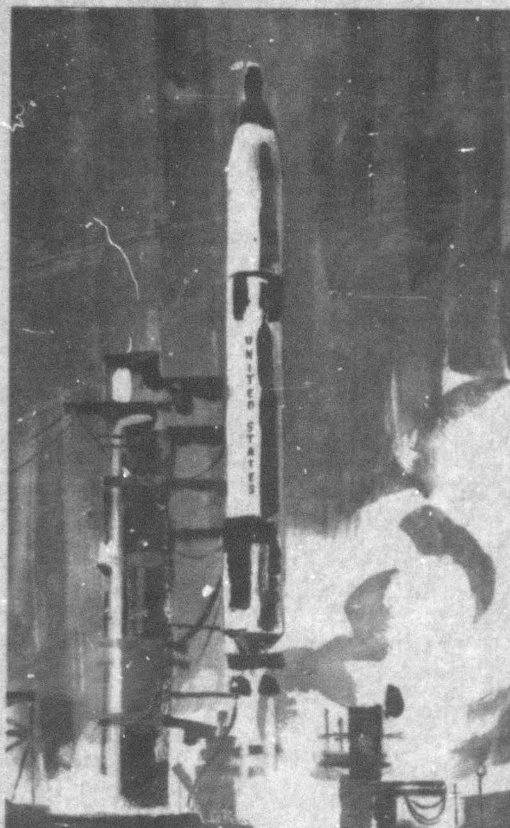
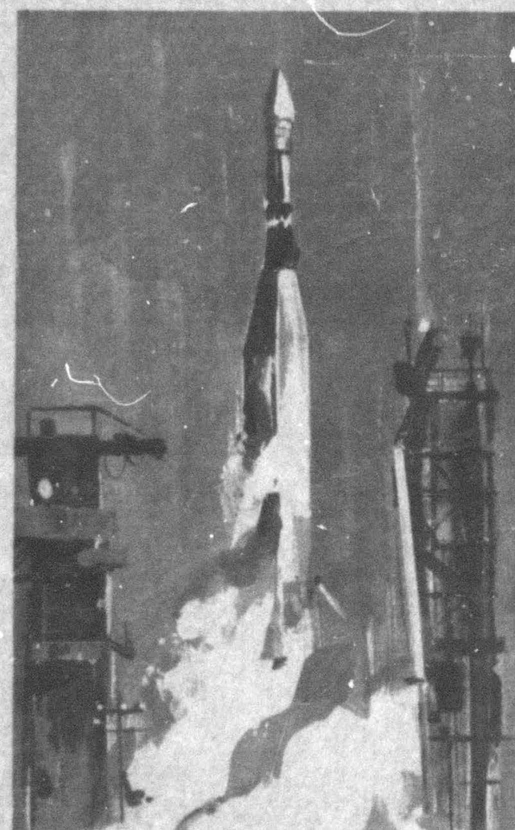
AEROSPACE REPORT NO.
TOR-1001(2126-80)-3

GEMINI PROGRAM

LAUNCH SYSTEMS FINAL REPORT

GEMINI/TITAN LAUNCH VEHICLE · GEMINI/AGENA TARGET VEHICLE · ATLAS/SLV-3





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Gemini Program
Launch Systems Final Report

Gemini/Titan Launch Vehicle
Gemini/Agena Target Vehicle
Atlas/SLV-3

JANUARY 1967

Prepared by GEMINI LAUNCH SYSTEMS DIRECTORATE
El Segundo Technical Operations
AEROSPACE CORPORATION

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Prepared for COMMANDER SPACE SYSTEMS DIVISION
AIR FORCE SYSTEMS COMMAND
LOS ANGELES AIR FORCE STATION
Los Angeles, California

Report No.
TOR-1001(2126-80)-3

GEMINI PROGRAM LAUNCH SYSTEMS
FINAL REPORT

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The information in this technical operating report is developed for the Gemini Program and is therefore not necessarily of broader technical applicability.

FOREWORD

This report is a summary of the management techniques employed in the development, systems test, and launching of the Gemini Launch Vehicle, Gemini Agena Target Vehicle and the Atlas, SLV-3 in support of the NASA Gemini Program. The period covered is from January 1962 through December 1966. Although the document was prepared in large part by Aerospace Corporation, significant contributions were made by the Space Systems Division, Air Force Systems Command and the major contractors involved (The Martin Company, Baltimore; General Electric, Syracuse; Burroughs Corporation, Paoli; Aerojet General, Sacramento; Lockheed, Sunnyvale; General Dynamics Convair, San Diego).

Each section in the report contains its own reference list. These are called out in the text by superscript numerals but are listed at the end of the section. A list of definitions is also appended to each section.

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TABLES

I. SUMMARY

A. SCOPE

The Gemini Launch Systems Final Report contains accounts of three vehicles involved in the ascent phases of the Gemini program: the Gemini launch vehicle, the Gemini Agena target vehicle, and the Atlas standard launch vehicle. Since the first two were not originally developed for the tasks specifically called for in a manned program, the emphasis will be on these vehicles; Atlas (SLV-3) required little or no modification for this program.

The report is somewhat historical in nature, in that it will attempt to trace the conversion of the Titan II and standard Agena into vehicles that would satisfy, respectively, the objectives of orbiting the spacecraft and providing a suitable vehicle for rendezvous and orbital maneuvers. Emphasis has been placed on techniques, controls, and management approach.

As can be seen in Table I. A-1, the flight program was highly successful in meeting all objectives while remaining comparatively free of anomalies. For example, the Gemini Launch Vehicle reliability record speaks for itself: 12 successful missions for 12 launches; 99.9% data recovery; 84 countdown hours - two holds; 1 1/2 hours lost, and 2 shutdowns; all vehicles delivered on schedule and below target cost; and a 75% reduction in test time from GLV-1 to GLV-12. The Agena Target Vehicle achieved four successful orbits out of six flights. Throughout the four flights the specially designed command and communication (C&C) subsystem performed to near perfection, receiving and executing successfully over 10,000 real time, stored program, and spacecraft commands. Discussions of the reasons for the excellent performance of these vehicles begin with a description of the quality and organization of management, which contributed in large measure to the efficiency and high motivation of the program.

Two significant GLV firsts were, 60-day launch centers and the simultaneous countdown of two vehicles. The evaluation and subsequent modification of test philosophy and procedures, both at the factory and the Eastern Test Range (ETR), made these pioneer efforts possible and this subject is discussed.

Problems specific to man-rating the Titan II ballistic missile are dealt with at some length. Techniques developed for the acquisition of trouble-free hardware included not only the exercise of extraordinary reliability assurance, but also the relentless pursuit of problem solutions in the form of continuous technical reviews and these will be described.

Significant firsts associated with the target vehicle were: the first docking of two vehicles in space, first use of non-spacecraft propulsion system for space maneuvering, and a manned altitude record of 741 n mi.

There is a short description of the configuration of the launch vehicles and target vehicle, subsystems, and modifications, as well as of specialized hardware problems. One of these problems concerned the refurbishment of GATV 5001, originally the first flight article, which underwent testing at ETR, remained there as a backup to 5002 and was returned to Sunnyvale after the flight of that vehicle. Special study efforts dealing with other problems resulting from systems test, flight anomalies, and general problems of all the vehicles are also included.

The history of each vehicle, from fabrication through launch and flight evaluation, has been summarized followed by an account of the significant trends that became evident as the program progressed.

Finally, the practical lessons learned from experience on the Gemini Program are briefly stated in the conclusions (Section V).

Table I–1 GEMINI FLIGHT SUMMARY

Mission No. & Sequence	Type Mission	Date	Astronauts	Azimuth	Perigee and Apogee	Launch Window	Mission Duration	GLV Objectives	GLV & Target Vehicle Flight Results	CD Aborts	Launch CD Holds
1	Orbital Unmanned No S/C Separation	8 April 64		72.0° (T)	87 n mi and 161 n mi	3.5 hr	4.25 Orbits	Structures, exit heating & guidance qualification	Successful All objectives achieved and S/C inserted within tolerance requirements	None	None
2	Ballistic Unmanned	19 Jan 65		105° (T)	87 n mi release	3.5 hr	1,848 n mi impact	Separation, Subsystem, qualification	Successful. All objectives achieved and S/C inserted within tolerance requirements	1-shutdown before L/O due to hydraulic switch-over to secondary system (12/9/64)	None
3	Orbital Manned 3 orbits	23 Mar 65	V. Grissom J. Young	72.0° (T)	87 n mi and 130 n mi	3.5 hr	Three orbits	Performance in meeting insertion requirements	Successful All objectives achieved and S/C inserted within tolerance requirements	None	24 minute hold due to oxidizer leak Stage I ducting
IV	Orbital Manned 4 days	3 June 65	J. McDivitt E. White	72.0° (T)	87 n mi and 161 n mi	2.5 hr	long duration 4–days	Performance in meeting insertion requirements	Successful All objectives achieved and S/C inserted within tolerance requirements	None	76 minute hold due to erector lowering proble
V	Orbital Manned 8 days	21 Aug 65	G. Cooper C. Conrad	72.0° (T)	87 n mi and 190 n mi	3.5 hr	long duration 8–days	Performance to S/C insertion; Joint GT GAATV CD compatibility (SLD)	Successful. All objectives achieved and S/C inserted within tolerance requirements	1-S/C problems 19 Aug 1965	None
VI	Orbital Manned Rendezvous with GATV	25 Oct 65	W. Schirra T. Stafford	Variable 80° to 105° (T)	87 n mi and 146 n mi	3.5 hr	2–days	Meet insertion requirements; Joint GT GAATV CD	GT Mission aborted prior to launch following loss of GATV after PPS ignition	1–GT–6 due to loss of Agena 25 Oct 1965	None-GT-6 CD terminated at T–42 minutes
VII	Orbital Manned 14–days	4 Dec 65	F. Borman J. Lovell	83.6° (T)	87 n mi and 183 n mi	2 hr	long duration 14–days	Meet insertion requirements	Successful. All objectives achieved and S/C inserted within tolerance requirements	None	None
VI–A	Orbital Manned Non-Docking Rendezvous with GT–7 S/C	15 Dec 65	W. Schirra T. Stafford	Variable 80° to 105° (T) (81.40 Actual)	87 n mi and 146 n mi	2 hr	1 day	Meet insertion requirements and rendezvous launch requirements	Successful All objectives achieved Launch window limitation requirements met and S/C inserted within tolerance requirements	1–Shutdown before L/O due to premature dropout of Pad Disconnect 3 DIM. GG oxidizer inlet blockage also present. (dust cap) 12 Dec 1965	None
VIII	Orbital Manned Rendezvous and Docking with GATV	16 Mar 66	N. Armstrong D. Scott	Variable 80° to 105° (T) (99.90 Actual)	87 n mi and 146 n mi	2.5 hr	3 days	Meet insertion requirements. Complete Joint GT–8/Target Vehicle launch CD	Same comment as for VI–A plus Agena successfully placed in rendezvous docking orbit	None	None
IX	Orbital Manned Rendezvous Docking with GATV, EVA	17 May 66	T. Stafford E. Cernon	Variable 80° to 105° (T)	87 n mi and 146 n mi	2.5 hr	3 days	Meet insertion requirements and GT/GAATV rendezvous-docking (M = 3); joint launch requirements	GT-Mission aborted prior to launch following loss of GAATV during boost phase due to engine hardover	1-Due to loss of GAATV during boost phase 17 May 66	None GT–9 CD terminated at T–85 min.
IX–A	Orbital Manned Rendezvous Docking with ATDA, EVA	3 June 66	T. Stafford E. Cernon	Variable 80° to 105° (T) (87.40 Actual,	87 n mi and 148.5 n mi	2.5 hr	3 days	Meet insertion requirements and GT/GAATDA Rendezvous-docking (M=3), joint launch requirements	GLV objective successfully achieved. No ATDA docking; shroud failure	1–Malfunction in S/C computer update loop	None
X	Orbital Manned GATV X-S/C Rendezvous & Docking, GATV VIII–S/C Rendezvous EVA	18 Jul 66	J.W. Young M. Collins	Variable 80° to 105° (T) (98.80 Actual)	87 n mi and 146 n mi	1.5 hr (Single-pane days) 2.5 hr (Two pane days)	3 days	Meet insertion requirements and GT/GAATV rendezvous-docking (M=4); joint launch requirements	All GLV objectives successfully achieved; Rendezvous and and docking accomplished	None	None
XI	Orbital Manned Rendezvous Docking with GATV, EVA	12 Sep 66	C. Conrad P. Gordon	Variable 80° to 105° (T) (99.9° Actual)	87° n mi and 151 n mi	1.5 hr (Single-pane days) 2.5 hr (Two pane days)	3 days	Meet insertion requirements and GT/GAATV rendezvous-docking (M=1); joint launch requirements	All GLV objectives successfully achieved; rendezvous & docking accomplished	1–Atlas ground problem 9/10/66	11 min. hold at T–97 sec of combined CD due to S/C hatch leak
XII	Orbital Manned Rendezvous Docking with GATV, EVA	11 Nov 66	J. Lovell E. Aldrin	Variable 80° to 105° (T) (100.6° Actual)	87 n mi and 151 n mi	1.5 hr (Single-pane days) 2.5 hr (Two pane days)	4 days	Meet insertion requirements & GT/GAATV rendezvous docking (M=3); joint launch requirements	All GLV objectives successfully achieved; rendezvous and docking with Agena GAT PPS chamber pressure dip at approx T+140 sec	None	None

1 GEMINI FLIGHT SUMMARY

GLV & Target Vehicle Flight Results	CD Aborts	Launch CD Holds	Significant GLV Flight Anomalies	Resolution of GLV Flight Anomalies
Successful. All objectives achieved and S/C inserted within tolerance requirements	None	None	1. Fuel sensor recovering 2. POGO potentiometer malfunction 3. Post-SECO oscillation 4. Yaw velocity error at insertion	1. Hooded screen designed to collect residues and prevent recovering of sensor optics. 2. Transducer redesigned, installation effective, GLV-3 (ECP MM-307). 3. Change of astronauts' procedures for S/C separation (Initialization of OAMS rate control delayed 2 seconds.) 4. Guidance equations modified by Aerospace to bias out effects of c.g. shift.
Successful. All objectives achieved and S/C inserted within tolerance requirements	1-shutdown before L/O due to hydraulic switch-over to secondary system (12/9/64)	None	1. High Stage I thrust and ISP 2. Stage I hydraulic pressure decay at engine start 3. POGO potentiometer failure	1. Empirical data from Titan II and GLV flights being compiled for more accurate performance prediction. 2. Gauss meter testing of the hydraulic pump components prior to launch. 3. See GLV-1 resolution remark.
Successful. All objectives achieved and S/C inserted within tolerance requirements	None	24 minute hold due to oxidizer leak Stage I ducting	1. Higher than predicted Stage I performance	1. Empirical data from Titan II and GLV flights to be utilized for more accurate prediction of engine performance.
Successful. All objectives achieved and S/C inserted within tolerance requirements	None	76 minute hold due to erector lowering problem	1. Roll transient at L/O due to fuel topping disconnect hangup 2. High Stage II fuel orifice inlet temperature	1. Hangup at disconnect due to dust plug stowage on vehicle position. Dust plug assemblies will be removed prior to liftoff. 2. Leak in hot gas cooler. ECP prepared for installation of internal bypass cooler.
Successful. All objectives achieved and S/C inserted within tolerance requirements	1-S/C problems 19 Aug 1965	None	1. POGO occurred during the period of T + 117 to T + 133 seconds	1. Oxidizer standpipes were uncharged at liftoff. Procedural changes made
GT Mission aborted prior to launch following loss of GATV after PPS ignition	1-GT-6 due to loss of Agena 25 Oct 1965	None-GT-6 CD terminated at T-42 minutes	No GLV Flight – See Flight Results column	
Successful. All objectives achieved and S/C inserted within tolerance requirements	None	None	1. The "Make" of OPP switch after ignition was within specs but slower than desired	1. The pressure orifice diameter was reduced for VI-A and subsequent flights.
Successful. All objectives achieved. Launch window limitation requirements met and S/C inserted within tolerance requirements	1-Shutdown before L/O due to premature dropout of Pad Disconnect 3 DIM. GG oxidizer inlet blockage also present. (dust cap) 12 Dec 1965	None	1. T/M signal strength drop during Stage II flight	1. Antenna deformation due aerodynamic heating. Antenna stiffeners added for subsequent flights.
Same comment as for VI-A plus Agena successfully placed in rendezvous docking orbit	None	None	1 RGS Pitchdown (18%) maneuver at L/O + 330 seconds	1. Low frequency noise in GE Mod III radar data. Attributed to tropospheric condition.
GT-Mission aborted prior to launch following loss of GAATV during boost phase due to engine hardover	1-Due to loss of GAATV during boost phase 17 May 66	None GT-9 CD terminated at T-85 min	No GLV Flight – See Flight Results column	
GLV objective successfully achieved. No ATDA docking; shroud failure	1-Malfunction in S/C computer update loop	None	1. Mod III radar $\dot{p}$ lateral rate channel drift at T + 315 sec	1. Attributed to tropospheric condition.
All GLV objectives successfully achieved; Rendezvous and and docking accomplished	None	None	1. Stage II fuel topping disconnect break at liftoff 2. Venting of Stage I Ox. tank after staging 3. Excessive Stage I oxidizer outage	1. Lanyard rigging change for subsequent flights. 2. Study determined occurrence not unique. No adverse effect on flight present. No corrective action required. 3. Increase mixture ratio dispersions commensurate with GLV and Titan II system histories.
All GLV objectives successfully achieved; rendezvous & docking accomplished	1-Atlas ground problem 9/10/66	11 min. hold at T-97 sec of combined CD due to S/C hatch leak	1. Propellant consumption mixture ratio lower than predicted (large Stage I oxidizer outage). 2. Difference in position indications between actuator 3_1, 4_1 VD amps and actuator position transducer.	1. Biased mix ratio for GLV-12 launch. 2. No corrective action required. Moisture in connector believed to have caused erroneous position transducer output.
All GLV objectives successfully achieved; rendezvous and docking with Agena GAT PPS chamber pressure dip at approx T+140 sec	None	None	1. 204 lb overload Stage I oxidizer 2. Stage II oxidizer pump inlet temp 10 deg high 3. Venting of Stage I oxidizer tank after staging	1. None 2. None 3. None Final Launch in GEMINI Series

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B. <u>GEMINI PROGRAM OBJECTIVES</u>

The primary objectives of the Gemini Program, which were established after it became evident to NASA that an intermediate step between Mercury and Apollo was required, were as follows:

- Long duration flights in excess of the requirements of the lunar-landing mission.
- Rendezvous and docking of two vehicles in earth orbit.
- The development of operational proficiency of both flight and ground crews.
- The conduct of experiments in space.
- The active control of re-entry flight path to achieve a precise landing point.

C. <u>SSD/AEROSPACE ROLE</u> - Gemini Launch Vehicle, Gemini Agena Target Vehicle, and SLV-3 (Atlas)

In 1961, during the proposal stage of the Gemini program, the Air Force Titan II ballistic missile was chosen as the launch vehicle for the Gemini spacecraft.[1] It was agreed that the vehicle could be used for this purpose with only minor modifications.

At this time, a document (Operational and Management Plan for the Gemini Program[2]) was drawn that, in general terms, delineated the responsibilities and division of effort required for the conduct of the program. To summarize, this document assigns the responsibility for development and procurement of the launch vehicle, launch complex, and launch operations to the Air Force under the over-all management of the NASA Program Manager.

More specifically, the Space Systems Division (USAF) was delegated the responsibility for Gemini launch vehicle development and procurement; Atlas procurement; GATV development and procurement; Atlas/Agena system integration; technical supervision for the launch of the GLV and Atlas/Agena vehicles; and range support as required by NASA.

Aerospace Corporation, a not-for-profit organization under contract to SSD, was given the responsibility for general systems engineering and technical direction of the GLV and technical surveillance of the Agena target vehicle and SLV-3. Aerospace was also responsible for the development and implementation of the GLV guidance equations.

In January 1962, a purchase request was issued to the Space Systems Division of the Air Force Systems Command for the development and procurement of a sufficient number of vehicles to satisfy the needs of the Gemini Program.

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[1] Superscripts signify references which are listed at the end of each section.

SECTION I. A

REFERENCES

1. Gemini Midprogram Conference Report No. NASA SP 121, National Aeronautics and Spacecraft Center, Houston, Texas, (23 to 25 February 1966).

2. NASA - DOD Operational and Management Plan for the Gemini Program Unnumbered Report Space Systems Division, USAF, Los Angeles Air Force Station, California, (29 December 1961).

SECTION I. A

DEFINITIONS

ATDA	Augmented Target Docking Adapter
CD	Countdown
EVA	Extra Vehicluar Activity
GAATDA	Gemini Atlas Augmented Target Docking Adapter
GAATV	Gemini Atlas Agena Target Vehicle
GAT	Gemini Agena Target
GATV	Gemini Agena Target Vehicle
GG	Gas Generator
GT	Gemini Titan - (Mission)
M =	Orbit Designation; i.e., M = 1, First Revolution (Spacecraft Term)
NASA	National Aeronautics and Space Administration
OPP	Oxidizer Pressurant Pressurization Switch
PPS	Primary Propulsion System
RGS	Radio Guidance System
S/C	Spacecraft
SECO	Sustainer Engine Cut-off
SSD	Space Systems Division
SLV	Standard Launch Vehicle
T/M	Telemetry
T/V	Target Vehicle
VD	Valve Drive

II. GEMINI LAUNCH VEHICLE

A. INTRODUCTION

1. PROGRAM MANAGEMENT

Figure II. A-1 presents the contractor and government organizations involved in the launch vehicle effort. It shows that 2 major government agencies, 5 major industrial contractors, and 43 industrial subcontractors participated in the Gemini launch vehicle development program. The major government agencies involved in the program were the two NASA centers (the Kennedy Space Center and the Manned Spacecraft Center) and the Air Force Systems Command (AFSC). Within the Air Force, the Gemini launch vehicle program was managed through the Space Systems Division Program Office, which was supported strongly by the Aerospace Corporation providing systems engineering and technical direction for the over-all Gemini launch vehicle program.[1]

The airframe contractor was the Martin Company, Baltimore, with 38 major subcontractors. The Aerojet-General Corporation, Sacramento, supplied the engine system; The General Electric Company, Syracuse, produced the airborne guidance system components; and the Burroughs Company, Paoli, supplied the ground computer and implemented the guidance equations. The Space Systems Division's 6555th Aerospace Test Wing at Patrick Air Force Base, Florida, was assigned the responsibility for preflight checkout of the launch vehicle at Cape Kennedy and for the launch operations.

The spacecraft contractor, the McDonnell Aircraft Corporation, is also shown in the figure because interface relationships were maintained with this contractor, especially in the areas of the malfunction detection system and backup guidance.

a. Space Systems Division (SSD)

The Gemini Launch Vehicle System Program Office (SPO) was originally established within the Deputy for Launch Vehicles, Space Systems Division, as a Directorate. However, from the beginning the Directorate was generally organized in accordance with the 375 Series of Air Force regulations. (See Figure II. A-2.) This organization started with a Director and Deputy Director, a Plans and Programs Office, a Configuration Control and Quality Assurance Office, and then the normal Engineering, Operations, and Contracts Divisions. When the SPO was created on 13 April 1965, the functions and organization were not changed. At that time the organization had a Director and Deputy Director with a Program Control Division and Configuration Management Division, Deputy Director for Engineering, Deputy Director for Procurement and Production, and Deputy Director for Test Operations (Figure II. A-3). Even though the organization was generally as stated in the 375 Series, the System Program Office obtained approval from Headquarters, Systems Command to change some of the minor functions.

A major function was changed in the GLV SPO, the Reliability and Quality Assurance being maintained in the Configuration Management Division. This was done since reliability is very closely allied with Configuration Management and was one of the prime considerations in incentive contracting, the desire being that it approach 100 percent on a manned program.

It was believed that in a program using a mature vehicle with a demonstrated high degree of reliability, all Engineering Change Proposals should be carefully analyzed concerning their effect upon this reliability. Furthermore, in order to maintain maximum configuration control, the Chairman of the Configuration Control Board was the Director or Deputy Director rather than the Chief of the Configuration Management Division. On the Gemini Program this authority was never delegated.

