardware, additional launches could be made at an increase in cost of approximately $600,000 per launch.