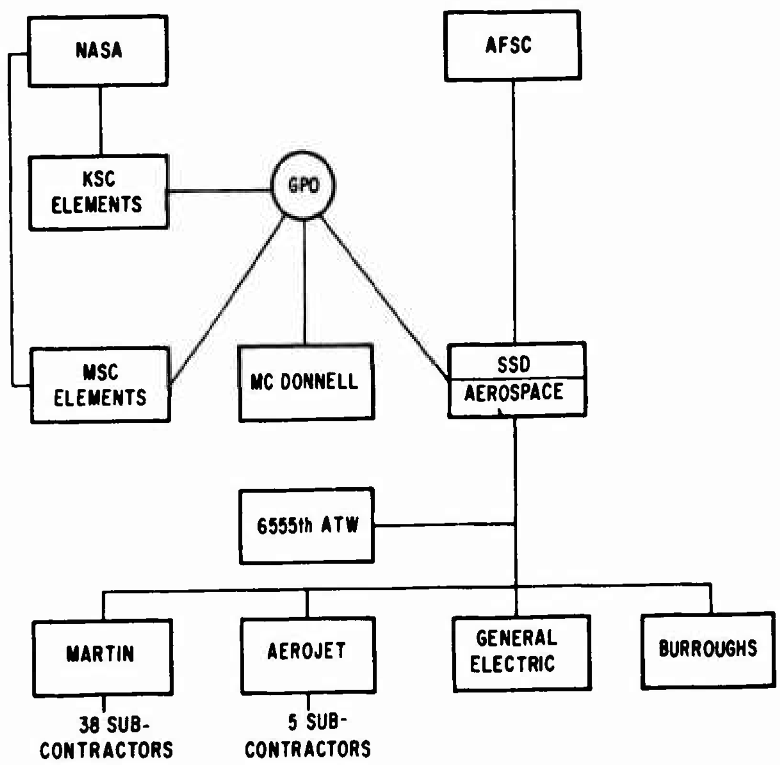


Figure II. A-1 Management Structure, Gemini Launch Vehicle

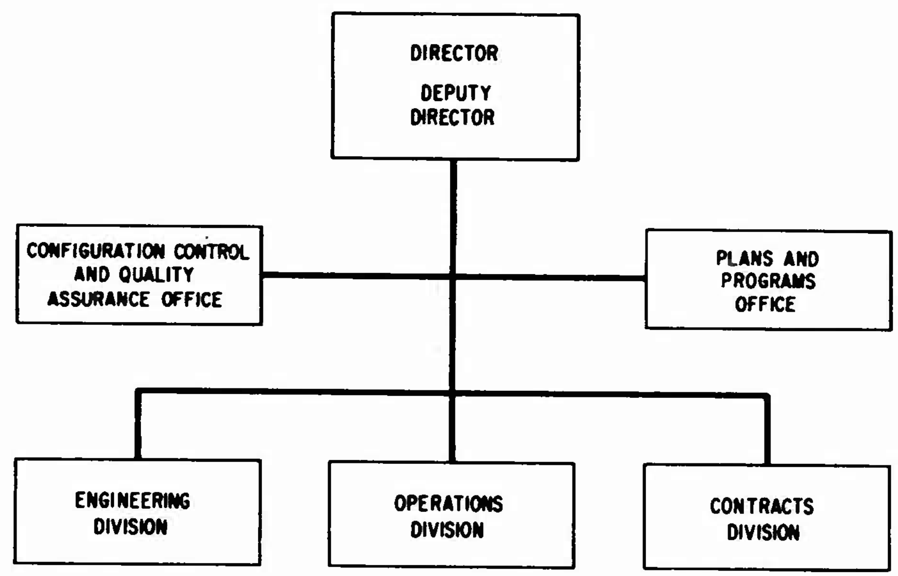


Figure II. A-2 Gemini Launch Vehicle Directorate, SSD

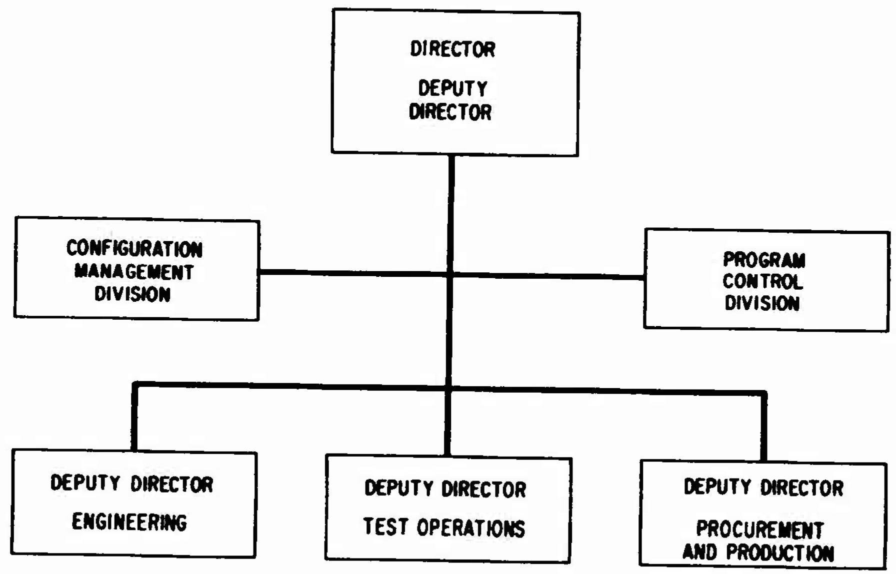


Figure II. A-3 Gemini Launch Vehicle System Program Office, SSD

The function of the Deputy Director for Test Operations differed somewhat from the 375 Series in that no deployment function existed on this program and therefore this Deputy Director was responsible only for the vehicle checkout and launch operation. The function of the Deputy Director for Engineering was much the same as in the 375 Series, except that reliability and quality assurance had been placed in the Configuration Management Division, as previously indicated.

In the duties of the Deputy Director for Engineering and Test Operation, a series of checks and balances were incorporated. The engineering staff was responsible for the development and in-plant testing of the vehicle prior to acceptance by the Air Force. However, at the time of vehicle acceptance, the Deputy Director for Test Operations was the Co-chairman of the Vehicle Acceptance Team along with the Aerospace Corporation personnel. In this manner the Deputy Director for Engineering had to sell his vehicle to the Deputy Director for Test Operations. Likewise, after the vehicle had been moved to the launch site, any change or modification had to have recommendation for approval from the Deputy Director for Engineering before it could be made.

The functions of the Gemini Launch Vehicle System Program Office were as follows:

1) Director and Deputy Director. Managed and directed the development, procurement, and test activities for the Gemini Launch Vehicle.

   Kept NASA Manned Spacecraft Center, Gemini Program Office (GPO), fully informed on activities concerning the Gemini Launch Vehicle.

2) Configuration Management Division. Managed the Gemini Launch Vehicle configuration identification, control and accounting system.

   Managed the reliability and quality program for the launch vehicle.

   Supervised contractor efforts in formulation, execution, and discipline of configuration control systems.

   Maintained records of approval configuration and actual configuration.

   Maintained reliability records and monitored qualification test program.

3) Deputy Director for Engineering. Responsible for over-all design, development, fabrication, assembly, and all in-plant testing through factory acceptance of the propulsion subsystems and the complete GLV to meet required technical and operational parameters.

   Approved technical content of specification definition and provided technical support on all matters pertaining to this equipment with the exception of the instrumentation and range safety subsystem.

   Coordinated engineering effort with the GPO.

4) Deputy Director for Procurement and Production. Managed and coordinated the procurement and production effort.

   Acted as principal contracting officer for contracts under GLV jurisdiction.

   Served as focal point on all procurement matters; negotiated contracts and Contract Change Notifications.

5) Deputy Director for Test Operations. Responsible for managing and coordinating all functions involved from the point of GLV factory acceptance through launch operations.

   Responsible for technical performance and support for the instrumentation and range safety subsystem, launch facilities, abort, trajectory, AGE, and astronaut safety aspects.

   Monitored installation of AGE, incorporation of changes thereto and of GLV changes after factory acceptance.

   Coordinated operations with the GPO.

6) Program Control Division. Responsible for managing and coordinating the following functions: Program evaluation and analysis from Program Evaluation Review Techniques (PERT), contractors, Aerospace, and Program Office reports.

   Prepared and maintained Development Plan, Management Review, and briefings for Commander and GPO.

Determined, defined, acquired, and controlled financial data to assure timely availability and efficient use of funds.

b. Aerospace Corporation

Aerospace Corporation placed the responsibility for the Systems Engineering and Technical Direction of the Gemini Launch Vehicle within the Manned Systems Division of the El Segundo Operations (Figure II. A-4).

Specifically, the Gemini Launch Systems Directorate was formed as the focal point (Program Office) of all Gemini activities. The Directorate was organized in three parts: Systems and Guidance Analysis, Airborne Systems Engineering, and Systems Test Operations (Figure II. A-5).

(1) Systems and Guidance Analysis Office

The Systems and Guidance Analysis Office was responsible for the guidance, performance, and flight test evaluation of the Gemini Launch Vehicle. The guidance function included responsibility for the design, implementation, and validation of the GLV guidance equations, and for the guidance program as wired into the Burroughs A-1 guidance computer at the Mod III radar site at ETR. The analysis of guidance performance on each flight, reflecting numerous NASA-directed mission changes, and incorporation of computer hardware changes were also supervised by this office. The Performance Analysis Section was responsible for generating payload capability predictions for the GLV and continuously analyzing flight test results to improve prediction techniques. Extensive studies were also conducted in the area of performance improvements. The Flight Test Evaluation Section within this office was responsible for all post-flight analysis, including generation of quick-look and final flight test reports and chairing post-flight evaluation meetings with the associate contractors.

(2) Airborne Systems Engineering Office

The Airborne Systems Engineering Office within the Gemini Launch Systems Directorate was primarily responsible for all airborne systems for both the GLV and the Gemini Agena Target Vehicle (GATV), i.e., both entire vehicles. Additionally, in conjunction with the Systems Test Operations Office, this group shared responsibility for the AGE associated with the airborne systems. The systems total responsibility covered all aspects of program office technical management and included interfacing with the Air Force, the contractors, and NASA, as well as with internal Aerospace Corporation support units.

The responsibilities within the group were divided according to functional subsystem areas, i.e., aeromechanical systems, flight mechanics, propulsion systems, etc.. To cover both the GLV and the GATV with the minimum number of personnel, the same functional system areas of both vehicles were under the supervision of a single Manager with subsystem engineers assigned to the various subsystems of each vehicle reporting to him. The Airborne Systems Engineering Director, in turn, had two Assistant Directors, one for the GLV and one for the GATV.

The airborne systems subsystem engineers were totally responsible for their assigned subsystems. As such, they were responsible for following their systems' development effort; approving specifications, ECP's, and test procedures; participating as members of vehicle acceptance teams; assuring that their respective subsystems were ready for flight; reviewing post-flight data; and evaluating and documenting the flight performances of their respective subsystems.

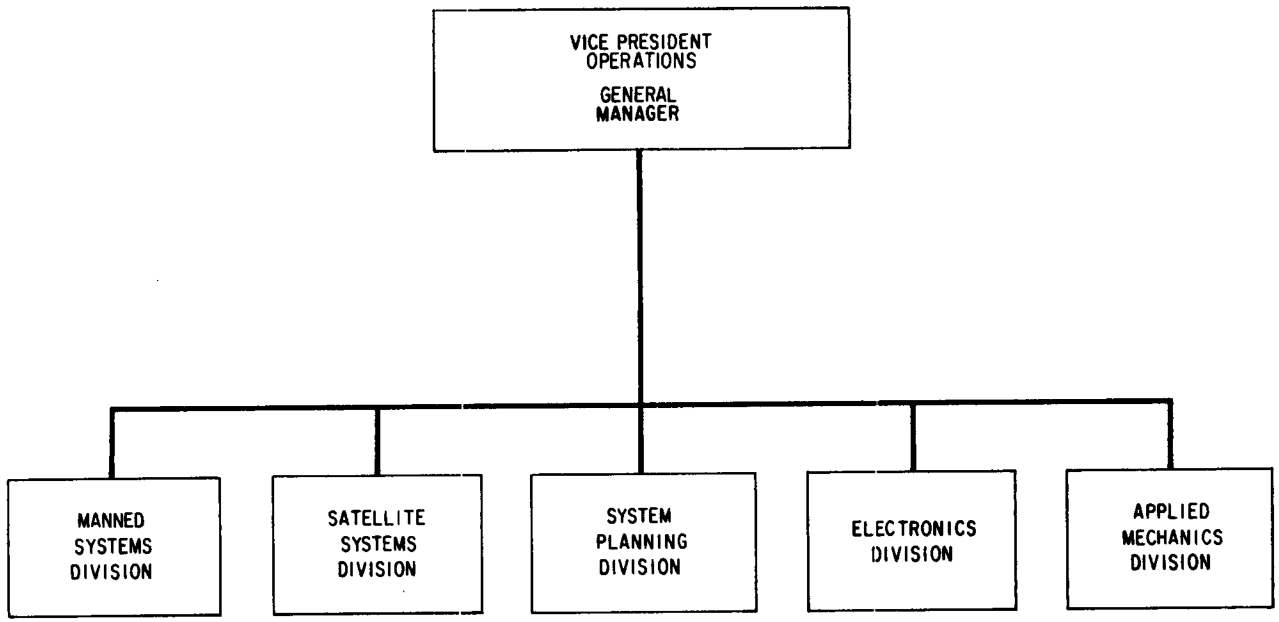


Figure II. A-4 Aerospace Organization Chart - Operations

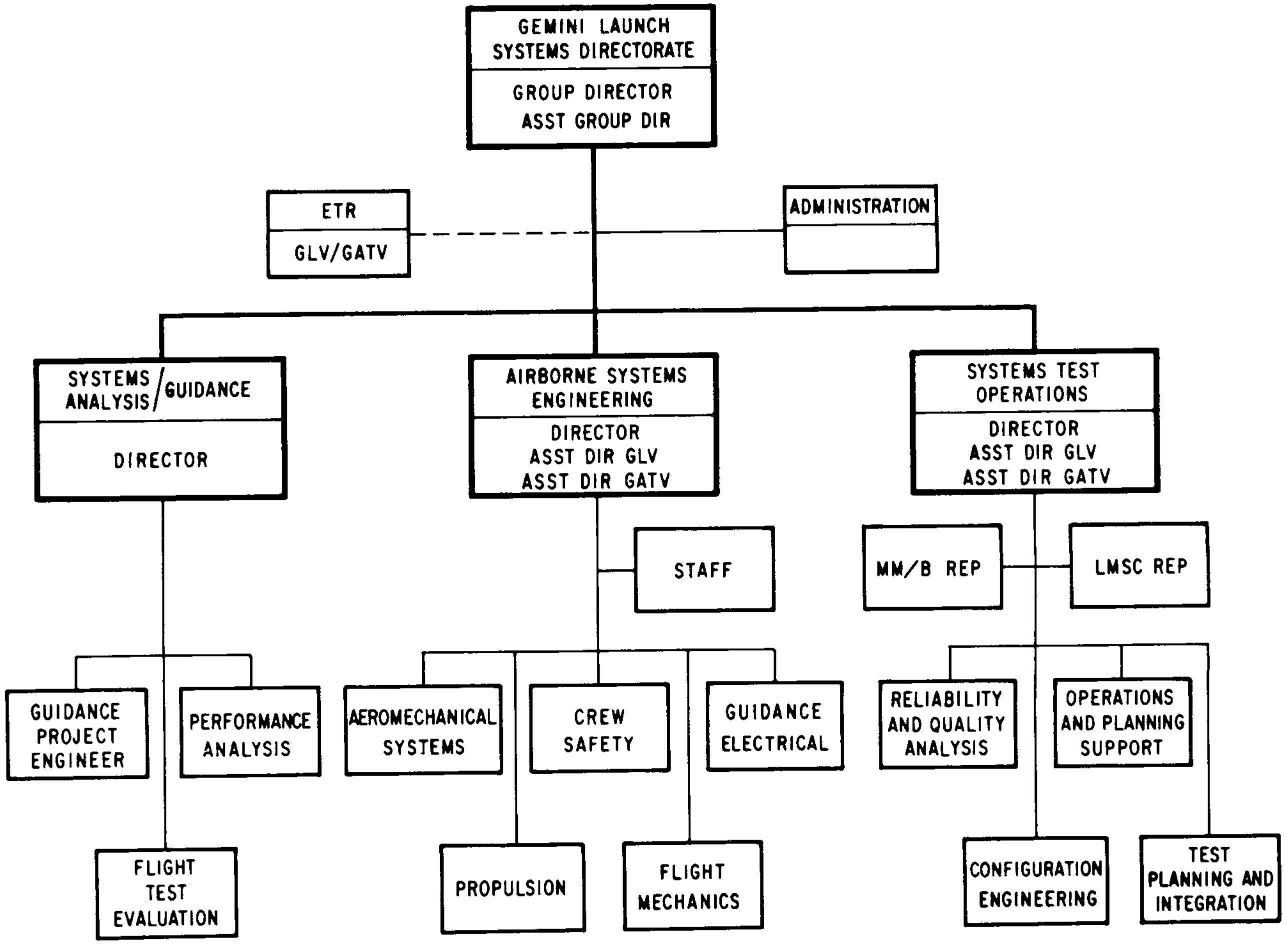


Figure II. A-5 Gemini Program Organization, Aerospace Corporation

(3) Systems Test Operations Office

The fundamental objective of the Systems Test Operations Office was to provide program control mechanisms that could be used by the systems engineers and management to properly define the technical acceptability and the operational readiness of Gemini hardware.

In support of this objective, all significant testing was defined down to a level of detail sufficient to assure continuity and control. An uncompromising reliability effort was developed to identify and dispose of any hardware weaknesses. A highly objective configuration control system was established and administered in such a way as to assure maximum discipline in this area by all agencies. An aggressive hardware acceptance program was also established to assure maximum product integrity. Each of these areas encompassed responsibilities for both the Gemini Launch Vehicle and the Agena Target Vehicle, and the utilization of manpower was left to first-level supervision. In order to provide effective liaison with the contractors, this office maintained permanent field representatives at Martin-Baltimore and Lockheed Missile and Space Company (LMSC).

The Gemini Directorate was staffed with specialists in all disciplines required to perform the general systems engineering effort and maintain the every-day contact with SSD and the associate contractors. However, when problems arose that required a more concentrated effort or specific technical or scientific skills, the Directorate was augmented with personnel from the engineering subdivisions (Electronics Division and Applied Mechanics Division) of the El Segundo Operations. These divisions were also called upon to supply manpower to staff special task teams, which had to be established from time to time.

(4) Guidance Equations

Aerospace Corporation also was responsible for the development and implementation of the Gemini guidance equations. The prime responsibility in this area rested with the Guidance Project Engineer in the Systems and Guidance Analysis group of the Program Office. Procedures were developed for the direction by the Program Office of groups within the Aerospace Electronics Division designing the guidance equations and validating the guidance program. Procedures of strict formality persisted throughout the program in the area of guidance equations verification, guidance program validation, and equations and program certification. Prior to each flight, the Gemini Launch Systems Directorate certified in a letter to SSD that the specific computer program and equations were satisfactory for guiding the vehicle into the prescribed orbit.

(5) ETR

The Aerospace organization at ETR is under the jurisdiction of El Segundo Operations. The Gemini activities were headed by a Project Manager who was technically responsible to the El Segundo Gemini Directorate. He, in turn, drew his local support for systems engineering from the ETR technical and operational staff. The members of this staff maintained constant contact with their West Coast counterparts, the subsystems engineers of the Gemini Launch Systems Directorate.

c. Contractors

(1) Martin Company

Development and production of the Gemini Launch Vehicle (GLV) occurred during a time when the Martin Denver plant was deeply involved in Titan II and III development and production. The nature of the Gemini Program required that the GLV be fabricated in a separate assembly line. To meet this requirement in Denver, it would have been necessary to combine two of the Titan assembly lines in order to make a separate line for the GLV. The concurrent vertical testing of the GLV and the Titan vehicles at the Denver facility would have resulted in delays to both the Gemini and Titan Programs.

There was adequate space for a separate assembly line in the Baltimore plant where the engineering cadre for Gemini, which had worked there on the DYNA SOAR Program, was located. As a result of these factors, it was determined that with the exception of tank fabrication, the GLV Program should be located at the Martin Baltimore facility. This decision provided for optimum utilization of personnel, tools, and facilities within the Martin Marietta Corporation.

Martin utilizes the program (line) and functional (staff) system throughout its plants and divisions. The program director (line) was responsible for the how, when, where, and budgetary controls required to achieve the program objectives. The functional directors (staff) were responsible for supplying talent and experience (people), and for the technical excellence of the work. Almost all program personnel were gathered together physically on the program, although only the assistant program managers were administered by the program director.

The Gemini Launch Vehicle program at Martin was organized on a project team basis, as are all major programs, with the major functional specialties being represented on the program managers' team (Figure II. A-6). The assistant program manager at this phase of the program applied his resources primarily in support of the activation of Complex 19 at Cape Kennedy (then Cape Canaveral). ESPD Operations represented Martin's Electronic Systems Products Division prior to its incorporation into the Bunker-Ramo Company. Organizational changes at the program level were kept to a minimum throughout the program, although there were some made primarily to keep abreast of the changing character of the work being performed, from design and development on through production and launch. They included the establishment for several years of a business manager when the volume of matters in the area increased.

The significant, organizationally unique characteristics of this program were at the subprogram level (i.e., within the functional organizations) and primarily in the Quality Assurance and the Engineering Departments. Very early, the Quality Assurance Department established a Failure Analysis and Corrective Action organization, which worked in very well-coordinated activities with the Engineering Reliability organization; these groups were located together.

There were two important philosophies established early in this program that required some special organizational features within the Engineering Department: The final acceptance testing of the launch vehicles was assigned to Engineering; and it was determined that <u>all</u> flight hardware would be <u>fully</u> qualified in its <u>final</u> configuration <u>before</u> delivery of the first-flight article to the Cape. To meet these responsibilities, Engineering established an Assistant Technical Director for Test and Reliability and, within that organization, a Manager of Acceptance Testing and a Manager of Component and Qualification Testing. The position of Acceptance Test Manager remained for the entire program, while that of Component and Qualification Test was abolished after that testing was completed.

It is the policy of Martin to keep organizations flexible and responsive both to program phases and needs as well as to personnel strengths; i.e., organizations are arranged to meet immediate program objectives as well as to augment and utilize an individual's capabilities. An example of the latter organizational activity was the change in technical directors in mid-1963 to provide for the changing character of the program, from design and development to production and flight.

GLV was a multidivisional program within Martin. The vehicle tanks were built in Denver; the remaining structure and assembly were built and the acceptance testing was done in Baltimore; and the checkout and launch took place within the Canaveral Division at The Cape. The program was managed and controlled from the Baltimore Division. The Denver and Canaveral Divisions had

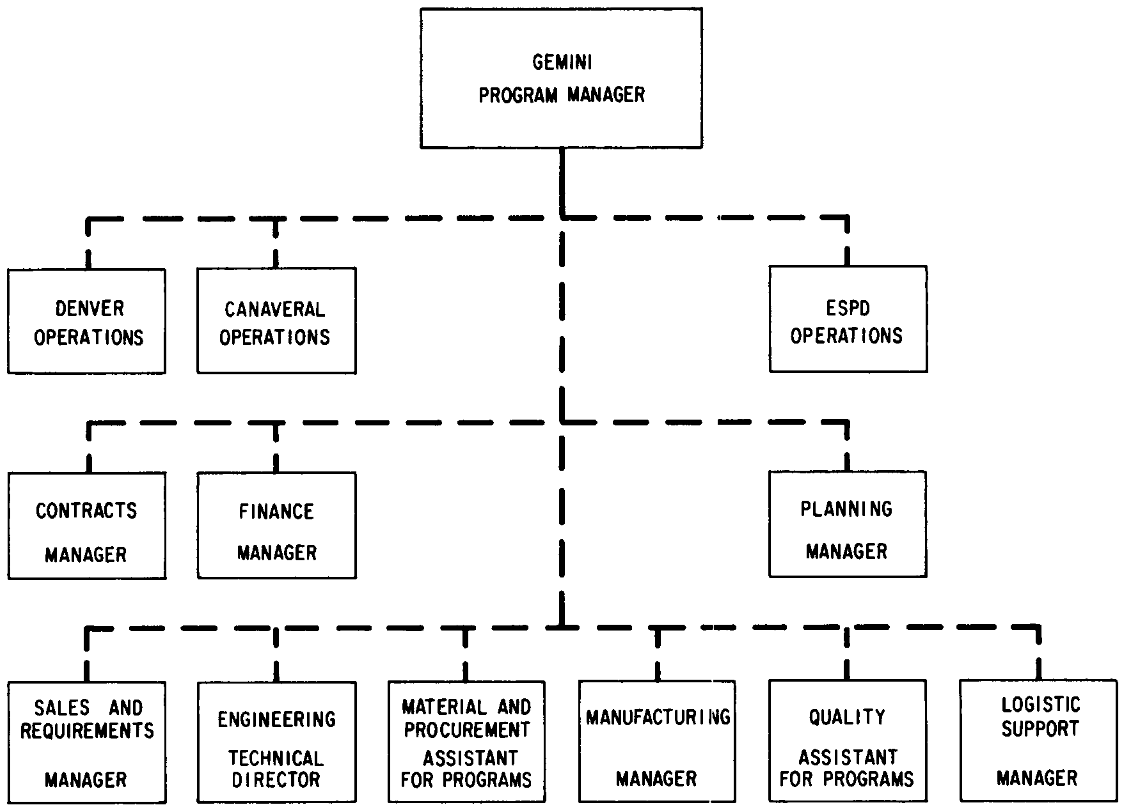


Figure II. A-6. Martin Marietta Corporation - Organization Chart - Gemini Program autonomous organizations within their divisions responsible through their own managers to the program director in Baltimore, and both divisions maintained a representative full time at the home plant for coordination and communication responsibilities (Figure II. A-7). In this organization chart, the divisional managers appear similar to those in any other functional area of responsibility.

Unique among programs of this nature, Martin established the position of Vehicle Chaperone. He was a man who lived and worked with an individual vehicle from the beginning of tank fabrication at Denver, through assembly and test in Baltimore, and checkout and launch at Canaveral. This chaperone constituted an excellent point of historical continuity, as well as adding significantly to the quality of the documentation at each buyoff point during the vehicle progress.

In general, special needs and unique objectives were serviced by special task teams or personnel assigned to ensure coordination across lines of functional responsibilities.

(2) Aerojet General Corporation

The Aerojet General Corporation (AGC) Gemini Program organization was structured to obtain the maximum autonomy, insofar as program requirements, controls, and management were concerned (Figure II. A-8). Attention was focused on direct-line reporting, not only on the primary Program Office functions (e.g., Contracts, Reliability, Pilot Safety, Program Controls, and Engineering), but also on the support organizations (e.g., Test, Quality Control, Product Support, Manufacturing, and Purchasing). Each of the support organizations appointed a Functional Program Manager who reported directly to the respective Division Manager, as well as to the Gemini Program Manager. These representatives were single-point contacts with full Gemini responsibility for their divisions. The AGC Field Managers at Baltimore and the Eastern Test Range also had direct communication with, and authority from, the Gemini Program Manager and the Product Support Division. On technical items, direction to the field was afforded by the Program Office.

The organizational structure described also closely paralleled that of the Air Force Program Office. As such, each AGC Program Office department manager was provided an Air Force counterpart, thus enabling them to discuss the program elements, as required. Again, single-point interface was the rule rather than the exception.

Development programs, such as the Gemini Stability Improvement Program (GEMSIP) and the Augmented Engine Improvement Program (AEIP), which were conceived for the benefit of Gemini, were directly under the cognizance of the Gemini Program Manager.

(3) General Electric Company

The General Manager of the Special Information Products Department reported directly to the General Manager of the Defense Electronics Division, which in the corporate structure of the General Electric Company is a part of the Aerospace and Defense Group. The Department, with headquarters at Syracuse, New York, has the basic organizational elements of engineering, manufacturing, marketing, finance, and employee relations and only those germane to the Gemini Program are described.

The Manager of Radio Guidance and Support Operations (RGO), who reported directly to the General Manager, assigned specific management responsibilities for defined program areas to a Program Manager. The Gemini Program Manager, who reported administratively to and received program direction from the Manager of Radio Guidance and Support Operations was responsible for all Gemini Program functions. The structure of the Gemini Program Management is illustrated in Figure II. A-9. Individual contributors were assigned from components within the Department to

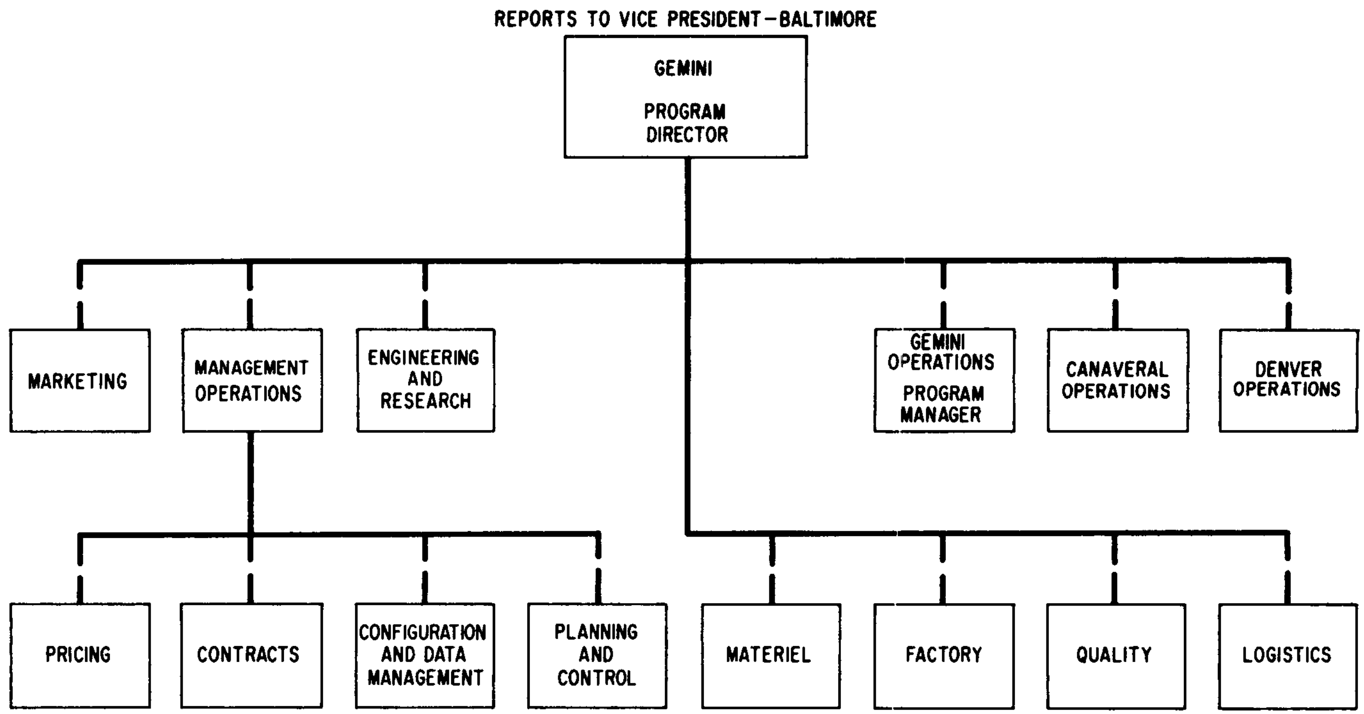


Figure II. A-7. Martin Marietta Corporation, Final Organization, Gemini Program

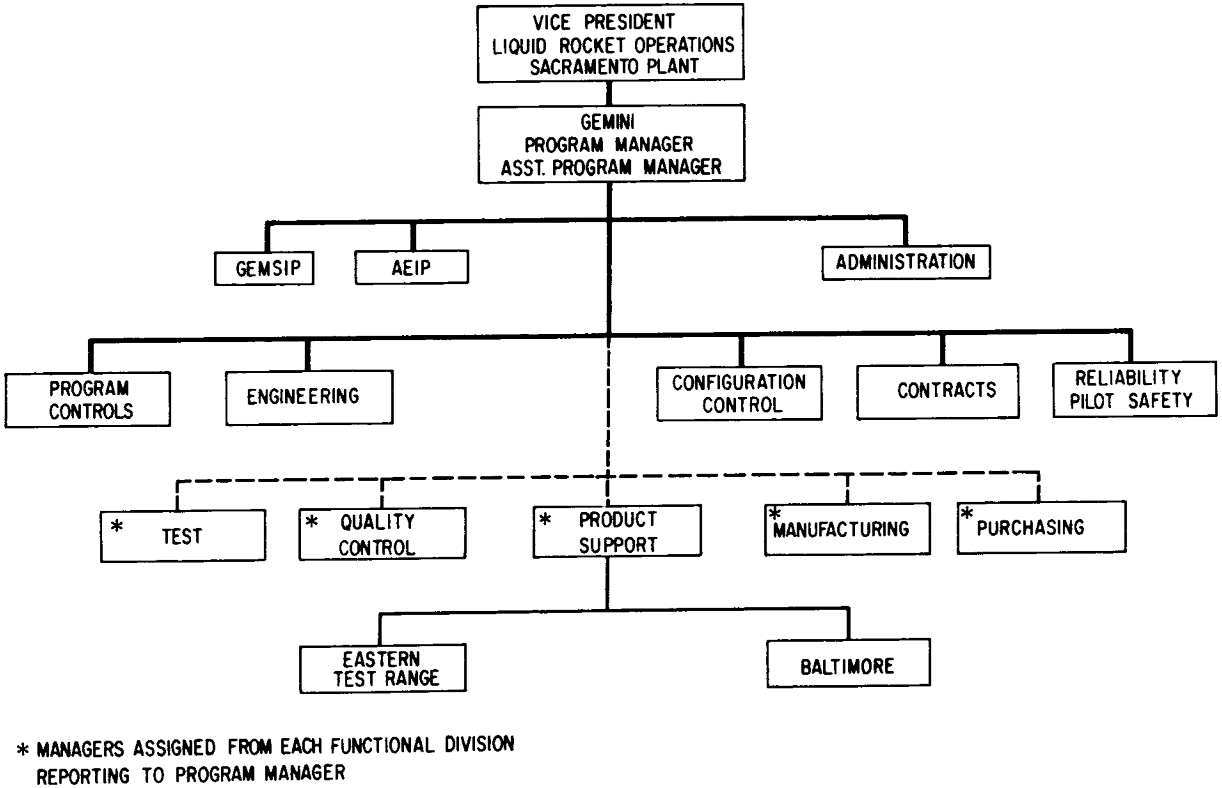


Figure II. A-8. Gemini Program Organization, Aerojet General Corporation

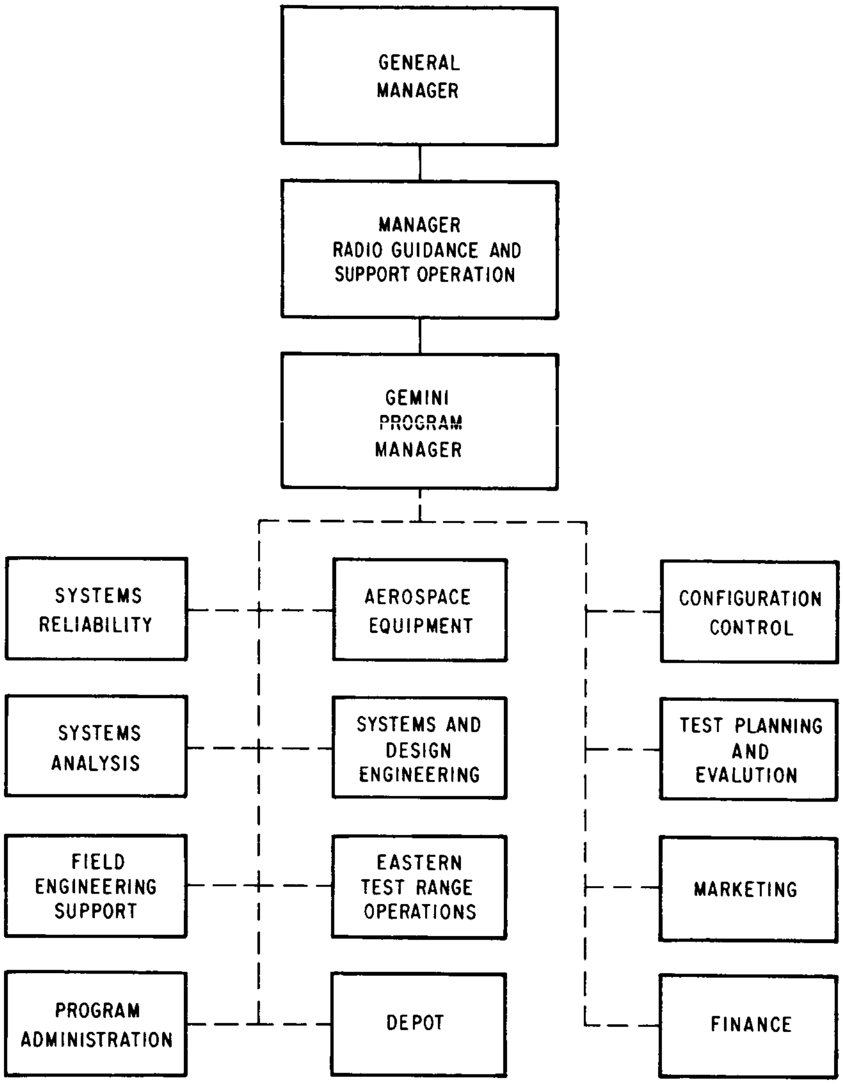


Figure II. A-9. Gemini Progam Organization, General Electric Company receive technical direction from the Gemini Program Manager, while doing administrative reporting to their respective managers. This allowed program requirements to be met within the framework of the departments' policies, instructions, and procedures, but created the flexibility necessary to meet the problems arising from changes in functional scope.

In addition, a Materials subsection was established in the RGO Aerospace Equipment Section to provide for purchasing, receiving, stocking, and kitting of material, and the shipping of finished products. Most of these services had been furnished by the Light Military Electronics Division of G. E..

The Quality Control organization was completed and staffed. It was composed of MOD III/Gemini quality assurance, vendor assurance, quality control engineering, and quality systems.

(4) Burroughs Corporation

For the Gemini Program, the Burroughs Corporation maintained its home office at Paoli, Pa., to which its ETR operations were responsible (see Figure II. A-10). Both field and home offices were separated into hardware and software functions.

The guidance program was written in Paoli and the system interface check programs at ETR. New equipment was manufactured in Paoli, then delivered to the field where it was tested and accepted.

The procedures established in the Burroughs Mercury contract for pilot safety were continued for Gemini. A key concept, contributing to the success of the program, was making all tests resemble the standard countdown and thus implying repeated launch rehearsal. Another important factor was the following mandatory documentation:

(a) Maintenance Records. In which were noted the equipment identification, specified intervals between maintenance checks, and the dates and results of the tests.

(b) Failure Reports. The results of all critical component failure analyses performed in Paoli were reported. Failure trends or the need for mass replacement were discerned and acted upon. Weekly failure reports were presented at the Mod III Working Group Meetings at ETR, and a bound report was published monthly.

At the end of the Gemini program in November 1966, Burroughs was able to report that there had been no critical failure since 26 October 1964.

d. 6555th Aerospace Test Wing

The mission of the Gemini Launch Vehicle division, 6555th Aerospace Test Wing was to technically direct and supervise the checkout, data evaluation, problem resolution, and launch of the Gemini launch vehicle at Cape Kennedy. In addition, the division provided management control of the launch vehicle associate contractors, integrated contractor and government efforts, and assured proper range support and data during the checkout and launch sequence.

The Gemini division established procedures and techniques that resulted in a close-knit operational team. This team was composed of The Martin Company for airframe and electrical systems; Aerojet General for engines; General Electric for guidance; Burroughs for the guidance computer; Air Force for quality-inspection; Aerospace for technical direction; and Pan American Airlines and Radio Corporation of America for range facilities and instrumentation. Daily planning and testing required the same teamwork with NASA, and Mc Donnell, the spacecraft contractor.

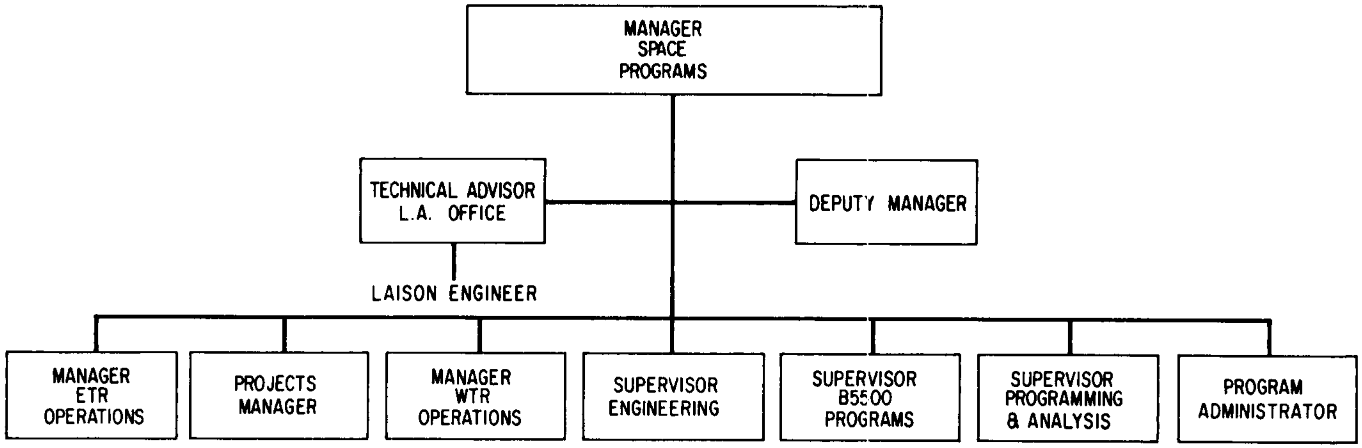


Figure II. A-10. Gemini Program Organization, Burroughs Corp.

To integrate testing, meet schedules, and provide efficiency and maximum launch vehicle reliability and astronaut safety, the division was organized into two branches, flight test operations and crew safety (see Figure II. A-11). The operations branch was responsible for test schedules and the technical adequacy of all work at Complex 19. The pilot safety branch provided a second level of technical review of all work accomplished and acceptance of launch site problem investigations and corrective action.

Gemini division personnel reviewed in detail all schedules, test procedures, hardware modifications, test results, and technical readiness of the launch vehicle systems for launch. All work done on the launch vehicle required prior approval in work authorization documentation. All checkout procedures were reviewed, approved, and signed before being run. Data derived from these procedures were checked from both an in-line operation point of view and a redundant review by specialized pilot safety working teams. All changeouts of a critical component had to be approved by a representative of the Gemini Launch Vehicle Division. Any component that failed had to undergo a thorough failure analysis and corrective action had to be established prior to subsequent flights.

The scheduling with the range of all Complex 19 Gemini tests, both launch vehicle and spacecraft, was performed by the Gemini division. By the consolidation of this function in one organization, testing requirements and schedules of spacecraft and launch vehicle testing were efficiently submitted to the range.

The Gemini Launch Vehicle Division was the agency that reviewed, approved, and levied all GLV data requirements on the range. Prior to each Gemini launch, a detailed review of all GLV data requirements was made to insure that all agencies received adequate data in time for post-launch evaluation. By this action, the range was able to reduce the amount of data being provided for each launch and to supply required data much earlier.

2. SSD/AEROSPACE/CONTRACTOR/NASA INTERFACE

a. General

Obviously, with such a large, diverse, and farflung group of organizations participating in the Gemini Program, the two major management problems were adequate and timely communications, and proper control and coordination of the activities of the separate participants. When these problems were further considered in the light of the relatively short time allowed for development and procurement of the launch vehicle, both NASA and the Air Force recognized early in the program that a system of cooperative program direction and problem reporting would be beneficial.

Time simply was not available for the conventional chain-of-command operation. Consequently, a launch vehicle coordinating organization was formed, headed by a Chairman from the NASA Gemini Program Office and an Associate Chairman from the Space Systems Division Program Office. The group was composed of representatives of all the government and industrial organizations that participated directly in the launch vehicle program, plus representatives of all government or industrial groups that had an interface with the launch vehicle program. The organization of this group went through a number of changes and eventually arrived at the form shown in Figure II. A-12. This panel-type organization had the advantage of grouping people of like specialties and resulted in smaller discussion groups, which allowed more detailed treatment of problems.

A normal coordination meeting lasted two days, the first of which was devoted to panel meetings. On the second day, reports from the panel chairmen were presented to the assembled committee, and

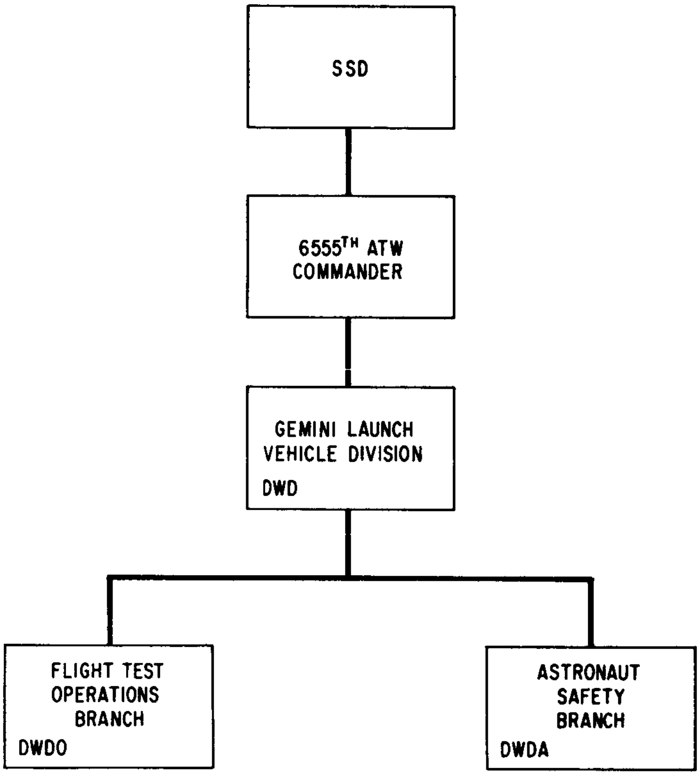


Figure II. A-11. 6555th Aerospace Test Wing

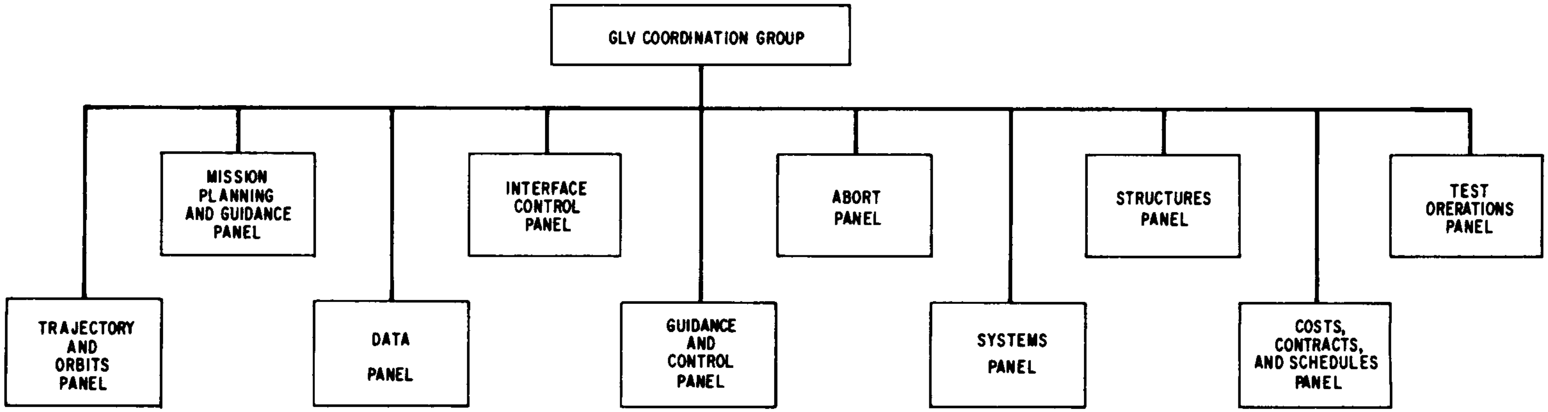


Figure II. A-12. Gemini Launch Vehicle Coordination Group and Reporting Panels recommendations for courses of action were proposed. This was followed by a government session devoted to discussions of action items and financial matters. Meetings were originally held at intervals of two weeks, later increased to three weeks, and then monthly. Halfway through the program, one meeting was scheduled before each mission. The product of this coordination group was a series of action items composed and approved by the group. Authority for the execution of these action items was provided by the NASA Gemini Program Manager to SSD.

In operation, the coordination group provided the status monitoring required to properly assess the progress of the launch vehicle program. It also made possible the rapid identification of problem areas in hardware development and, more importantly, it brought to bear the talents of a large group of knowledgeable people on these problems. The effects of proposed solutions on other facets of the total program were evaluated quickly, and knowledge of changes was disseminated rapidly. A short description of how some of these panels functioned follows.

b. Interface Control Panel

This panel was established early in the program to effectively control the interface between the launch vehicle and the spacecraft and to issue joint drawings and reports of the interface area. The panel did not make policy decisions and referred policy problems to NASA GPO and SSD/Aerospace for resolution.

The panel was given the responsibility of generating the Gemini Spacecraft/Launch Vehicle Interface Specification and Control Document (ISCD-1), that formally established and maintained the launch vehicle/spacecraft compatibility. It contained requirements (text and drawings) covering the mechanical, electrical, and AGE interfaces, and the associated testing necessary to validate interface integrity. It also specified in detail the procedures to be followed in proposing changes to the specification.

Throughout the Gemini program, largely due to the efforts of the Interface Control Panel, no serious interface problems were encountered.

c. Systems Panel

The GLV Systems Panel was one of the most important and widely used of all the GLV panels. Its membership represented NASA, SSD, Aerospace, Martin, Aerojet, and McDonnell. It was an instrument whereby the major program participants could review both their individual and mutual program systems problems, which permitted better understanding of the problems and their program impact. Dissemination of this type of information to all parties involved was very beneficial, preventing a large impact on the program from problems due to lack of knowledge of their existence. Problems affecting hardware that were brought forth in other panels, such as Abort and Interface, were also referred to the Systems Panel for necessary action. Conversely, hardware problems affecting procedures, pilot safety, etc, were referred to the cognizant panels by the Interface Panel.

d. Abort Panel

The Gemini Abort Panel was chartered to define, study, and report on problems associated with crew escape in the event of a launch vehicle failure during powered flight. It consisted of NASA members from the Astronaut Office, Crew Training, Flight Operations, and Gemini Program Office; industry members from McDonnell, Martin Baltimore, Martin Cape, Aerojet, and Aerospace; and USAF members from SSD and the Test Wing. The panel provided a means for coordinating study efforts requiring supporting data from various member groups and permitted immediate review and dissemination of the results.

e. Interfaces - Software

Coordination in the mission planning, launch guidance, and performance areas was effected through the creation of three panels by NASA. There were the Mission Planning and Guidance Panel, the Launch Guidance and Control Panel, and the Trajectory and Orbits Panel. The Mission Planning and Guidance Panel coordinated input from the other two, assigned action items, and resolved areas of disagreement. The Launch Guidance and Control Panel interpreted the mission plans in terms of requirements for guidance hardware and software. The Trajectory and Orbits Panel established requirements for trajectory simulation, assigned studies on the feasibility of various mission plans, and evaluated the effects of these on the GLV payload capability.

Their meetings were attended by spacecraft and target vehicle contractors, as well as SSD/Aerospace and GLV associate contractors. As mission plans were finalized, it was no longer necessary to continue the Mission Planning and Guidance Panel, so it was dissolved and its remaining functions were taken over by the GLV Coordination Panel. All action items requiring response from the GLV associate contractors were implemented by Aerospace through the various technical direction meetings held with these contractors.

Since the relation between the spacecraft design weight and the GLV payload capability was a critical one throughout the program, it was handled as a separate topic in the GLV coordination meetings.

Data on the predictions of GLV performance capability was transmitted to NASA through a formal monthly performance report published by Aerospace Corporation, which became the official data source used by NASA/GPO in publishing their own performance report for NASA Headquarters. This report also contained summaries of various payload improvement studies and other analytical investigations, and was further expanded to include flight test results when this phase of the program was initiated.

f. Data Panel

The Data Panel resolved problems associated with data acquisition, processing, and delivery schedules. The end product of its meetings was a thorough working knowledge on the part of all the GLV contractors and NASA as to the source of the best data, its availability, and the types of data received by each contractor.

Following the Aerospace Post-flight Evaluation Meeting (L + 15 days), the Aerospace and SSD members of the NASA Mission Evaluation Team contributed to and coordinated on the GLV section of the NASA Mission Report.

3. CONTRACTS

The contracts for the Gemini Launch Vehicle and its engines were some of the first to have been completed in the U. S. Air Force implementation of the multiple-incentive fee feature. The incentives in these contracts were based on cost, delivery, countdown performance, and flight performance.

On the whole, the incentive fee was a very effective contracting tool in the program. It certainly helped to keep the importance of the key events (cost, delivery, countdown and flight performances) constantly in the minds of the contractors.

The Cost Incentive feature on the Martin and Aerojet contracts differed in one respect. The Martin contract employed a ±3 percent dead-band feature, whereas the Aerojet contract started the cost sharing as soon as the target cost was underrun; both contractors were on a 90/10 sharing arrangement. It is concluded that a dead band is not desirable; moreover, it is also possible that a greater sharing arrangement (e. g., 80/20) would provide an even greater incentive for cost underruns.

The negative incentive fee for delivery delays worked exceptionally well. If the original delivery schedule included in a contract proves to be the realistic throughout the program, the contractor should certainly be penalized for failure to meet those dates.

The countdown incentive likewise proved to be a valuable tool. It is particularly so when there are short launch windows and/or rendezvous flights. It was mandatory that the contractors make every possible effort to assure an on-time launch. (It is believed that the amount of countdown incentive fee could vary depending on the type of mission, with the maximum countdown incentive fee given for very short launch windows necessary for rendezvous flight.)

Needless to say, the maximum incentive fee must be given for flight performance, which naturally determines the success of the mission. With the extremely high cost of each mission, all possible efforts to assure success were necessary. Flight performance incentives were one of the most effective ways to insure this success.

During the course of the Gemini contract, there was only one area that caused much concern. When the incentive fees were negotiated, the parties involved had no idea of the evolution the Gemini flight plans would undergo. Each launch was assumed to have large launch windows; however, as the launch windows evolved from some eight hours to only a few seconds, it was difficult to negotiate the countdown incentives with the contractors. (It is realized that any program will change considerably. However, when the incentive fee portion of a contract is negotiated, a clause should be included allowing for the evolution that a program will most certainly undergo. This, of course, is true only for countdown incentives; delivery and flight performance are basically fixed throughout the program.)

4. PROGRAM DOCUMENTATION

Throughout the Gemini Program heavy emphasis was placed on thorough and complete documentation. The following significant Gemini documents are deemed unique in being beyond the limits of standard program documentation in scope and/or application.

TOR-469 (5126-81)-3, SSVL Exhibit 65-1, Gemini Launch Vehicle Acceptance Specification established and defined the role of the GLV Acceptance Team and outlined procedures to be followed by the Contractor and the VAT at MM-B in connection with formal AF vehicle acceptance.

TOR-169(3126)-19, Gemini Launch Vehicle Pilot Safety Program - AMR described a contractual task for all associate contractors and agencies supporting the GLV program at AMR (later changed to ETR). The prime objective was to assure astronaut safety by achieving the flight readiness of a GLV incorporating the maximum inherent design reliability, assuring adequacy of the GLV MDS, and providing assurance for accomplishing mission objectives.

TOR-169(3126)-6, Complex 19, Demonstration Criteria for the Gemini Launch Vehicle set forth the basic requirements to be satisfied in the demonstration to the Air Force of the readiness of Launch Complex 19, AMR, and its associated AGE to support the GLV Program mission.

TOR-269(4126)-17, Gemini Launch Vehicle Engine Acceptance Requirements defined and established the acceptance procedures to assure that all Gemini engines (both Stages I and II) and applicable spares/ ECP modification lists conformed to the specified Gemini pilot safety standard and were satisfactory for use on the GLV.

424-1020002, Launch Vehicle Acceptance Test Specification. This Martin document was the engineering test specification for combined system tests used for acceptance. The fact that all changes to it had to be approved by SSD/Aerospace provided excellent control of the launch vehicle testing activities.

424-1430002, Launch Vehicle System Tests, VTF/ETR. This Martin document identified checkout and tests performed on the launch vehicle at VTF and ETR in preparation for launch. It was controlled by SSD/Aerospace as was 424-1020002.

SECTION II. A

REFERENCES

1. Gemini Midprogram Conference Report No. NASA Sp 121, National Aeronautics and Spacecraft Center, Houston, Texas, (23-25 February 1966).

SECTION II. A

DEFINITIONS

AEIP	Augmented Engine Improvement Program
AFSC	Air Force Systems Command
AGC	Aerojet General Corporation
AGE	Aerospace Ground Equipment
AMR	Atlantic Missile Range
ECP	Engineering Change Proposal
ESPD	Electronic Systems Products Division
ETR	Eastern Test Range
GATV	Gemini Agena Target Vehicle
GEMSIP	Gemini Stability Improvement Program
GLV	Gemini Launch Vehicle
GPO	Gemini Program Office
ISCD	Interface Specification and Control Document
LMSC	Lockheed Missiles and Space Corporation
MDS	Malfunction Detection System
MM-B	Martin Marietta Baltimore
PERT	Program Evaluation Review Technique
RGO	Radio Guidance Operations
SPO	System Program Office
SSD	Space Systems Division
VAT	Vehicle Acceptance Team

B. PILOT SAFETY PROGRAM DESCRIPTION (METHODS OF OPERATION)

1. PHILOSOPHY AND HISTORY

The concept of "Pilot Safety" as it applies to man-carrying launch vehicles was developed during the Mercury Program from concerns expressed by the Air Force in 1959, as to the safety of the astronauts. At that time, the Atlas weapon system booster was intended to be a launch vehicle for this nation's first man-in-space venture. A program for pilot safety was then proposed by the STL Mercury Program Office and received enthusiastic acceptance by NASA and Air Force. A special NASA, Air Force, Aerospace, Contractor task team was formed to study and implement this approach. [1]

It was apparent that there was a difference between the vehicle's demonstrated reliability and the obvious goal of maximum crew safety, and it would have to be resolved by the following approaches:

1) Maximize vehicle reliability by design change in components or through redundancy in selected systems.
2) Establish a management-supported extra-effort program encompassing all aspects of manufacture and test to insure end product excellence or product integrity.
3) Provide a malfunction detection and abort system fundamentally designed for crew safety.

Although step 3) above would not directly influence vehicle reliability, the procedures and techniques used in developing the Gemini Malfunction Detection System, the slow drift malfunction operation, and abort criteria reflect all of the disciplines developed in steps 1) and 2). This facet of Pilot Safety is discussed in greater depth in Section II. C.

Since in terms of translation of a weapon system into a reliable manned launch vehicle, the Gemini Program faced the same conceptual problems as Mercury, the NASA-DOD agreement[2] directed that a Pilot Safety Plan be established for Gemini similar to that used on Mercury.

Gemini involved many new contractors, but a significant measure of continuity existed, because many of the NASA and Aerospace Corporation personnel were transferred to the new program. This allowed an extensive expansion of ideas and concepts, as the transition took place prior to final contract negotiations with the new supplier. Reliability testing, receiving inspection practices, critical component control, and configuration management are examples of the areas influenced heavily by the Mercury Program.

SSD/Aerospace implemented this requirement through meetings, briefings, and contractual negotiated documents.[3,4,5] Each of the three approaches greatly expanded those used on Mercury and again the fundamentals of the total Pilot Safety Program were very successfully applied during the Gemini Program.

a. Design Improvements

The design improvement area was thoroughly investigated by design reviews and a systematic and continuous failure-mode and effects analysis to identify weaknesses that could be corrected by component improvement or redundancy. The extent of this effort is described in Section II. C.

Examples of design changes identified and adopted for the Gemini launch vehicle are as follows:

1) A redundant flight control and guidance system, which could be automatically or manually commanded to take over and safely complete the launch phase in the event of a primary system failure.
2) A redundant Stage I hydraulic system to support the flight control system.
3) Two completely independent electrical power busses and associated inflight sequencing.
4) Redundant shutdown capability for the Stage II engine.

b. Product Integrity

Product integrity in itself is a very elusive term and proper interpretation is difficult. The Gemini agencies, by working together, were able to contribute ideas and techniques to this aspect of the Pilot Safety Program. This resulted in a management-supported pattern of rigorous technical monitoring and control of all phases of design, development, engineering change, production, inspection, testing, handling, acceptance of hardware at all levels, and launch operations. The foundation for these concepts was a dynamic management effort by all agencies to achieve pride of workmanship and motivation of personnel.

Special detailed mechanisms were established to insure an effective failure analysis and corrective action system, a computerized and highly accurate configuration control system of hardware and test procedures, and an effective hardware acceptance program specifically designed to insure delivery of flight-ready vehicles to the launch site. Many of the techniques and controls of Product Integrity are expanded in this section of the report. Pilot Safety tends to emerge from Gemini as a general attitude of teamwork at all levels with the common objective of finding, discussing, and resolving problems.

The reliability improvements achieved by this massive effort toward product integrity are unfortunately not quantitative, but the confidence achieved through this unified control of detail made possible such milestones as a launch schedule encompassing 10 manned missions in 20 months, the record-breaking 7/6 rendezvous mission, and the 6-week recovery of GLV-2 from an on-the-pad shutdown through analysis, redesign of a major component, retest, and launch.

2. CONFIGURATION CONTROL - REVIEW

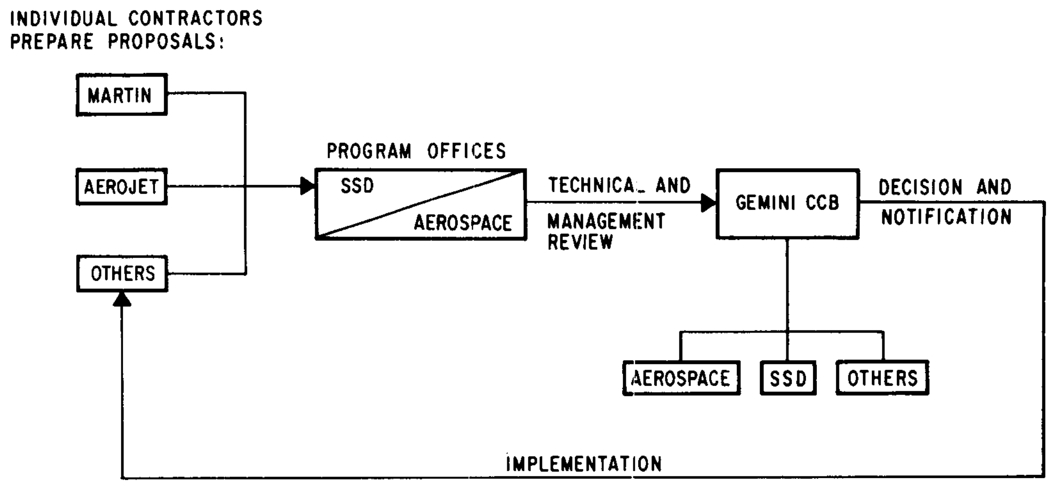
The configuration management and control requirements for the GLV were in general agreement with AFSCM 375-1 and ANA Bulletin Number 445 (see Figure II. B-1). The broad requirements of these documents were applied specifically through the contractual imposition of a series of CCB Instructions, which provided detailed guidelines and standards for contractors to follow in discharging their management requirements. These instructions included detailed work in such areas as:

- Configuration Control Requirements
- Preparation and Use of ECP's
- Facility Change Procedure
- Associate Contractor CCB Support
- Gemini Configuration Index Procedures
- Specification Change Procedures
- Interface Documentation Maintenance

Significant, amplifying attachments to the CCB Instructions included "Serialization," FACI-First Article Configuration Inspection," and "BOI-Break of Inspection for Experiment and Test."

In September 1963 the FACI on Gemini Launch Vehicle No. 1 was held at Martin-Baltimore. This was a milestone in that it represented the first instance in which the first launch vehicle on a given program was baselined prior to delivery. FACI's of AGE and facility items occurred as they became available and acceptance testing was completed.

After establishment of the hardware baseline, all engineering changes were processed through the Gemini Configuration Control Board. Changes prepared by Associate Contractors were forwarded



FOLLOWING CONTRACTOR IMPLEMENTATION, SSD/AEROSPACE VERIFICATION OPERATIONS TAKE PLACE AS A MINIMUM AT TIME OF FACTORY ACCEPTANCE AND AT ETR PRIOR TO COMMITMENT TO LAUNCH

Figure II. B-1. Gemini Vehicle Configuration Management simultaneously to SSD and Aerospace. Following their internal distribution to responsible system engineers, informal coordination between Aerospace and SSD was, in most cases, achieved. The formal Gemini Configuration Control Board was chaired by the SSD Program Director. Members included Air Force representatives from engineering, operations, contracts, and budget; and representative(s) from Aerospace Corporation. NASA and Associate Contractor representatives were also usually in attendance.

When all aspects of a change had been considered in open forum, the chairman made his decision. The intent always was to achieve the best technical product and performance within the dollar, schedule, and other limitations imposed. Approved changes were made as directives to the contractor(s) by appropriate contractual action.

Contractor in-house configuration management techniques permitted an almost real-time capability to identify differences between equipments and insure compatibility with supporting/interfacing equipment, documentation, and spares.

The Martin Company, as integrating launch vehicle contractor, published the Gemini Configuration Index (GCI) monthly. This index, machine produced, included <u>all</u> associate contractor equipment and presented the status in the following five basic classifications:

I. End item configuration index
II. Approved ECP/End item index
III. End item requirements schedule
IVA. End item modification status
IVB. Chassis/Spares modification status.

In addition to the GCI, which was a management document, Martin also produced a number of tab runs, typical examples being:

Airborne vs AGE Compatibility
Vehicle-to-Vehicle Comparison Status
Open Item Status - Both Airborne and AGE
Procedure Changes vs Vehicle Configuration
Individual Vehicle - Complete Identification

The system was "closed loop" and insured that all changes in an item were scheduled and given appropriate status, that open items were continually identifiable, and that nonauthorized work did not exist.

Prior to final acceptance of each engine at Sacramento, and each GLV in Baltimore, representatives from SSD/Aerospace conducted a configuration review of the item to be accepted. In the case of the GLV, the review accorded with the requirements of SSVL Exhibit 65-1, Gemini Launch Vehicle Specification, and insured the accuracy of configuration documentation and specification compliance.

A similar review was made at ETR to verify that actions to be taken after DD-250 and shipment were properly identified and accomplished. This highly accurate procedure did much to enhance total confidence in each vehicle. Complete assurance could be given to Pilot Safety review teams that intended modifications resulting from corrective action or design were incorporated as required.

The entire GLV configuration activity was characterized by a methodical approach, attention to detail, and an uncompromising attitude by management toward making the system one of high usefulness and integrity and, therefore, of wide utilization. It was considerably refined and represented a significant improvement over any previous configuration management system, and it has received a great deal of attention from USAF and NASA officials in terms of setting the example and standards for programs to follow.

The relatively new concept of value engineering was contractually negotiated between the Air Force and each of the major Gemini hardware contractors.

As it was managed on Gemini, the program yielded an average dollar return of 6:1. On a program like Gemini, where production runs were small and astronaut safety was of prime concern, it was apparent that the advantages of Value Engineering as an economy measure were considerably minimized. Many Value Engineering change proposals were rejected after careful consideration by management of the cost savings vs the potential reliability degradation.

As the program matured and hardware reliability was established, it became practical to eliminate several contingency efforts and effect a savings in this manner. As a result, the Value Engineering aspect of the Gemini program was concentrated effectively in the areas of ground equipment, software, and checkout; but the impact on flight hardware was kept to a minimum in the interest of high reliability.

3. RELIABILITY/QUALITY ASSURANCE

a. History and Implementation

The primary objective of the Gemini Launch Vehicle Reliability/Quality Assurance (R/Q. A.) Program was to assure maximum safety of the astronauts. To support this objective, disciplines were established early to (a) ensure realistic and enforceable reliability and quality requirements during the design and qualification test phases of the program; (b) maintain, or improve, the high reliability and quality standards established during the design into the manufacture, production testing, subsystem and system testing, and flight; and (c) develop a closed loop failure analysis/corrective action system second to none.

Each associate contractor had a well-defined R/Q. A. program at the start of his work on the GLV, but it was soon discovered that the disciplines suitable for the needs of other programs were not adequate for a man-rated endeavour. The methods of operation described in this section evolved during the life of the program into a well-integrated, management-oriented function through the combined efforts of the contractors, SSD, and Aerospace.

The original concept of the GLV program was to make use of flight-proven hardware. A modified version of the Air Force Titan II was selected, based on its payload capability, and on the fact that it promised to be an inherently reliable vehicle, because of the use of hypergolic propellants and the simplified mechanical and electrical systems. More than 30 Titan II flights were scheduled for prior to the launch of the first GLV and, as a result of these flights, a high level of confidence could be established in the hardware that would also be used on the GLV. Another bonus was obtained from five of the Titan II flights - a "piggyback program" provided for flying a complete malfunction detection system (the only completely new subsystem), as well as several other Gemini-peculiar components monitored by telemetry but not functionally a part of the Titan II system. All data obtained from these and subsequent Titan II and III flights was carefully analyzed for its effect, if any, on the GLV.

b. Quality Control-Problem Reporting

The success (or failure) of any missile program often depends upon the effectiveness of the Failure Analysis/Corrective Action (F. A. /C. A.) system used by the contractors. The best F. A. /C. A. system will fail if not supported completely by top management. This program was never without such support from NASA, SSD, Aerospace, and each contractor, including financial backing when required. The procedures given describe Martin and Aerojet activities only. Those of General Electric were similar but at a much lower level of effort due to the limited number of components involved. The G. E. R/Q. A. effort was effective and it is not intended to minimize their contribution to the program.

(1) Martin Company Gemini Problem Investigation Status (GPIS), Automatic Reporting System (MARS).

The Martin F. A. /C. A. system is described by both the block diagram in Figure II. B-2 and as follows:

(a) M/B-Generated MARS Tags and MDR's. These tags and Malfunction-Discrepancy Reports were picked up daily by Quality Engineering (Q. E.) from reporting areas, including Quality Test, Reliability Test, Special Engineering Test, Receiving Inspection, Fabrication, Assembly. the Vertical Test Facility and Associate Contractor. While in the pick-up area, reports were reviewed by Q. E. and were closed when: the corrective action applied only to the local area; conclusive and immediate action could be taken to prevent recurrence; and previously-fabricated similar items did not contain the reported deficiency. The cause and corrective action were entered on the report. Closed reports were reviewed with the resident (Baltimore) Aerospace representative for concurrence.

(b) MARS and MDR's Not Closed in Local Area. These were analyzed in detail by Q. E. and Reliability Engineering (R. E.). A preliminary investigation was made of each report to determine the cause of the malfunction and the required corrective action. When investigation proved that either no bona fide discrepancy existed or that a straightforward and simple corrective action was required, the report was closed by noting the cause and corrective action on the MARS or MDR. Again, the closed reports were reviewed with the Aerospace representative for concurrence.

(c) MARS and MDR's Neither Closed in Local Area or After Prefunctory Investigation. These were investigated in depth by Q. E. and R. E., with assistance from other parties and agencies. Q. E. was responsible for leading the investigation of apparent non-design problems and R. E. for apparent design problems. As soon as the need for detailed investigation was apparent, a Gemini Problem Investigation Status (GPIS) was created. Investigation and GPIS control are discussed later. NOTE: If a failed-part analysis was needed to define the condition, a GPIS was initiated. Cases were even documented by GPIS where corrective action was underway or recently completed, provided that the action was not simple, obvious, and straightforward. This ground rule was intended to prevent waiting to open a GPIS in the hopes that the problem could be handled by MARS only.

(d) Martin/Canaveral-Generated MARS. These were also picked up daily from all reporting areas by the M/C Q. E. If a malfunction occurred during other than normal working hours, and the area Q. E. representative felt that an immediate failed-parts analysis or investigation was required, the cognizant system Q. E. was called to initiate appropriate action. As at M/B, each MARS was analyzed in detail and treated in the same manner.

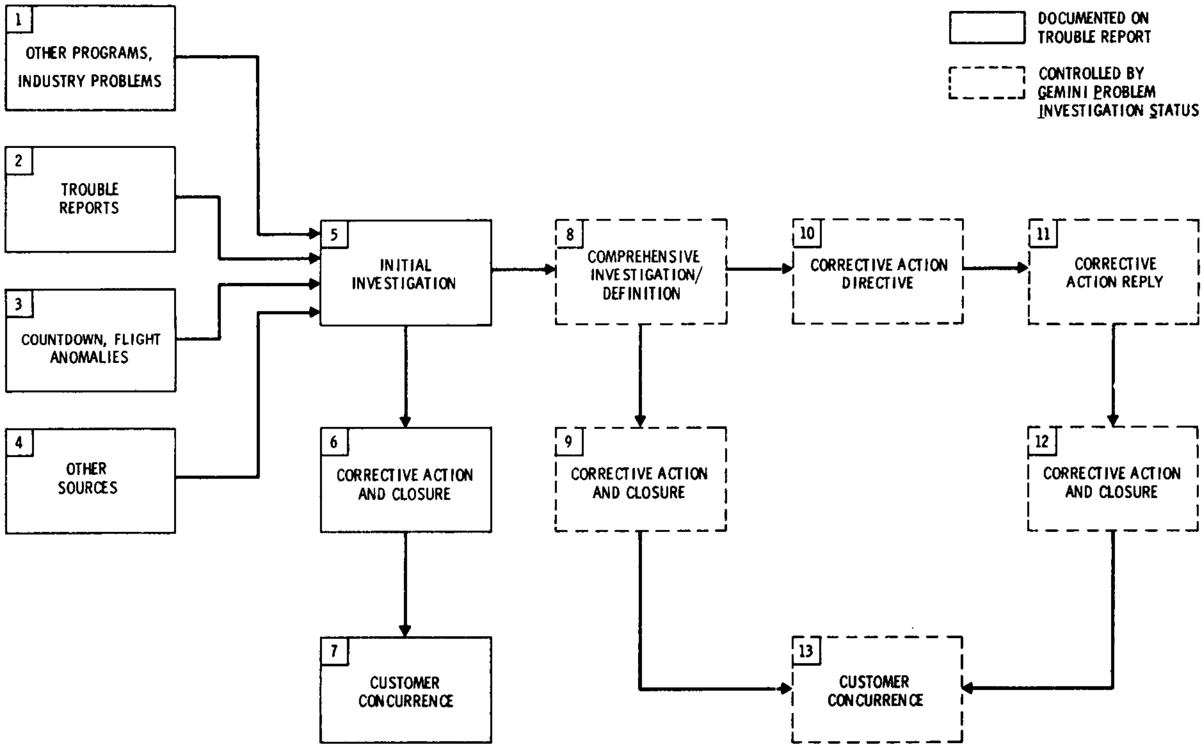


Figure II. B-2. Corrective Action System

(e) Martin/Denver-Generated MARS. These too were picked up daily by the local Q. E. and relayed by telecon to the M/B Q. E., and then a copy was mailed to them. At the time of the phone report, the M/B Q. E. and R. E. evaluated the problem in the same manner as if it had been found at M/B. M/D engineering assisted in the evaluation and investigation directed by M/B.

(f) Vendor-Generated MARS. These or their equivalent were reported to M/B Q. E. by telecon or TWX within 24 hours after the malfunction. They were generated from the point in the production cycle where the vendor declared the equipment ready for Acceptance Testing. The telecon or TWX was followed by a formal report to M/B. As before, each problem was treated as though it had occurred at M/B, except that vendor personnel assisted in the evaluation and investigation when requested by Martin.

(g) Additional Review of Reports. In addition to the review of individual reports previously described, which, of course, included trouble report tabulations to determine failure history, Q. E. and R. E. examined the system and part tabulations indicating the total failure experience, to be sure that the individual report evaluation had not overlooked a failure trend. Malfunctions that recurred, but were not under investigation (and covered by a GPIS), were selected for additional investigation. If, as a result, it appeared that a condition could exist requiring corrective action, a GPIS was created and investigated in the usual manner.

(h) MARS Prepared at M/B and ETR on Associate Contractor Equipment. These were reviewed by Q. E. and R. E. at M/B and M/C for information and status. A separate report was written by the Associate Contractor and given to Martin for information only, when possible. Corrective-action information received from the Associate Contractor was entered in the MARS. A GPIS was automatically opened for an Associate Contractor problem that occurred during a countdown or launch. GPIS's were also opened on Associate Contractor problems when Martin judged that the action taken by the Associate was either inadequate or not moving rapidly enough to a conclusion. In every case, closing a GPIS written against an Associate Contractor problem was by SSD/Aerospace direction and final.

(i) M/C and M/B R. E. and Q. E. Postlaunch Evaluations. R. E. and Q. E. from M/C and M/B evaluated the launch preparations, countdown, and flight performance of each vehicle immediately after launch and selected problems for investigation. The M/C Q. E. had the prime responsibility for evaluation of the launch preparation results, and the M/B R. E. and Q. E. for the countdown and flight data. Evaluation included the investigation of each anomaly or marginal condition reported at the Quick Look meeting and subsequent data reviews. A GPIS was originated for each condition that required extensive investigation or even remotely appeared to have an impact on future tests or flights.

(j) Titan II and III Problems Affecting the GLV. Problems originating on the Titan II or Titan III programs were immediately investigated for impact on the GLV. Problems from other programs reported by SSD, BSD, NASA or Aerospace, were also analyzed in detail for possible effect on Gemini. Although piece-part traceability was not funded or planned in this program, in some instances it was performed by the contractors and proved invaluable when searching for suspect parts.

(k) Special Management Tool. One special management tool was created at M/B to expedite testing at the VTF and yet preserve evidence at the source of the problem. When a malfunction occurred at

VTF and the causes were not obvious, the test conductor immediately stopped the test and convened the Malfunction Disposition Group for the failed system. This group, composed of the engineering test manager or test conductor, a reliability engineer, a quality engineer, and a design group engineer, defined in detail the circumstances at the time of failure, prescribed additional tests in an attempt to isolate the problem prior to hardware removal, and formulated a problem investigation plan. The group was on call 24 hours a day, 365 days a year, and had to meet prior to continuing the testing. The primary value of such a group was to keep good hardware from being pulled from a vehicle in a random fashion, in order to meet schedules.

(l) Number of MARS Issued per GLV. A trend chart of the number of MARS issued per GLV is shown in Figure II. B-3.

(2) Aerojet General Corporation (AGC) Inspection Report (IR)

The failure reporting and corrective action system for the Gemini program was based on the Inspection Report (IR).

The IR was initiated by Aerojet Quality Control inspectors in the receiving, inspection, fabrication, assembly, and test area, to provide a continuing history of all discrepancies found in a particular assembly or during a specific operation.

Prior to acceptance of each engine by the Space Systems Division, the Gemini Reliability and Pilot Safety Group, including contractor and Aerospace personnel, reviewed all component data and appropriate IR's.

(a) Factory Operations. Application of the IR reporting system to the three major areas of interest (receiving inspection, manufacturing, and test area operations) is summarized as follows for factory operations:

1. Receiving Inspection. Discrepancies or failures noted during visual inspection or acceptance testing of vendor components were recorded in an IR by a Quality Control inspector. Inspection supervision and/or Quality Engineering made an initial action decision to rework, or to submit to the Material Review Board (MRB). Parts requiring MRB decision needed disposition approval by signature of Quality Control, Design Engineering, and AFQA representatives. After Quality Control disposition, the buyer of the Material Division determined whether parts would be repaired or reworked at the AGC plant, or returned to vendor.
2. Manufacturing Operations. Discrepancies discovered during the fabrication or assembly process that could not be remedied without using formal procedure were documented in an IR by a Q. C. inspector. Inspection supervision and/or Quality Engineering made an initial action decision to rework, reject, or to submit to Material Review Board/Engineering Review Board (MRB/ERB). The actions referred to MRB required AFQC concurrence and certification.
3. Test Area. Discrepancies discovered during visual inspection, functional checkout, and testing that could not be resolved without formal procedures, were documented in an IR by a Quality Control inspector. A Quality Control Engineer then reviewed the items to determine initial disposition. Those actions referred to MRB required AFQC concurrence with disposition, cause, and corrective action by signature.

(b) Field Operations. Discrepancies discovered during visual inspection, functional checkout, or static testing at field sites were documented by Martin-Marietta Corporation (MMC) QC personnel on Martin Automatic Reporting System (MARS) forms. Verified discrepancies were transferred to an IR Master (referencing the initial MMC failure report) by Aeroject Field QC personnel. An Aerojet QC representative participated in the Martin MRB actions regarding Aerojet hardware. The results of these actions were documented by an Aerojet representative on IR masters, and the cognizant AFQA representative at the field site signed the master to verify hardware disposition.

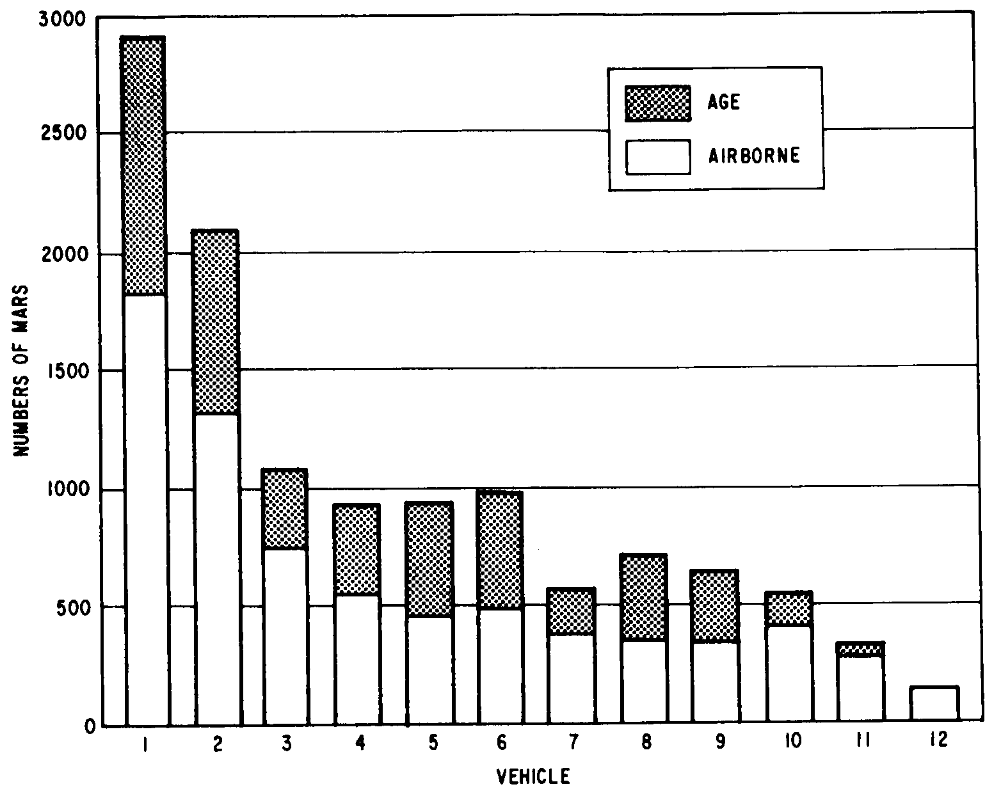


Figure II. B-3. MARS Issued Per Vehicles

The IR's initiated in the field were forwarded to AGC/Sacramento for completion of the disposition, cause, and corrective-action portions of the form, or for information purposes in cases where the cause and corrective action had been entered on the form at the field site, or where no cause and corrective action were required. Telephone communications and TWX were also used to provide immediate information on items requiring action by AGC/Sacramento. This information alerted personnel involved to collect the necessary data to support the failure analysis prior to the receipt of discrepant hardware. Application of the IR system to the field operations follows.

1. Routing of IR's After Initial Actions. Following the initiation of the IR and the actions described, the IR was either routed to Q. E. or, with the hardware, to the Discrepancy Analysis Area (DAA) or Air Force Bonded Stores (AFBS). In either case, Q. E. had over-all responsibility for completion of the IR. Exception to the routing described was made in some cases with respect to discrepancies reported on vendor-supplied hardware.

2. Procedures for Completing IR's When the Discrepancy Could Be Readily Identified As a Quality Control or Engineering Problem. When an IR was initiated for a problem readily identified as one of Q. C. requiring no failure analysis, Q. E. completed the action on the IR without referring to Design Engineering or Reliability.

   When a problem was readily identified by Q. E. as an engineering problem not requiring failure analyis, the IR was referred to Gemini Design Engineering for determination of cause and corrective action. Upon its receipt, Design Engineering was responsible for coordinating with Gemini Reliability for any case history or other reliability information that might contribute toward solution of the problem. Following determination of cause and corrective action, Design Engineering coordinated, as required, with Reliability, with reference to impact of the corrective action on the reliability of the engine system. Design Engineering then forwarded the completed action to Q. E. for closeout of the IR and submittal to Aerospace for approval.

3. Procedures for Completing IR's When the Discrepancy Could Not Be Readily Identified As a Quality Control or Engineering Problem. When the type of discrepancy could not be readily identified as a Q. C. or engineering problem, an investigation of the discrepant hardware was initiated in the DAA (in-house discrepancies) or AFBS (field discrepancies).

   With reference to in-house discrepancies not requiring failure analysis, Q. E. proceeded unilaterally with the determination of cause and corrective action. If, during the initial Q. E. investigation, it was determined that an engineering problem existed and no failure analysis was required, Design Engineering was notified. Design Engineering then determined cause and corrective action. In cases where a failure analysis was determined to be in order, the failure analysis proceeded as described in Section II. B. 3. d, which follows.

c. Problem Investigation/Failure Analysis System

(1) Martin Company

A detailed investigation was conducted on all problems, unless, as discussed in Section II. B. 3. a, proper corrective action could be immediately implemented or was not required. As stated before, when a detailed investigation was initiated, a GPIS was created. This was a means of formally logging each investigation; it did not necessarily mean that a problem or condition that required correction existed, but only that an investigation had begun, was under corrective action cognizance, and that the results would be fully documented. It was just as important to log investigations that resulted in no corrective action, as those requiring remedial action. Creation of a GPIS also resulted in the creation of a problem folder, where it and other pertinent documents were stored for convenience during the investigation, for follow-up of corrective action, and for rapid retrieval if required for future use in the program.

A GPIS could be initiated by a request from anyone on the Gemini Program, each was assigned a number and coded to identify the subsystem affected. R. E. and Q. E. at M/C and M/B were directly responsible for conducting the investigation, but could request assistance from any department or agency needed to complete the task.

The purpose of the problem investigation was to define the cause of the problem; its extent, such as all units or only certain lots or pieces; the effect of the condition on production operations, test, countdown, and flight; the cost and schedule impact of "living with the problem," or of corrective action; and to define the action that must be taken to prevent recurrence of the condition. The methods used to investigate and the personnel involved were determined by the nature of the problem, but the personnel were usually drawn from R. E., Q. E., and the design engineers at M/B and M/C. The quality engineers were responsible for leading the investigations of apparent non-design problems, and the reliability engineers for those of apparent design problems. The results of the investigation were documented or referenced in the GPIS as the investigation proceeded and all appropriate documents were placed in the problem folder. This had the specific advantage of preventing notes being kept on nonrecoverable pieces of paper in engineers' desks.

The first objective of an investigation was to find the cause of the problem. When this could be done by other means, a failed-parts analysis was not made. Such an analysis was conducted when required and authorized by M/B Q.E., whether at M/B, M/C, or at the vendor's facility. The decision as to where to conduct a failed-parts analysis depended on the urgency (including the spares balance), facilities, past failed-parts analysis performance by the vendor, and past experience with the hardware. It was usually desirable to perform an analysis at the vendor's plant, but with a witness present from M/B who was responsible for assuring its adequacy and feeding this information to all interested parties. A failed-parts analysis of a vendor item performed at M/B or M/C was witnessed by a representative of the vendor, when possible. The basic program philosophy was that every failed-parts analysis should be continued until the cause was determined or, if this were not possible, until all reasonable efforts had been made. If the cause could not be found, the problem was thoroughly reviewed by Q.E., R.E., and the appropriate design group engineers before terminating an analysis. If necessary, additional units of the hardware were obtained and further tests and analysis conducted on these units.

Frequently, problems occurred during the last stages of test at the launch site and time did not permit the normal step-by-step procedure. In that case, the failed parts were returned to the nearest facility best equipped to do the failure analysis. The parts were flown to the facility and, in many cases, by Air Force jets to expedite the analysis. Every means available (overtime, weekends, etc.) was used to complete the analysis, establish the mode and cause of failure, and then evaluate the flight hazard with respect to this known condition. Occasionally, it was possible to take short-term corrective action on the vehicle installed on the launch pad. That might have been a one-time inspection of that vehicle, an abbreviated test of one particular condition, or that the probability of occurrence was so low that the risk was considered acceptable. When final action was impossible for a particular vehicle, the problem was brought to the attention of management at the level where the decision to launch could be made.

A GPIS trend chart is shown in Figure II-B-4.

(2) Aerojet General Corporation

(a) In-house IR Customer Interface. IR's written in Receiving Inspection and Manufacturing on components reviewed by the Pilot Safety Review Team in accordance with Aerospace GLV Engine Acceptance

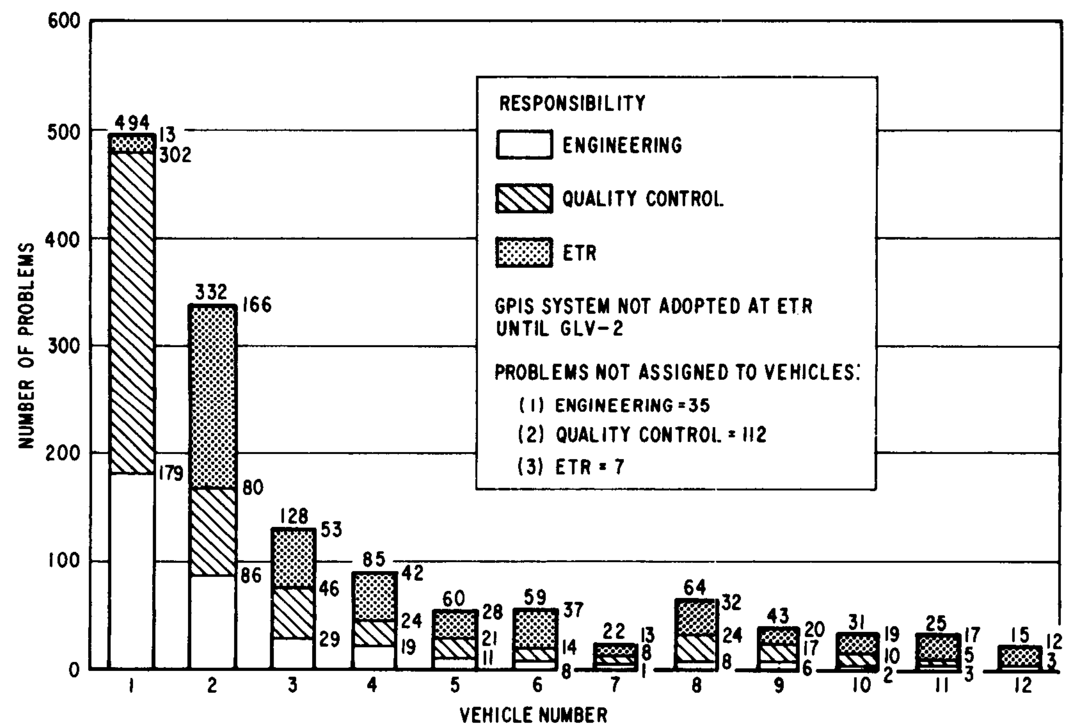


Figure II. B-4. Gemini Problems Investigation Status - New Problems by Vehicle

Requirements[6] were presented to Aerospace Corporation by AGC Pilot Safety Group. This was done prior to engine assembly of the respective components against which the IR's were written. Aerospace signed off each component package to verify that all IR's therein were satisfactory and closed, except as noted on the certification signoff sheet. In cases of exceptions, the unresolved discrepancies were documented on an AGC IR form and submitted to AGC Q.C. by Air Force Quality Assurance (AFQA) for resolution.

IR's generated during engine buildup and test were presented by QC to the Sacramento Aerospace Office on an incremental basis. IR's requiring MRB action were processed through AFQA before presentation to Aerospace. The signature of the Aerospace representative on the IR constituted final closeout. Any unresolved discrepancies noted by Aerospace or AFQA were documented on an AGC IR form and submitted by AFQA to AGC QC for resolution; upon resolution, an IR was resubmitted to the Sacramento Aerospace Office for signature approval.

Prior to the formal acceptance-panel meetings conducted for each engine, there was a pre-panel discussion of IR's by representatives of AGC and Aerospace. QE represented AGC and was supported by Reliability and Design Engineering with respect to engineering problems.

The engine log book contained an IR summary sheet, which listed the IR number and status for all those written against the engine assembly and components that remain on the engine. The IR status was certified by signature of AGC, AFQA, and Aerospace on the summary sheet.

(b) Field IR Customer Interface. The cognizant AFQA representative at the field site signed the IR to verify hardware disposition and, in those cases where cause and corrective action were completed in the field, to indicate concurrence with the corrective-action statements. IR masters were then returned to AGC, Sacramento. Copies of IR's originating at M/B were distributed by AGC Field QC to Aerospace, AFQA, and M/B, and to Aerospace and AFQA at Sacramento. Distribution of ETR IR's included Aerospace, Patrick Test Site Office (PTSO), Martin at ETR, and Aerospace and AFQA at Sacramento.

With reference to IR's where cause and corrective action had been entered in the field, QE-Sacramento reviewed the field action and submitted the closed IR to Aerospace and AFQA at Sacramento for signature approval.

IR's returned to Sacramento for determination of cause and corrective action were completed and presented with necessary backup documentation by QE to the Sacramento Aerospace Office for review. In the case of IR's originating at ETR, Aerospace coordinated with Aerospace/6555th ATW prior to approval. Signature approval of the close-out action was made on the IR master by Aerospace and AFQA at Sacramento. Any deficiency noted in the action was documented by AFQA on an AGC IR form and presented to QE-Sacramento for action as required.

Quality Control at Sacramento forwarded completed copies of the IR's with Aerospace/AFQA approval signatures and the necessary backup documentation to the cognizant AGC field sites; these went to representatives of the customer, for the initial field distribution. Backup documentation consisted of reports such as failure analysis and quality engineering, but did not include blueprints, engineering test directives, shop orders, and similar documents, unless specifically required.

AGC field QC presented closed-out ETR IR's to the Pilot Safety Working Team at ETR for review. If the close-out action was not acceptable to the team, the 6555th/ATW/Aerospace-ETR notified the AFQA/Sacramento of additional action required. AFQA/Sacramento documented the deficiency and requirements on an AGC IR form and presented it to QE for action and reprocessing of the IR questioned by the ATW.

To reduce response time for IR close-out, pertinent information from IR's initiated after flight minus 30 days (F-30 days) was transmitted by AGC Field QC at ETR to Field QC at Sacramento by TWX. Immediately following determination of cause and corrective action, Field QC/Sacramento transmitted this information to Field QC/ETR by TWX. Distribution was made expeditiously by Field QC/ETR. These actions did not negate the requirement for processing the IR in the normal manner.

All IR's initiated after F-30 days were reviewed and acted upon from the standpoint of flight safety only. Final close-out action not affecting flight safety was taken in a timely manner, but with lower priority.

Problem areas identified by the Martin Company that possibly required AGC action were brought to the attention of Aerospace/SSD by means of a Martin Gemini Problem Investigation Status (GPIS) form, with a copy to the field AGC representative. AGC action was in accordance with SSD direction, and AGC response was to the customer.

(c) Problems Requiring Long-range Corrective Action. AGC's Gemini reliability group maintained a system for identifying, resolving, and providing status for problems of a continuing nature, or that required an extended period of time to resolve. These problems were designated as Gemini reliability problems. The group was responsible for determining the problems to be included in the system from one or more of the following sources:

- Transfer of a problem from one or more IR's when extended investigation or customer approval of an ECP is required.
- Selection from existing problem folders.
- Selection from problems experienced in the Titan II or III programs.
- Identification from historical records of malfunctions or discrepancies.
- Evaluation of recommendations from other departments.

In cases where a problem was transferred from an IR to the Gemini reliability problem category, the IR was normally closed by a statement in the corrective-action block of the IR to the effect that action had been transferred to the Gemini reliability problem list (designated by the appropriate problem number), and the IR was closed. QE or Design Engineering could request the transfer of a problem from an IR to the reliability problem file.

Each Gemini reliability problem was assigned an identifying number, and a folder containing pertinent information was maintained for each. A Gemini reliability problem report describing the problem and indicating its corrective-action status was maintained for each. In addition to providing over-all status, this form indicated the status of implementation of interim and final corrective action with respect to each Gemini engine. Copies of all active Gemini reliability problem reports were included in the Monthly Summary and Summary Data Report, which was distributed to SSD/Aerospace and within AGC. Copies of newly-opened Gemini reliability problem reports were provided on an incremental basis to the Sacramento Aerospace office as well as to Gemini Engineering and QE.

The Gemini reliability group assigned action to other departments for the resolution of problems, assisted in problem solutions, and provided follow-up to assure timely completion of necessary actions. When the necessary information had been developed to resolve a problem, the group closed out the problem by updating the reliability problem report form. The completed report summarized the problem, described the corrective action taken to prevent recurrence, and referenced pertinent backup documentation. Following close-out of a problem, the file of information pertinent to it was maintained intact to provide a historical record for future reference.

(d) Reliability Monitoring. The Quality Control Division sent copies of IR's to the Gemini reliability group as soon as practicable after completion. The Gemini reliability team reviewed them for reliability implications and provided Gemini Design Engineering with copies, as appropriate. In any case where the review of an IR's malfunction or failure history, or other information, indicated a possible reliability degradation of the rocket engine, the Gemini reliability group investigated the potential problem area. If this investigation indicated that the close-out action of an IR had not provided adequate resolution of the discrepancy, the reliability group informed QE of the unresolved discrepancies and requested that the IR be reopened. QE then reopened the IR and completed the corrective action. If QE did not agree with reliability group on the requirement for reopening the IR, the AGC Gemini Program Manager determined the action to be taken.

(e) Reliability-Problem Customer Interface. The status of the Gemini reliability problems requiring long-range corrective action was included in the Monthly Summary and Summary Data Report distributed to the customer. Copies of newly-activated Gemini Reliability Problem Reports were provided to the Aerospace Sacramento office on an incremental basis. When a Gemini reliability problem had been closed, the Reliability Problem Report on that problem was submitted to the Aerospace Sacramento office for approval of the close-out action. Aerospace indicated approval by signature on the report. Any exceptions to the action taken were documented by Aerospace on an AGC IR, which was presented to the Gemini reliability group for additional action, as required. The group was the customer interface regarding all aspects of the Gemini Reliability Problem System.

(f) Failure Analysis. A failure analysis was made with respect to each IR written against a Gemini critical component and also for IR's written against non-critical components as appropriate. Responsibilities and procedures for the failure analyses were as follows:

- For in-house discrepancies, the requirements for failure analysis were determined by QE. When this involved a part or assembly that had been determined (through initial QE investigation or otherwise) to have failed or malfunctioned during any functional acceptance test, the failure analysis plan was prepared jointly by QE, Gemini Design Engineering, and the Gemini reliability group. The planning document was signed by representatives of these organizations. Design Engineering and Reliability designated the steps in the failure analysis where they would participate or review results. The failure analysis was made by QE with Design Engineering and Reliability participating to the extent specified in the failure analysis plan, and representatives of these organizations signed the IR master to indicate concurrence with the statements of cause and corrective action entered thereon.
- For field discrepancies, the requirements for failure analysis were determined jointly by QE, Gemini Design Engineering, and Gemini reliability group. Procedures for failure-analysis planning, failure analysis, and signoff of documentation with reference to field discrepancies were identical to those for investigation of in-house discrepancies involving hardware that failed or malfunctioned during functional acceptance test.
- For discrepancies involving vendor-supplied items, failure analysis was sometimes made by AGC in the Sacramento plant on vendor-supplied items that had failed or malfunctioned. In these cases, the procedures were the same as those described in the preceding paragraph. In certain cases, the failure analysis was made by the vendor and representatives of QE, Design Engineering, or Reliability participated, as appropriate.

(3) Examples of Failure Analysis During Gemini Program

The following brief examples of the Gemini failure analysis in action demonstrate the importance of identifying the physics of failure. This is the critical step that allows management to weigh the risk the problem presents against the economy of a positive fix.

(a) Relay Problem. In September 1964 a relay in the switchover module of the autopilot adapter assembly operated intermittently when energized. The relay was removed and opened for failure analysis. Solder particles were discovered inside the contact chamber, one adhering to the normally closed movable contact of one set of contacts, while other particles were imbedded on the plunger shift. It was concluded that the solder particles intermittently jammed between the plunger and the plunger guide and caused the malfunction.

A survey team, composed of engineers from Aerospace/SSD and Martin Company, conducted a rigorous audit at the vendor's facility and found (in part) that:

- There appeared to be a complete lack of quality and contamination control during assembly, with no special precautions taken to eliminate foreign particles from the relays during assembly
- The relays were not assembled in a clean room
- No extra precautions were taken to ensure that solder and flux were not introduced during final sealing operations.

The same survey team then visited other relay manufacturers, compared and evaluated their facilities and products, and then recommended that the relays in question be replaced with those manufactured by another vendor. This was accomplished on both airborne equipment and AGE (after the search of all relays manufactured by this vendor revealed that the AGE also contained his product).

Months later in the program it was found that the Gemini Agena Target Vehicle had relays produced by this vendor and had also experienced some trouble with them. With the overwhelming evidence collected on the GLV problem, it was a simple matter for Aerospace to convince Lockheed to remove this vendor from the approved list for the program.

(b) Transistor Problem. During a routine functional preproduction monitoring test (PMT) of an autopilot adapter assembly, intermittent output readings were noted. After the electrical connectors were cleaned, the intermittent condition persisted. The adapter package was returned to the clean room where the trouble was isolated in a module assembly. During further troubleshooting the erratic readings changed to a "no-output" condition.

The above incident and several others reported against the same component for improper module circuit voltages and failures were attributed to piece-part performance in the circuit design. The failed modules were put into failure analysis and tests were developed and made to determine the cause and physics of failure. Nondestructive tests were performed initially and then carefully-planned destructive piece-part examination followed.

In all instances, the module failures were attributed to those of U. S. Army Type 2N328A transistors. The modes of failures were the following:

- Residual acid within the transistor acted on the junction areas, resulting in either emitter or collector detachment. Due to the acid attack, the condition was progressive and could not be determined until the failure occurred. This was the principal mode of failure.
- An emitter detachment was the result of an overetched and, therefore, weakened junction.

The conclusion was that the basic cause of failure was due to inadequate rinse of the acid etching solution during transistor manufacture. The deterioration was gradual, and the effects thereof were eventually precipitated by the heat of current actuation.

Identical transistors were removed from stock and inspected. After careful removal of the caps, it was found that two units had already failed with open leads and that all units had various stages of acid deterioration.

Resolution of the problem to support program requirements was authorized as follows:

- All modules containing the 2N328A transistors were scrapped. These were replaced by new modules containing only JAN type 2N328A transistors. For GLV-2 (the next vehicle flown), the autopilot adapter assembly had an assembly dash number change, which indicated the transistor change.
- For GLV-3 and follow-on, the autopilot adapter assembly had still another dash number change, because specially procured JAN type transistors with more stringent receiving inspection and test were incorporated.

d. Vendor Control

Control of suppliers who furnished hardware for the GLV was accomplished by analyzing the suppliers' qualifications; by explicit contractual control, including acceptance testing, special Gemini critical-component handling, and special motivational programs for vendor personnel; and by special quality and reliability audits.

Each associate contractor used approximately the same method in obtaining assurance, initially, that a vendor could produce hardware that would meet the Gemini reliability requirements. A prospective vendor's qualifications were thoroughly reviewed through an evaluation and rating program; he was analyzed primarily on his capability to deliver consistently a high-quality product on the production schedule required for the GLV program. Each associate contractor used approximately the same rigid contractual controls on his vendors. The following list contains only a small sample of the requirements that were imposed on each vendor. They included stipulations that the vendor shall provide for contractor approval of:

- Quality assurance system
- Design evaluation data
- Drawings—assembly, schematic, detail, etc.
- Weight prediction and final report
- Stress analysis
- Qualification test procedures and reports
- Acceptance test procedures and reports
- Reliability—prediction, test demonstration, etc.
- Functional test reports
- X-ray quality requirements
- Shelf-life materials identification
- Weld procedures
- Critical part time/cycle reporting
- Process approval
- Inspectors—vendor, customer, government
- Traceability
- Spares parts list

Vendors that survived the initial screening and were agreeable to all the special Gemini requirements imposed on them were the best available at that time and, for the most part, performed excellently.

During the early production phase, major suppliers were invited to attend meetings conducted by representatives from the Gemini Program Management Office, Procurement, Receiving Inspection, Quality, and Reliability. The prime objective of these meetings was to generate a program of worker motivation within the vendor's facility.

The techniques and procedures used by AGC were made authoritative by their incorporation in Corporate or Sacramento Plant Standard Practice Manuals, Quality Control Instructions, Procurement Procedures, and Gemini Project Directives.

The reliability program at M/B included reliability audits of critical-component vendors on a semiannual basis. This was done to supplement normal quality surveillance and to ensure vendor adherence to definite reliability[7], procurement[8], and critical component[9] specifications.

The intent of the audits was not to direct suppliers in the running of their operations, but to advise them how existing controls and records could be modified to provide the performance required for Gemini. The audits fulfilled the following specific needs:

- Assurance of adherence to engineering requirements
- General vendor liaison
- More accurate interpretation of what was expected by the contractor
- Better understanding of vendor capabilities, consideration, and problems
- Expected corrective action of vendors
- General vendor motivation

Motivation was achieved by: the presence of an audit function; presentation of program material, such as models, posters, pennants, and films; appraisal of strong and weak points; and, in rare cases, visits with astronauts. The appraisal of strong and weak points was very effective since vendors were generally interested in being part of the Gemini image and desirous of taking advantage of disciplines associated with man-rating for attracting future business.

The audits were of considerable value to the program. Literally dozens of discrepancies, even though generally of minor impact, were noted and corrected. The audits were comprehensive, being subjected to the checklist shown in Figure II. B-5. Its use, together with a small selected audit team, afforded a common basis for comparing vendors and determining what could be expected from them (particularly small vendors) in the manufacture and test of man-rated hardware. It additionally yielded maximum information without placing an undue burden on the vendor.

An audit report was prepared after each inspection, with copies provided to the quality source control, Procurement, and Design Engineering. The suppliers were apprised of their evaluation by Quality after preparation of audit reports, although preliminary critiques were generally made immediately after the audits. In almost all cases, three to five working days of notice were given vendors prior to an audit. This was a short enough time to preclude any major clean-up, and still allow an accurate observation of the conditions at the vendor's facility. Nearly all of the vendors were cooperative and welcomed the audits.

In summary, the audit program was highly successful, resulted in numerous corrective actions, enhanced the intelligence of all parties, was economical, and promoted better hardware and supporting documentation. The results are graphically shown on Figure II. B-6.

e. Reliability/Quality Assurance Recommendations

It is difficult to assess the total contribution of the Reliability/Quality Assurance programs on the GLV, but following is a summary of the procedures found effective and that are recommended for future programs — manned or unmanned:

COMPANY ______________

CATEGORY

COMMENT - If discrepancy noted.

TRACEABILITY

1. To Critical Part level ______
2. Below Critical Part level ______
3. Incoming parts or lots ______

RELIABILITY

1. Specific personnel ______
2. Failure-flagging system ______
3. Data collection system ______
4. Analysis and prediction capability ______
5. Records of discrepancies below Critical Part level ______

FAILURE ANALYSIS

1. Analyses documented ______
2. In-house capability ______
3. Implementation capability ______
4. Effort and records below the TWRT level ______

ENGINEERING

1. Specific personnel ______
2. Drawing change approval ______
3. Unsolicited design changes ______
4. Packaging design ______
5. Latest Martin drawing ______

DESIGN REVIEW

1. Scheduled ______
2. Documented ______

PRODUCT IMPROVEMENT

1. Not considered ______
2. Submitted ______
3. Incorporated without Martin approval ______

CRITICAL PARTS

1. Storage areas ______
2. Handling considerations ______
3. Decals, posters and tagging ______
4. Personnel indoctrination ______
5. Knowledge of part's GLV function ______
6. Critical Parts list approved ______

MANUFACTURING

1. Receiving inspection ______
2. Area access control ______
3. Clean room certification and/or operation ______
4. Personnel certification ______
5. Reference drawing or model identification ______
6. Discrepancy reporting ______
7. Cleanliness ______
8. Test tool calibration ______
9. Assembly inspections ______
10. Subassembly design changes ______

TEST

1. Test tool calibration schedule ______
2. Acceptance test procedure approved by Martin ______
3. Calibration procedures ______
4. Calibration facility ambient environment ______

RE-CYCLING

1. Operating time records ______
2. Re-work records ______

CONTROLS

1. MB-1053, 1054 and 1055 ______
2. In-house procedures ______
3. In-house processes ______
4. Supplier specification drawings ______
5. In-house and suppliers communication ______
6. Calibration records ______
7. Martin source inspector ______
8. Supplier surveys ______
9. Supplier certification ______
10. Martin communication ______

Figure II. B-5. Vendor Audit Check List

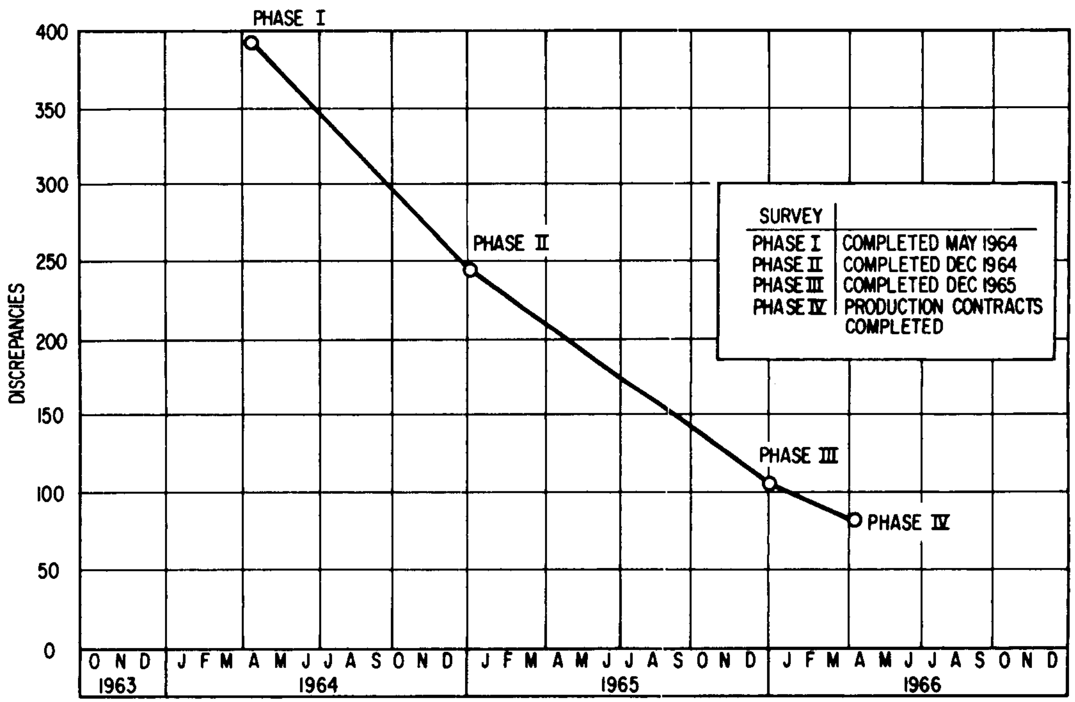


Figure II. B-6. Reliability Vendor Audits - Total Number of Descrepancies Observed/Phase

1) Establish a closed loop corrective action system. This forces a review and positive close-out of each problem.

2) Assign corrective-action engineers to each individual system. This provides an expert for failure analysis and corrective action.

3) Create a strong field C.A. organization to handle problems and C.A. at the source.

4) Maintain close coordination between Quality and Reliability Engineering to provide complete coverage of all aspects of any problem area.

5) Develop a Malfunction Disposition Group to prevent the loss of failure data. This is an excellent method of attacking problems and for preventing further degradation of vehicle systems through additional testing.

6) Insist on a trend data review. This forces a review of questionable system parameters before failure.

7) Reject hardware that exhibits a transient or unexplained malfunction. Most causes of transient malfunctions can be found if properly analyzed. Use of hardware for which the failure cause has not been determined invites substandard failure analysis and permits hardware having an unknown failure probability to fly.

8) If possible, analyze all available flight test information from other programs (i.e., Titan II), if similar hardware is used.

9) Develop a GPIS system. The GPIS was one of the most effective management tools used on the GLV program. It has been called a "super" MARS, IR, trouble report, etc.; but it was a means of sorting through the thousands of problems to ascertain the significant ones and of presenting them concisely and explicitly to all levels of management for action. Top management does not take (or have) the time to review all of the problems on a program; but with this system all of the significant problems were brought to its attention, especially since the customer was in the closure approval loop. A total of 1413 GPIS were written during the program, 1010 airborne and 403 AGE. This is exclusive of between 300 and 400 more that were called "Problem MARS" and opened at ETR prior to the start of the GPIS system at ETR (GLV-2).

10) Establish a method of vendor control that is explicit contractually. The instructions imposed initially upon a vendor can often mean the difference between "acceptable" and "hi-rel" hardware, with very little additional cost (if any).

11) Conduct reliability audits at the facilities of vendors of critical components. Impress on them that this is being done to help them produce better hardware.

12) Maintain interest and a desire for success at all levels by keeping all personnel informed of the national importance of what they were doing. Perhaps a manned program has an advantage in this respect that other programs do not have. Personal motivation, however, on this program was thought to be directly proportional to management enthusiasm and contact - and the same would probably be true for all programs.

f. Surveillance — Titan Family Flights

An extremely important consideration in the total effort to assure man-rating of the Gemini launch vehicle was the impact of the Titan II and III flight test programs. One of the fundamental concepts of the launch vehicle pilot safety program on Gemini was a rigorous review of flight test results and an assessment of the problems experienced. Aerospace conducted these reviews as a continuing effort throughout the Gemini program.

The early history of the Titan II program shows ten flight failures out of twenty. These were basic design and operational weaknesses, which have been correctly identified and eliminated in both the operational vehicle and the Gemini launch system. Repetitive gross deficiencies such as Stage II gas generator contamination and POGO have been highlighted and extensive corrective action taken to eliminate these problems from the Gemini launch vehicles.

Although the follow-on flight tests, in terms of numbers and percentages, were much more successful than the preceding twenty, the data yielded indications of incipient weaknesses, such as anomalous propulsion operation in terms of slower-than-normal thrust buildup, slight out-of-tolerance steady-state performance, and other similar problems. These were continually assessed for Gemini and corrective action was taken. This process of constant refinement of the Gemini vehicle provided increments of reliability improvement.

In December 1965, with the launches of GT-6A and GT-7 marking the halfway point of the Gemini Program, a decision was made by SSD/Management to review and evaluate again, prior to the launch of GT-8, all past flight anomalies and problems pertaining to the entire family of Titan vehicles. A brief description of this effort is contained in Section II. E.

Total surveillance of Titan family flights included the following efforts in chronological sequence:

1) Titan II – R & D Flights – 32 Total. Attending quick-look and post-flight reviews at STL/San Bernardino and/or Martin/Denver, and coordination of anomalies affecting the Gemini Launch Vehicle with these contractor personnel.

2) Titan II – DASO (Demonstration and Shakedown Operational) Flights - Transition Between R & D and Operational – 5 Total. Attendance at post-flight reviews held by STL and BSD with SAC participation; coordination with these agencies.

3) Titan II – Operational Flights – 23 Total. Followed launches at VAFB and attended post-flight reviews when they were held at VAFB and Martin/Denver; coordinated with these agencies, as well as OOAMA on post-flight data and anomalies.

4) Titan III – 12 Total. Kept abreast of all Titan III launches and anomalies; coordinated same with Aerospace and Martin/Denver and attended post-flight reviews.

5) All Titan II Flight Test Data and Reports. These were analyzed for impact on the Gemini Launch Vehicle.

4. THE ASSETS PROGRAM

Two-thirds of the way through the program, the Martin Company established an Assets Team whose job it was to assure an adequate supply of components and parts to carry through the testing at VTF and ETR and the launches of GLV 9 through 12. Similar efforts were undertaken by the Aerojet General Corporation to insure an adequate logistics posture from the standpoint of the rocket engines. The Martin team was chaired by Logistics and composed of representatives from Engineering, Quality, Contracts, Planning, and the launchsite personnel. Appropriate authority and priority for effective control were established. While economy constrained oversupply and program completion prohibited undersupply, the overriding team guideline was that there be no compromise in hardware reliability. The task was divided into the following two parts:

Inventory and Reliability Status Determination

Procurement and Assets Control

Status determination required an inventory of all parts, components, and assemblies by serial number and an immediate identification of critical and marginal items from a logistic and a flight worthiness standpoint. Engineering and Quality evaluated or reevaluated specific component histories when required, and performed or updated statistical analyses on component failures for prediction of asset requirements and for comparison with established quantities.

Procurement and assets control involved consideration of changing lead-time situations as Gemini-peculiar parts assemblies phased out or where vendors had ceased production of piece parts and components that had become obsolete. The Assets Team therefore established status and decision-making controls for continuous use, intended for quick and effective reaction to any potential shortage, or to any new or changing failure mode that would affect the statistical failure predictions mentioned.

Initially, four items were identified for priority action: the PCM Encoder, TARS, Signal Conditioners, and Destruct Initiators. Actions included reorders, rework, failure analyses and firm agreements on turnaround time and maintenance of reliability standards. In some cases these agreements were made through Air Force and Martin visits to the supplier's facility. It is felt that the emphasis given to a normal logistics function by the formation of a special team and tying it closely to reliability is an effective tool in maintaining product integrity. Prelaunch testing component replacements on GLV-12 and the flawless performance of these later vehicles bear this out.

5. ACCEPTANCE

A major portion of the effort in the Pilot Safety Program assuring maximum product integrity, was the means used by SSD/Aerospace Corporation to accept contractor hardware for the program. End product acceptance versus incremental acceptance of hardware has remained controversial throughout the Gemini program. Although the engines, tanks, and finished vehicle were formally accepted separately, this should not be misinterpreted as sequential or incremental acceptance, since these articles represented final end items manufactured at separate facilities. From a contractor's standpoint, incremental buyoff by the customer agencies as the product completes functional milestones in test and manufacture within the same facility is desirable. This minimizes the risk of identifying high impact problems late in the production cycle and thus aids in scheduling. SSD/Aerospace insisted that in the case of vehicle production, surveillance by local customer representation and proper technical and quality control by the contractor should minimize any schedule effects of end product acceptance and that a formal concentrated review by a proficient acceptance team would have a competitive influence on all personnel to do a good job. They also insisted that this acceptance be given top management attention by the contractor. End product acceptance provides the contractor with flexibility in manufacture and test such that top level management is more likely to be involved in these activities and thus more interested in hardware acceptance.

Incremental acceptance, however, limits this flexibility and eventually hardware acceptance may have a tendency to be managed at lower levels with lesser impact, professional incentive, and management attention to problems. As it happened, both approaches were taken to some extent. Regulations and criteria for hardware acceptance agreed upon by the contractors and SSD/Aerospace were highly disciplined and contractually binding. The Gemini engines justified a form of incremental acceptance, since repeat of the static firing cycles would have considerable impact on delivery. The engines were very carefully examined before the static firings to minimize the necessity of a retest.

The tanks and launch vehicles were accepted as end products quite successfully. The Vehicle Acceptance Team found it extremely difficult to find significant problems, since the contractor took this formal acceptance exercise as a challenge.

The acceptance activity began with the propellant tank assembly performed by Martin/Denver. After satisfactory integrity testing, a formal review and rollout inspection was made by SSD/Aerospace and, upon official acceptance, the tanks were shipped to Martin's assembly plant in Baltimore for splicing and final assembly.

The Stage I and II rocket engine systems were manufactured and tested at Aerojet General Corporation's plant in Sacramento, California. These systems also underwent official review and acceptance by SSD/Aerospace and, if satisfactory, were airlifted to Baltimore.

The launch vehicle guidance equipment was manufactured and tested by General Electric in Syracuse, New York. Although formal acceptance was not conducted manufacturing was reviewed very carefully (II. B-30).

Gemini launch vehicle final assembly checkout and acceptance was conducted at Martin's Space Systems Division, Middle River, Maryland.

a. <u>Launch Vehicle Tanks</u>

The tank rollout inspections were held at M/D in two phases. During Phase I, the production history, test history, and all weld defects and the associated stress analyses were reviewed, with the appropriate M/D personnel, by M/B engineers and quality assurance personnel. There was a requirement for a written stress analysis showing positive margins of safety for all weld defect areas. This analysis was checked and, where necessary, rewritten by M/B structures engineers. During the Phase II portion of the rollout, the same material, in corrected form, was reviewed by SSD and Aerospace personnel. In addition to the above documentation, pre- and post-hydro x-rays of all defects were reviewed by Aerospace metallurgists to verify the nature and disposition of all weld defects.[10]

All peculiar problem areas, such as patches on the tanks, lock-bolt repairs of leaks, and any not specifically covered in the weld acceptance criteria were reviewed in advance of the rollout by M/B, SSD, AFQC, and Aerospace, and a resolution was agreed upon.

b. <u>Engines</u>

The design principle for Stage I and II Titan II engines was reliability through simplicity. Hypergolic, noncryogenic propellants are used that can be stored separately in the ballistic missile and do not require special checkout procedures during the countdown. Both the Stage I and II engines were qualified according to the applicable Air Force specifications and exhibits. The Gemini launch vehicle requirements, however, necessitated numerous engine-design and operational changes in the engines of the Titan II Weapon System. These evolved from crew safety requirements, and were for astronaut warning in case of incipient failures, and increased reliability of component operation.

The Gemini Pilot Safety Program was initiated as a tool to attain more stringent controls in the fabrication, handling, testing, and accepting of components and engines for the Gemini launch vehicle. Its main objective was the attainment of the quality and reliability necessary to ensure the success of a man-rated launch vehicle propulsion system.

Engine acceptance procedures, followed by personnel on the Gemini Program, are documented in the applicable Aerospace[11] and Aerojet[6] reports. The acceptance procedures were designed to ensure that all Gemini engines and applicable spares/ECP modification kits conformed to the specified Gemini pilot safety standards and were satisfactory for use on the Gemini launch vehicle. Engine acceptance required the following sequence of events:

- A detailed subsystem/component review was conducted by AGC and by the Pilot Safety Review Team of SSD/Aerospace/AFPRO prior to the start of engine build. All critical components were required to have the approval of the Review Team prior to initiation of engine assembly.
- Prior to acceptance firing of the assembled engine, a detailed system review was conducted by the Pilot Safety Review Team in the course of which the team examined the final engine assembly records and decided upon the acceptability of the engine for acceptance testing.
- A pre-acceptance test meeting was held. At this time, the Review Team formally announced the acceptability of the engine for testing or detailed the action required to put it in an acceptable condition.
- The acceptance test series was designed to be accomplished in a minimum of two steps, in accordance with the requirements set forth in the Engine Acceptance Test Specification, in an attempt to demonstrate the acceptability of the engine as defined in the Model Specification.
- Following completion of the acceptance test firing, performance and post-test hardware reviews were conducted to ensure that the engine and its components still met the Gemini standards.
- A formal acceptance meeting was held whenever an engine was deemed acceptable to the Review Team.

c. Launch Vehicle

Complete assembly of each stage was accomplished with the stage in a horizontal position. After final assembly, a series of integrity tests were made consisting of airborne wiring-continuity tests, engine-vehicle interface leakage tests, and hydraulic system functional tests. Upon their completion and after the application of a specified paint pattern, the vehicle was erected and mated into the vertical test cell at Martin/Baltimore.

The whole vehicle was then subjected to a detailed visual inspection and, following this, to an extensive series of subsystem functional verification tests. This test phase ended with the successful completion of a combined system acceptance test (CSAT).

The CSAT trial runs were informal until the history of testing indicated that the formal CSAT could be efficiently performed immediately following the subsystem functional verification test. The purpose of the trial runs was to verify the capability of the launch vehicle and the AGE systems to operate together as required during the CSAT. During these tests, there was troubleshooting and minor adjustments and modifications were made in preparation for the formal acceptance test. Systems that had been modified were reverified, using applicable subsystem functional verification test procedures to achieve readiness for the CSAT.

The final phase of vertical testing consisted of the CSAT, which embodied one countdown and two simulated flights to demonstrate the primary and secondary guidance-flight control-hydraulics system and other redundant features of the launch vehicle. During the first run, all subsystems performed as in a normal flight. During the second run, a malfunction was simulated in order to switch to the redundant system. The CSAT was a formal demonstration of the capabilities of all launch-vehicle and AGE systems to function together as required at the vertical test fixture.

The demonstration of total vehicle system operation represented by the CSAT exercise established a production milestone, which then became the foundation of all subsequent prelaunch testing. It was upon successful completion of this test and the Vehicle Acceptance Team review of the test data that permission was given by the VAT to de-erect the launch vehicle.

(1) Vehicle Acceptance Team

The role of the Gemini Launch Vehicle Acceptance Team (VAT) was originally outlined in TOR-169(3126)-16[12], covering GLV-1 through GLV-4, and was superseded by SSVL Exhibit 65-14 for GLV-5 and subsequent. These documents established and outlined the acceptance procedures to be followed by the contractor and the VAT at the contractor's facilities to assure, prior to formal vehicle acceptance by the Air Force, that:

All applicable factory acceptance tests and inspection procedures were completed satisfactorily.

The GLV configuration was complete and up to date, as authorized.

The GLV documentation, as required was complete.

Contractor and AFPRO review teams reviewed test data and made hardware inspections independent of the VAT throughout the acceptance program. Test data, hardware documentation, and final vehicle inspections were accomplished in sequence by (1) the Contractor, (2) the AFPRO team, and (3) the VAT, to insure that each established its position independently. The Certificate of Completion was initiated for all significant phases of acceptance and indicated the documentation and data reviewed, including open items requiring completion prior to final acceptance. The completed Certificates of Completion became a part of the Launch Vehicle History and were signed by the Contractor and the local AFPRO. A consolidated customer position was presented to the Contractor at the Vehicle Acceptance Meeting. Discrepancy reports were prepared at each step. Associate Contractors (AGC and GE) were invited to participate during Contractor reviews, as necessary.

The VAT was composed of system engineering specialists from Aerospace, SSD, and AFPRO who monitored their particular systems throughout the assembly and testing period in Baltimore and were on call for support of prelaunch activities at ETR. Their responsibilities are identified in Figure II. B-7, which illustrates the management, technical, and support assignments used for GLV-12 and is representative of all previous VAT's.

Technical evaluations were made by members of VAT in the general categories of production, reliability/quality assurance, test, and hardware status of the specific specialty area in the Systems and Support Groups.

(2) VAT Activities

The VAT was allotted five days for an independent data and documentation evaluation after the Air Force Quality Control Office certified satisfactory completion of Combined Systems Acceptance Test (CSAT) procedures.

To fulfill the requirements established for VAT, six major areas were established to evaluate the acceptability of the launch vehicle (the seventh item (g) is a description of the ROPA system). They were as follows:

a) Subsystem Data Review. All the system engineering specialists reviewed their system's performance in all aspects of vehicle testing (subsystem, retesting, and CSAT). During review of the test data, every action of every system was perused in detail. Anomalies were annotated with satisfactory reasoning, or the components involved were replaced and the test rerun. Upon conclusion of system tests and during data review, the vehicle was held in a bonded condition. No access to the vehicle either by customer or by contractor personnel was possible without the written permission of the Air Force plant representative. This control was necessary to make sure that if an investigative retest was required the vehicle would be in the identical configuration as it was when the test data was generated.

b) Configuration Check. VAT members conducted detailed reviews of the GLV configuration (1) prior to power-on, (2) just before Combined Systems Acceptance Test, and (3) just prior to acceptance recommendation by VAT.

c) Critical Component Data Packages. VAT specialists reviewed the data packages of the installed critical components in their respective systems for discrepancies and any out of specification performance.

A requirement was established, beginning with GLV-2, that on vehicle acceptance, complete data packages would be made available containing pertinent data on all critical components installed on the launch vehicle. The data package contents for each critical component were specified in Aerospace Corporation document 1972.7-48, "Data Package Requirements for Martin GLV Critical Components," (2 July 1964.) Data packages were maintained for a total of 178 individual critical components covering 56 types. Contents of the data packages varied with the individual types of components, but in general consisted of the following:

- Check sheet of package contents
- Chronological history and accrued time
- Specified data covering all tests on the component
- Specified data covering failures and corrective action.

d) Gemini Problem Investigation Status (GPIS) Reviews. As a part of the VAT activities, a GPIS review was made under the chairmanship of the Aerospace Gemini Reliability Group. The GPIS for each subsystem was reviewed by a special group composed of one reliability engineer, one quality engineer, and one subsystem engineer from Martin Baltimore; one quality engineer from Martin ETR; one reliability engineer and one subsystem engineer

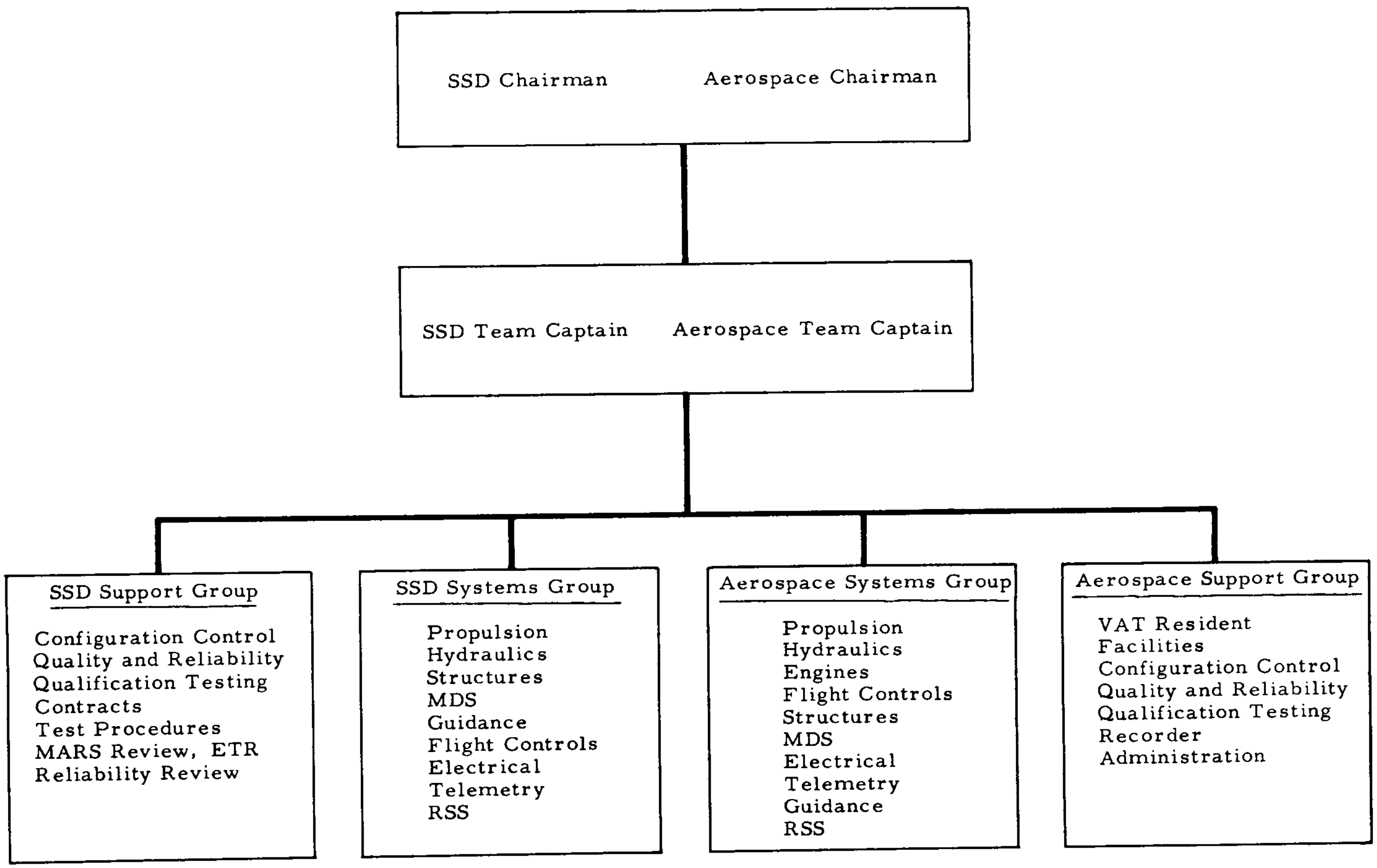


Figure II. B-7. Vehicle Acceptance Team

from SSD; one subsystem engineer and one reliability engineer from the 6555th ATW; one reliability engineer from Aerospace, ETR; and one subsystem engineer from Aerospace, Los Angeles.

All GPIS's, open and closed, were presented for review by Martin Co. The status of the open GPIS's was thoroughly reviewed and pressure brought to bear on the organization(s) responsible for completing whatever was required to close the problem. This open status could include an incomplete failure analysis, the need for formal documentation of a failure analysis, a pending ECP, an interface problem, etc. The closed GPIS's were completely evaluated to determine if the action taken was adequate to assure no recurrence of the problem; or, at most, if no action was possible that the top management of each group represented and NASA had been made aware of the risks involved. (It should be noted that a GPIS was formally closed only when a unanimous decision was reached by the reviewing team, with one exception - a GPIS could be closed by the unilateral action of the customer when it was believed that the cost to correct the problem far outweighed the risk involved. This action was taken less than ten times during the life of the program). Only when a problem was closed by mutual consent, or in rare cases by the customer, was it officially removed from the books.

An average of 75 GPIS's were reviewed at vehicle acceptance.

e) Documentation Review. Data and documentation requirements for VAT reviews were outlined in detail in the reference contractual documents. This material enabled VAT members to make a thorough evaluation of the vehicle in the following major categories:

- Launch Vehicle Acceptance Certification and History
- Contractual Specifications
- Test Specifications and Procedures
- Test Data and Reports
- Quality Certification
- Trouble Reporting and Corrective Action
- Support Documentation

f) Physical Inspection. VAT members made a physical inspection of the total vehicle during the acceptance period. Aided by flashlights and other inspection equipment (mirrors, magnifying glasses, etc.), they determined the general condition of the vehicle. Inspections covered the following areas:

- Tanks, domes, ducts, and tubing assemblies
- Installation of components
- Integrity of electrical wire and plug installations
- Structural assemblies and verification of proper torques
- Engine installations
- Adequate clearances.

The discrepancies found were written up by Air Force Quality Control personnel, who had the contractor accomplish correction. Over the span of the program there was an average of 10 to 15 discrepancies per vehicle.

Time allotted to VAT for this activity averaged three hours. This, together with the fact that certain portions of the vehicle were not accessible, limited the detail scope of the inspection. However, inspection of those portions of the vehicle that were accessible and viewable, represented a good baseline of the vehicle's condition for the final walkdown inspection conducted prior to flight by members of management.

The program benefited from this exercise, in that the people performing the work and inspections accepted the VAT inspections as a challenge. Manufacturing and Quality personnel made check sheets of earlier VAT discrepancies and ran extra inspections prior to VAT of a new vehicle. It was apparent that the contractor personnel were taking pride in the performance of their work, because as the program progressed it became more and more difficult to discover discrepancies.

In conjunction with the above inspections, onboard component audits were made and verified against the configuration tab-run for the particular vehicle. Component part numbers and serial numbers were taken by the VAT members and the contractor was asked to verify these units as legal parts according to his published contracting documents. An average of 35 items was checked in this exercise. Results of this check indicated that controls and documentation were effective since no improper components were present.

g) Resolution of Problem Area (ROPA). On finding a discrepancy or problem area in any of the vehicle acceptance activities that could not be readily resolved with their contractor counterparts, VAT members would initiate a Resolution of Problem Area (ROPA) form.

The issuance of a ROPA was a flag to the contractor and VAT management indicating that an item was in question. The ROPA served its purpose well and was a dynamic tool for resolving problems affecting the acceptability of the vehicle.

The ROPA system was beneficial to the program in two ways:

> It provided an orderly and documentary means of resolving a discrepancy; and contractor personnel backed up the presentation of possibly questionable items with additional data and documentation to substantiate their technical position just to preclude the issuance of a ROPA.

(3) Contractor Critique and Post-VAT Activity

Recommendation for GLV acceptance rested with the VAT chairman. If the VAT recommended rejection, the chairman formally advised the AFPRO and the Contractor of the reasons for rejection and requirements for hardware replacement, further tests, and/or documentation. If additional testing was required, a plan to correct the deficiency was proposed by the Contractor and approved by the AFPRO and VAT Chairman after evaluation. The contractor was advised of any contingency tasks (open work items) to be performed prior to shipment and formal acceptance on Form DD 250 by AFPRO. These tasks, together with itemized lists of hardware and documentation to be shipped to ETR with the vehicle, were included in the Acceptance Summary Report with Form DD 250. A Certificate of Completion and attached Acceptance Summary Report listed all contingency tasks (open work items) and the recovery plan prior to the official signing of Form DD 250, which was signed and approved upon successful completion of packing for shipment.

d. General Electric MOD IIIG Task Team

In May 1962, a design review of the MOD IIIG transponder system revealed serious deficiencies in the manufacture and test processes. Further reviews during the next twelve months emphasized this point and the need for a requalification program because of the large number of significant design changes incorporated in the two years since these units were flight qualified. As a stopgap measure, acceptance vibration testing was raised from a level of 3-1/2 to 6-1/2g. However, the results of 6-1/2g testing pointed up a serious quality control problem, in that the MOD IIIG was designed for a 6g rms random flight environment, whereas the expected Gemini environment was 12g rms random.

An AF/Aerospace/GE NASA Gemini Task Team was established in June 1963 to review the Gemini MOD IIIG missileborne equipment quality and reliability, to review GE's plans and recommendations for improvements, and to establish joint action plans for further improving quality and reliability. The task team existed for one year and agreed upon a number of actions in the areas of management, design, manufacturing, quality control, and reliability.

(1) Management

To give top attention to the Gemini program, the GE Gemini program managers were realigned to report directly to the General Manager of the Radio Guidance Operation (RGO).

(2) Design

Since the expected flight environment was double that for which the equipment had been designed, the GE recommendation to incorporate isolated baseplates in the units was accepted. The success of this design can best be measured by the completion of qualification testing, including 20g rms random vibration, without failure, and the success of the GLV-1 flight.

(3) Manufacturing

A producibility design review was held in conjunction with Design, Reliability, and Quality Control organizations. Specific changes were incorporated in drawings, sequence assembly instructions, and tooling to eliminate possible shorts and to improve producibility, solderability, cleaning, and spot-potting.

(4) Quality Control

A Corrective Action and Repair (CARE) procedure was instituted to assure managerial review of corrective action on all potted module failures on the Gemini program. In addition, the repair action was reviewed to make sure that the problem had been eliminated.

All parts still in stock and not assembled as of the beginning of June 1963 and parts received after that date for the Gemini program were 100 percent electrically tested. An extensive review was made of all major procurement items and rf components to determine what environment could cause failures and if prior testing could eliminate the problem. As a result of the review, the Gemini klystrons were tested for temperature, the power monitors and local oscillators for vibration and temperature, and the rate beacon rf assemblies for vibration.

(5) Reliability

Inputs to the Gemini Task Team from Reliability were derived from the results of a Qualification and Evaluation test program on laminated (as distinct from isolated) baseplate equipment. The Phase I changes incorporated in the Gemini units built later were a result of this program and included several piece-part changes, new bracketry, and encapsulation to prevent wire breakage.

A "Top Ten" reliability problem was generated to highlight particular problems and assign responsibility for corrective action.

(6) Conclusions

These efforts resulted in the production of 8 isolated baseplate systems (-100 contract) with Phase I changes incorporated; completion of qualification testing of the isolated baseplate system without failure; and reduction of failures in acceptance vibration testing on Gemini by a factor of 7 on the last 11 systems.

6. SPECIAL GEMINI PROGRAM REVIEWS

As a means of insuring hardware integrity, the management of the major organizations used Special Gemini Program Review groups or teams. These teams would evaluate their readiness to support major tests, hardware acceptances, technical reviews, and launches. Listed by organization are the significant review teams with brief descriptions of their activities.

a. Aerojet General Corporation

(1) Gemini Design Review

The decision to use the Titan II engines as the propulsion system for the Gemini launch vehicle instigated a specific requirement for an engineering technical review whereby all aspects of the design disclosures were evaluated by engineers of Design, Development, Quality, and Reliability. This detailed review of each critical component and assembly was made to determine the changes required to attain the degree of confidence necessary to certify the engine as man-rated. Special design changes were evaluated, demonstrated, and qualified for incorporation into the engine system to add to the safety of the astronauts.

(2) Pilot Safety Team Technical Review

To ensure fabrication compliance with design and Aerojet General specifications, the Pilot Safety Team reviewed in technical detail each engine system, serialized spare, ECP kit, and end item. Discrepancy reports on each critical part were evaluated and the rework for correcting the deficiency was carefully analyzed for adequate action. Components with performance data deviating from nominal were required to have special inspections to verify the integrity of the hardware. In many instances, components were disapproved for use on a Gemini engine if an anomaly could not be satisfactorily explained or verified through a retest.

The engine assembly was reviewed before acceptance testing; and special handling and inspections took place in the test area.

After engine acceptance, the Pilot Safety Team made a detailed review of all rework, inspections, and assembly operations to insure that the engine system delivered to the Air Force was acceptable for flight.

(3) Reliability Problem Reviews

Reliability, Quality Control, Design, and Project Engineers conducted detailed technical reviews of each Gemini reliability problem. Failures and discrepancies that occurred on any of the Titan engines were analyzed and evaluated as a possible Gemini reliability problem. Three specific categories were established: (1) problems relating to test failures or having a significant impact on the engine system; (2) problems of a continuing nature; and (3) problems requiring extended periods to resolve. The investigation, analysis, and evaluation of the corrective action of each problem was reviewed continually until resolved. In many instances, the corrective action involved the processing of Engineering Change Proposals or Engineering Test Directives. This required additional technical reviews to qualify, demonstrate, and validate the ECP or ETD, as applicable.

(4) Gemini Preflight Review Committee

Engineering established a flight operations function to maintain continuous monitoring of the engines in the field to insure that any work or problems were reviewed by the cognizant design engineers. One important function of Gemini flight operations was the control and management of the preflight of each engine system scheduled for the launch of a Gemini vehicle. A documentary file was established for each GLV, which encompassed the history, configuration, and actions required to attain the status of launch readiness and certification for flight.

The authority for insuring that each Gemini engine was acceptable for launch was vested in the Preflight Review Committee. Its purpose was to: (1) maintain continuous monitoring of problems on all Titan engines; (2) take corrective action to increase pilot safety, reliability, and quality assurance; (3) predict the probability of success if no further corrective action were possible; and (4) release the engine propulsion system for flight. The primary responsibility for the preflight function was assigned to the flight operations of Engineering and Reliability. All functional support managers participated actively in decisions involving corrective action and certification. A minimum of three preflight reviews were held for each GLV system.

(5) Launch Readiness Review

The status of each engine system scheduled for the launch of a Gemini vehicle was reviewed for top management after the final Preflight Review Committee meeting. The technical details of all reliability problems, the failure analyses of all current Titan family problems, and a history of the engine systems were presented by Reliability and Engineering managers. Each problem area was discussed as to cause, corrective action, and impact on the vehicle scheduled for flight. Human error problems connected with Titan family engines were analyzed by Reliability for: the most probable success ratio, precautions taken on the Gemini engine, verification inspections, pilot safety documentation, and for their impact on successful flight. The history of flight anomalies was reviewed for the most probable causes and the actions required for the vehicle scheduled for launch.

The Launch Readiness Review was concluded by the Program Manager's verbal confirmation that the cognizant design engineering and functional managers considered the engine systems acceptable for flight from the technical analysis.

b. Martin/Baltimore

(1) Qualification Test Task Team

A team concept was employed for the basic organization of the GLV qualification test program. The team included selected specialists from engineering (test, design, and reliability), procurement, quality, manufacturing, and planning. Engineering had the basic responsibility with over-all direction coming from the Program Director. The task team directed the efforts of the individual subsystem group that had the responsibility for implementing and expediting all qualification test requirements for a group of components.

Coordination of vendor activities was the responsibility of individual team members. Preliminary documentation, expediting, and on-the-spot approval at the vendor test site were accomplished, and technical assistance was provided for the setup and start of testing. The objectives were to help the vendor perform on a technical level comparable to the Martin system, and to expedite all aspects of the program. In all cases, quality and reliability were not to be compromised.

(2) Assets Task Team

An Assets Task Team was created after the GLV-8 flight with the responsibility for insuring that serviceable program assets, both production and spare units, together with all required paperwork, were available when needed to accomplish the production, testing, and launch of GLV-9 through -12.

A description of the team's members, purpose, and control is given in Section II. B. 4.

(3) Malfunction Disposition Group

In the event of an equipment malfunction at the Vertical Test Facility (VTF), if the failure was associated with airborne and AGE hardware and its causes were not obvious, the test conductor convened a Malfunction Disposition Group. This group did the following:

- Made an on-the-spot investigation and established the failure circumstances.
- Planned additional VTF tests necessary to diagnose or isolate the problem.
- Decided on disposition of the failed components.
- Assigned follow-up responsibility for failure analysis.

A similar procedure was followed at Cape Kennedy

The VTF Malfunctions Disposition Group held over 200 meetings and greatly aided in producing a highly reliable launch vehicle.

(4) Data Analysis Team

The Data Analysis Team reviewed items to insure component and system integrity for each GLV. Any discrepancies were resolved through review meetings attended by the design, reliability, quality, and test personnel. The team reviewed the following:

- Data packages on the GLV critical components to confirm that the components installed on the launch vehicle were within allowable specification limits. (The data packages contained the acceptance data and any other significant test data associated with that component prior to its installation in the GLV.)
- All test procedure accomplishments and the data obtained during the VTF and ETR subsystem and system tests.
- All telemetry data obtained from the various GLV tests, including the launch countdown and flight, and analyzed for problems that might indicate a component anomaly.

(5) Engineering Inspection Team

Prior to the Combined Systems Acceptance Test in the Vertical Test Facility, there was an engineering inspection of each GLV participated in by a representative of each airborne system. The inspection was intended to reveal engineering items of installation and fabrication that were marginal or unsatisfactory, and any comments pertaining to the inspection were entered in a log book. At the conclusion of the inspection, discussions were held with the quality representatives and the necessary corrective action was taken.

(6) Technical Reviews

Reviews of airborne (A/B) hardware were made early in the design phases, and on a continuing basis throughout the program. The responsibility for conducting the design reviews was assigned to the airborne systems engineering groups and personnel from various areas of the program supported them. The purpose of the reviews was to discuss in depth the capability of the hardware to satisfy program requirements, to examine the design for factors of safety, fail-safe conditions, possible deficiencies and modes of failure, and to evaluate design improvements. In addition and after the GLV baseline configuration, all changes in the AGE and airborne hardware and circuits were reviewed at the technical director level.

The frequency of, and customer participation in, design reviews are best illustrated by the following example: The airborne circuit change (in GLV-9 and follow-on) consisting of using malfunction-detection thrust-chamber pressure switches in lieu of thrust-chamber pressure switches for the launch sequence and staging functions, was reviewed by Martin on five occasions over a period of twelve months; and with NASA, Aerospace, and Aerojet on three other occasions over a period of nine months. Thus, eight design reviews were held in a twelve-month period from initial design to launch vehicle test at VTF.

A total of 127 documented design reviews for all subsystems were conducted during the program.

(7) Launch Integrity Team

Three to four weeks before each launch, Martin convened a Launch Integrity Team (LIT) meeting for the purpose of insuring launch vehicle readiness. Each subsystem of the vehicle was reviewed by the LIT team, which was composed of top-level technical managers from the various Martin divisions (Baltimore, Denver and Cape Kennedy). Representatives from Aerojet General, General Electric, Burroughs, and McDonnell assisted in this review.

At the meetings, Martin design engineers and associate or subcontractor representatives were required to make presentations on the following:

- Action items from the previous LIT review
- Changes resulting from the previous flight
- Differences from the previous flight
- History of system and test results
- Summary of problems
- System flight test objectives and associated instrumentation
- Action required before launch.
- System checkout requirements (holds and shutdowns).

The LIT meeting was not confined to hardware topics; performance and analysis areas were also reviewed. A LIT list of items and recommendations was generated for action prior to launch. The customer did not attend these meetings, but did receive a copy of the LIT recommendations.

(8) Launch Support Team

Several days prior to each GLV launch, a team of specialists from Martin-Baltimore went to Cape Kennedy to assist the launch team in handling expeditiously any matters that might have had an effect on the GLV and associated AGE integrity.

c. General Electric

(1) Factory CARE Board

The purpose of the CARE (Corrective Action and Repair Evaluation) Board was to establish the procedures and controls for corrective action, repair, or scrap of defective potted assemblies. All potted modules that failed and all failures occurring in missileborne units after assembly were presented to the CARE Board. It consisted of a unit manager from Quality Control engineering, Reliability engineering, Design engineering, and Manufacturing engineering.

(2) Depot CARE Board

A Corrective Action and Repair Evaluation program similar to the one at the factory was conducted at the depot. The principal distinction was that in addition to factory representatives from

Reliability and Design, this board included representatives from the Depot Repair and Maintenance Section. Its administrator was appointed by, and reported administratively to, the manager of the depot, while receiving technical direction in the performance of his duties from the Reliability Program Manager.

The Depot CARE Board program was applied to all missileborne equipment and to subassembly failures that occurred at field sites or internally at the depot.

(3) Gemini Flight Readiness Board (GFRB)

The GFRB was organized to provide the Gemini Program Manager with a recommendation on the Go-No-Go capability of the Radio Guidance System prior to a scheduled Gemini launch.

This board was chaired by the Gemini Program Manager and composed of representatives from Reliability, Systems and Design Engineering, Aerospace Equipment Engineering, and System Test and Evaluation. It was responsible for the status review of missileborne, AGE, and ground equipment assigned to support the scheduled Gemini launch, of systems functions, and of related equipment and other functions.

(4) Missileborne Equipment Review

The Missileborne Equipment Review was an assigned activity of the Depot CARE Board. The board evaluated the history of each piece of missileborne equipment and recommended a classification for it (e. g., flightworthy, non-flight qualified, or recommend for ground test only (GTO)).

The goals of this activity were to review the history and status of equipment before assigning it to a missile for launch, and to detect units with potential unresolved problems; the latter were denied flightworthy classification until the problems were resolved. This activity provided the customer and General Electric with high preflight confidence that the equipment would meet the objective of 100-percent mission success.

d. <u>Burroughs</u>

The procedures established in the Burroughs Mercury contract for pilot safety were continued for Gemini. A key concept contributing to the success of the program was the writing of all tests to resemble the standard countdown, thus implying repeated launch rehearsal. Another important factor was the following mandatory documentation:

- Maintenance records in which were noted the equipment identification, the specified intervals between maintenance checks, and the dates and results of the tests.
- Failure reports containing the results of all critical component failure analyses performed in Paoli. Failure trends or the need of mass replacement were discerned and acted upon. Weekly failure reports were presented at the Mod III Working Group Meetings at ETC, and a bound report was published monthly.

At the end of the Gemini Program in November 1966, Burroughs was able to report that there had been no critical failure since 26 October 1964.

e. Aerospace Corporation

(1) Launch Readiness Reviews (Scrubdowns)

Prior to the launch of a GLV, the Aerospace Program Office held a review meeting with the GLV associate contractors. Its main purpose was to ensure that each contractor and the Aerospace Corporation mutually agreed on the technical status of the vehicle and the identification of existing or potential problems that required corrective action prior to launch.

The agenda for these reviews covered contractor status reports on the following items:

- Interface testing
- Testing status of system(s)
- Open documentation
- Time-sensitive and critical components
- Countdown procedures
- Open tasks remaining
- Problems and corrective actions

(2) Technical Direction (T. D.) Meetings

During the early phases of the program (1962 to 1964), a number of Propulsion system oriented T. D. meetings were held with Aerospace/SSD/AGC, and M/B participating. They were not T. D. meetings in the strict sense, but rather were utilized as status review and coordination sessions between the agencies mentioned. At the beginning of the program, there were many problems that required detailed interface coordination between engine and the vehicle manufacture. These problems could be categorized as both hardware- and paper-type (specifications, performance, operating characteristics, etc).

By holding meetings bimonthly, it was possible to reduce the amount of time required for coordination, as the one briefing was given to all interested parties and followed by discussion and recommendations for problem resolution. These meetings also provided a convenient place for documentation and the assignment of action items and their answers/resolutions. Further, the meetings insured that all parties were cognizant in writing of propulsion problems, thereby reducing the possibility of any specific item "falling through the crack" and not receiving adequate review and coordination.

When problems requiring this type of coordination were reduced to a number that could be handled on an individual basis, the meetings were discontinued.

(3) Engine Design Review

In August 1963, SSD/NASA/Aerospace, and AGC agreed that it was desirable to make a detailed design and engineering review of the Gemini engine systems to determine their adequacy for use on a manned program. This review was held in two phases. Phase I basically covered hardware and performance, and looking for any areas/components that were weak or marginal and would require either redesign or special testing/handling. Phase II covered specifications and procedures, including a review of basic engineering drawings and field procedures (ETD's) utlized for component replacement and final checkout of launch.

The review took place at both AGC and Aerospace and continued at a substantial level of effort through November. The review, its results, and recommendations have been documented by both Aerospace and Aerojet.[13, 14] There were no items considered to be constraints on the launch that were not under active evaluation; however, there were many areas determined to be weak and to require additional evaluation and possible component redesign. A number of the items were included under the AEIP.

With respect to specification and procedure adequacy, the design review effort revealed that AGC did not have adequate procedures and specifications to sufficiently control manufacturing, inspection, and testing of Gemini-class hardware. It was mutually agreed that corrective action in the documentation area be handled as a portion of the AGC Critical Parts Program. It can be concluded that both phases of the basic design review did uncover areas of weakness that required active resolution to assure the adequacy of the engines for manned application.

f. <u>All Agencies</u>

(1) Launch Vehicle Technical Reviews

Approximately three weeks before the flight of a Gemini Launch Vehicle, a presentation was given to the NASA Gemini Program Office. It was presented by SSD/Aerospace personnel and covered performance estimates, vehicle test status, problems, and corrective actions. An update of this review meeting was conducted at Cape Kennedy five to seven days prior to launch.

(2) Design Review Board

Prior to the first manned launch utilizing a Gemini Launch Vehicle, a Design Certification Review was conducted by a special NASA Board, chaired by the NASA Associate Administrator for Manned Space Flight. SSD, Aerospace, NASA and contractor personnel presented details of all subsystem design, function, component testing, and status to support the GT-3 mission.

(3) Coordination Meetings (II. A. 2)

The management and technical coordination meetings between the Air Force/Aerospace, NASA, Martin, and the associated contractors provided a satisfactory means for each manager to keep current with the program philosophy, objectives, goals, requirements, and problems. As such, each agency or company was able to associate and identify with its respective responsibilities, and to work objectively and cooperatively with the others in the accomplishment of tasks.

At these meetings, the full cooperation of all concerned was evident. The GLV-2 shutdown incident is an example of the importance of the coordination meetings. This action could not have been accomplished in six weeks without the full cooperation and coordination of the entire team.

7. ETR SURVEILLANCE

The same methodical approach to testing, data analysis, documentation, and failure investigation used in support of product integrity was vigorously pursued by all Eastern Test Range (ETR) Gemini agencies. The complexities of multiple interfaces and mutliple countdowns added emphasis to the extreme need for close coordination and attention to detail. The concepts of the Pilot Safety Program[5] were included in the normal organizational structure and given support by all agencies. A series of Pilot Safety Working Teams were established, by system, with the fundamental purpose of assuring that all aspects of product assurance were maintained (Figure II. B-8). These teams reported regularly to management groups and their inputs were integrated into the Gemini Launch Vehicle Working Group.

The Gemini Launch Vehicle Working Group (GLVWG) was the governing body in the administration of the program at ETR. Its executive chairman was the Chief, Gemini Launch Division, 6555th Aerospace Test Wing. The technical chairman was the Head of Operations, Aerospace/ETRO. Other Government agencies in the GLVWG were NASA and the Patrick Test Site Office (Air Force Quality Assurance). The members included technical representatives of the integrating contractor (Martin) and all the associate contractors (Aerojet, G. E., Burroughs, and Pan American).

The GLVWG convened weekly. It issued the Launch Test Directive (LTD), which controlled and specified the Gemini test program in Florida. In addition, the GLVWG issued the milestone schedule and was responsible for resolving interface or coordination problems between contractors or agencies. The GLVWG also coordinated milestone range support and maintained current information on the progress of the launch checkout, the problems during checkout, and logistics.

The GLVWG had two working groups, which reported on a weekly basis: the Facilities Working Group (FWG) and Pilot Safety Active Review Team (ART). In major problem areas requiring action, the FWG and ART referred matters to the GLVWG for resolution and direction.

The GLVWG also acted as the Pilot Safety Status Review Team on F-1 Day. The Active Review Team made a formal report of launch vehicle readiness on F-1 Day. The Status Review Team determined the readiness to launch and presented their recommendations and status to the Flight Safety Review Board.

8. FLIGHT SAFETY REVIEW BOARD

The Flight Safety Review Board was a concept adopted by Air Force/Aerospace Corporation management during the Mercury Program and carried into the Gemini Program specifically to determine flight readiness of the hardware to support a manned mission.

This board constituted the final focal point activity of the Pilot Safety Program and convened purposely as close to launch day as practical (usually one day before). This was done so that routine prelaunch activity, during which troubles could develop, was minimized. The board had to satisfy itself as to the status of the launch complex AGE and airborne hardware from detailed technical briefings by Aerospace Corporation and USAF 6555th Aerospace Test Wing personnel and make the ultimate decision as to whether the USAF would commit the vehicle to launch in support of the NASA mission. These briefings were a summary of an independent technical evaluation of the vehicle by Aerospace Corporation and a comprehensive review of the checkout history and launch readiness status conducted by the 6555th ATW. All participating contractors were present during this board meeting for special questioning and final commitment for manned flight.

This discipline acted as a powerful and effective management tool in terms of motivation for all agencies. The role of each launch team participant was significantly magnified as this final review approached. Problems experienced between launches relating to each agency were reviewed for proper resolution and the associated risks assessed for the final time. It would have been extremely difficult for a problem of significance to have escaped the intensive reviews that were necessary to prepare for this final status report.

The structure and basic scope of the Flight Safety Review Board is presented in Figure II. B-9.

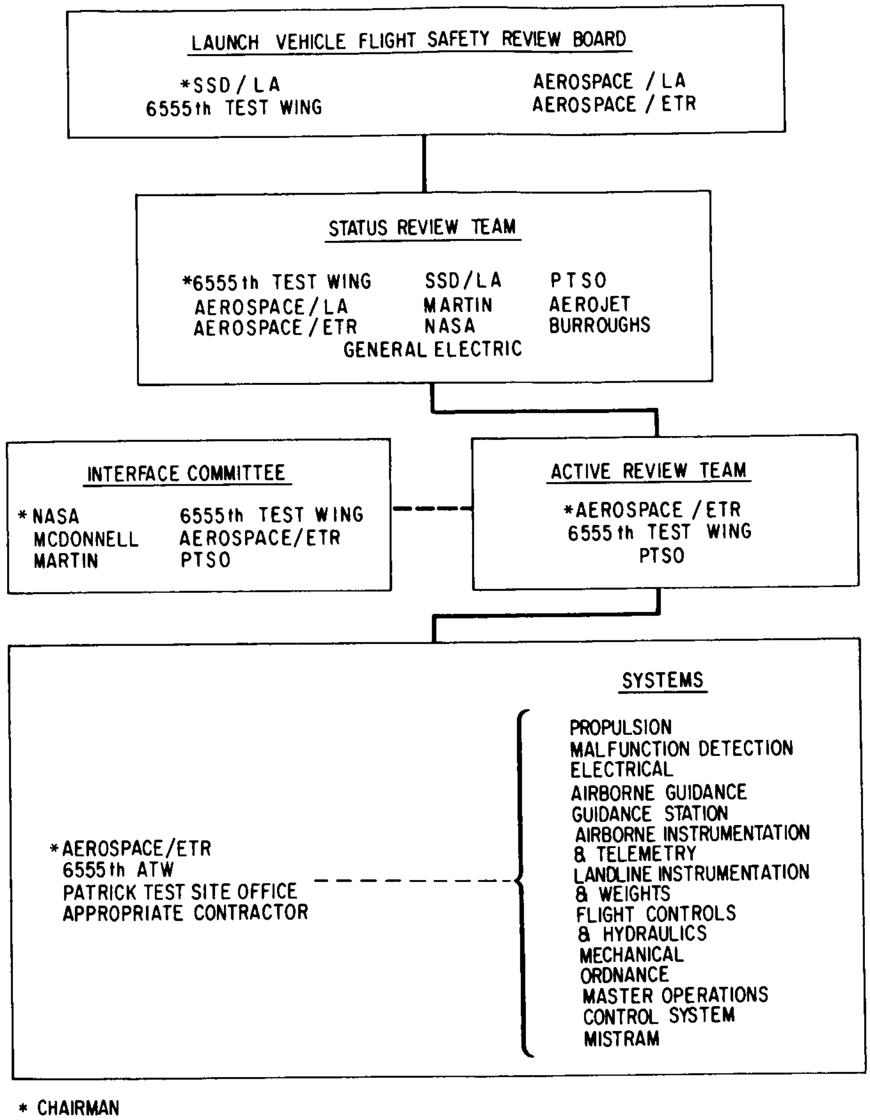


Figure II. B-8. Pilot Safety Working Teams

FLIGHT SAFETY REVIEW BOARD

BOARD MEMBERS

- AIR FORCE/SSD COMMANDER (CHAIRMAN)
- AIR FORCE/SSD DEPUTY FOR LAUNCH VEHICLES
- AEROSPACE CORPORATION, CORPORATE OFFICE
- AIR FORCE TEST WING
- AEROSPACE CORPORATION, EASTERN TEST RANGE
- NATIONAL AERONAUTICS AND SPACE ADMINISTRATION (OBSERVERS)

THE CULMINATION OF ALL PREVIOUS EFFORTS FROM A SYSTEMS ENGINEERING EVALUATION OF SUCH THINGS AS:

- PREVIOUS TITAN FAMILY FLIGHT HISTORY (T-II, T-III, GEMINI)
- FABRICATION AND TEST HISTORY (FACTORY AND FIELD)
- GUIDANCE EQUATIONS AND RELATED SOFTWARE

F-1 DAY FINAL PRESENTATION FOR LAUNCH VEHICLE FROM PERFORMANCE AND RELIABILITY STANDPOINT TO COMMIT FOR MANNED FLIGHT

- PREVIOUS PROBLEMS - CORRECTIVE ACTIONS CONSIDERED AND TAKEN
- OPEN PROBLEMS - POTENTIAL EFFECTS ON MANNED FLIGHT AND ASSOCIATED RISKS

Figure II. B-9. Flight Safety Review

9. FLIGHT EVALUATION

Early in the program, meetings were held among SSD, Aerospace, and Associate Contractors to determine data requirements, schedules, and the types of reports required for flight evaluation. The final plan is outlined in Figure II. B-10. The data received by each Associate Contractor is shown in Figure II. B-11.

The Martin-Marietta/Baltimore Company, Aerojet General Corporation, General Electric Company, and Burroughs Corporation used individual comprehensive data reduction and analysis programs to provide data for subsystem analysis. These data were reviewed independently by the respective contractor systems engineer and by Aerospace, first to look for major anomalies to be presented and discussed at the Post-flight Evaluation Meeting following each launch, and then to be extensively presented in a final 45-day report.

The Aerospace Corporation developed its own programs parallel to those of the Contractors, which encompassed all data and reports (Figure II. B-12). This permitted an individual analysis to be made of each system and revealed any problems that may have occurred. There were three main data programs: the telemetry data book, BEEP (Best Estimate of Engine Parameters), and PFRP (Post-flight Reconstruction Program). The telemetry data book consisted of 250 pages in which one telemetry function was presented per page. The presentation was in engineering units vs time with expanded time scales at critical flight times or in problem areas. Copies of these data books were sent to NASA as an aid to their Mission Evaluation Team. Through the BEEP flow rates, thrust, Isp, mixture ratios, etc. were determined. This program combined telemetry and tracking data for the determination of these parameters. The PFRP utilized the telemetry and tracking data to determine flight-control parameter drifts. The data was then analytically evaluated by the system personnel, reviewed for major anomalies for the Quick Look report, and then in detail for presentation in the Aerospace 60-day Post-flight report.

The Gemini Launch Vehicle Working Group, described in Section II. B-7, published its post-flight report 15 days after launch.

Under a special provision, four M/B systems specialists were assigned to NASA/MSC as working members of the NASA Mission Evaluation Team. These four assisted in the preparation of the NASA Gemini Mission Reports and were assigned to MSC/Houston from launch + 5 to launch + 20 days. Information obtained from the Aerospace and GLVWG reports, the Aerospace Post-flight Evaluation Meeting, and Contractor inputs was used to prepare and coordinate the GLV Section of the NASA Mission Reports.

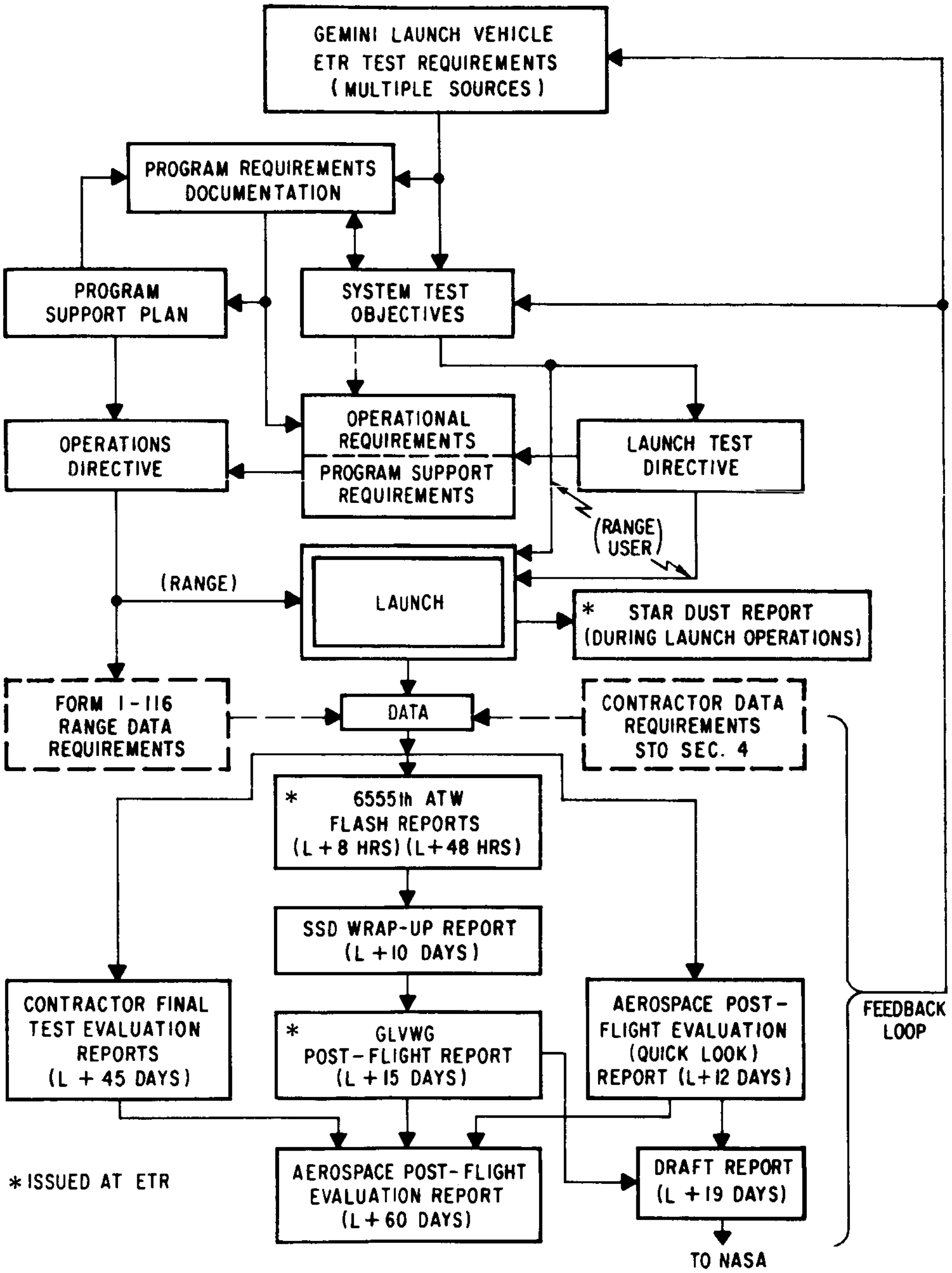


Figure II. B-10. Test Documentation Flow

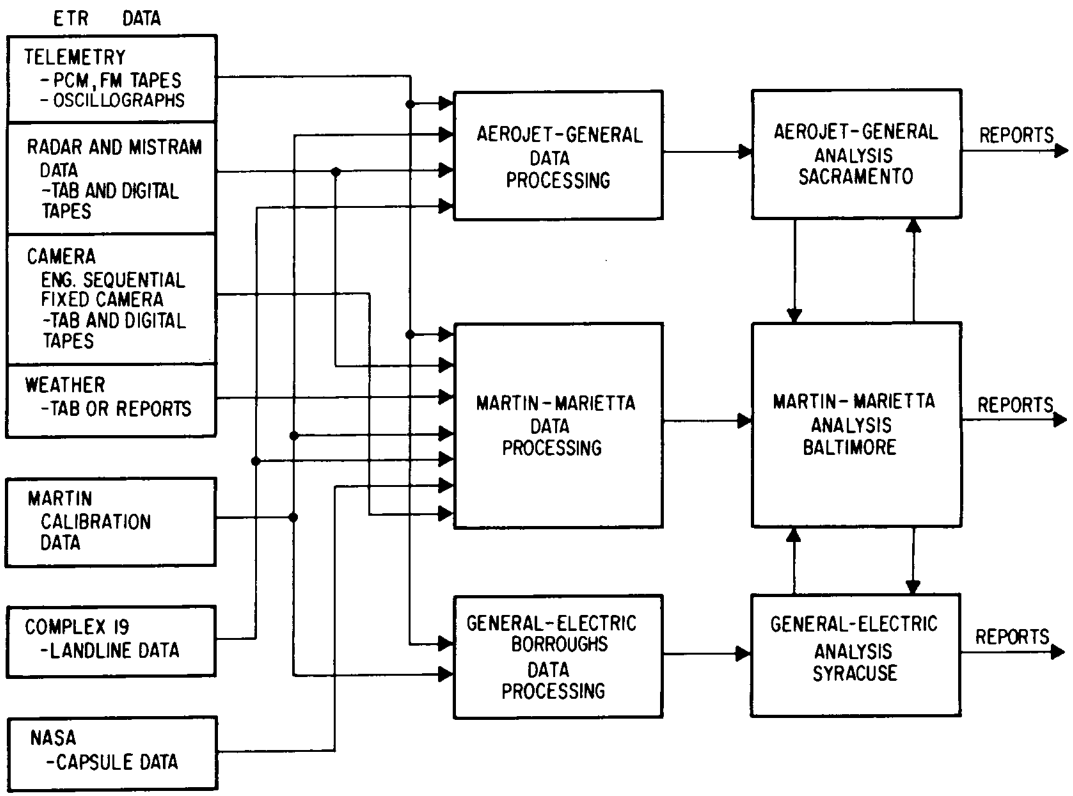


Figure II. B-11. Associate Contractor Data

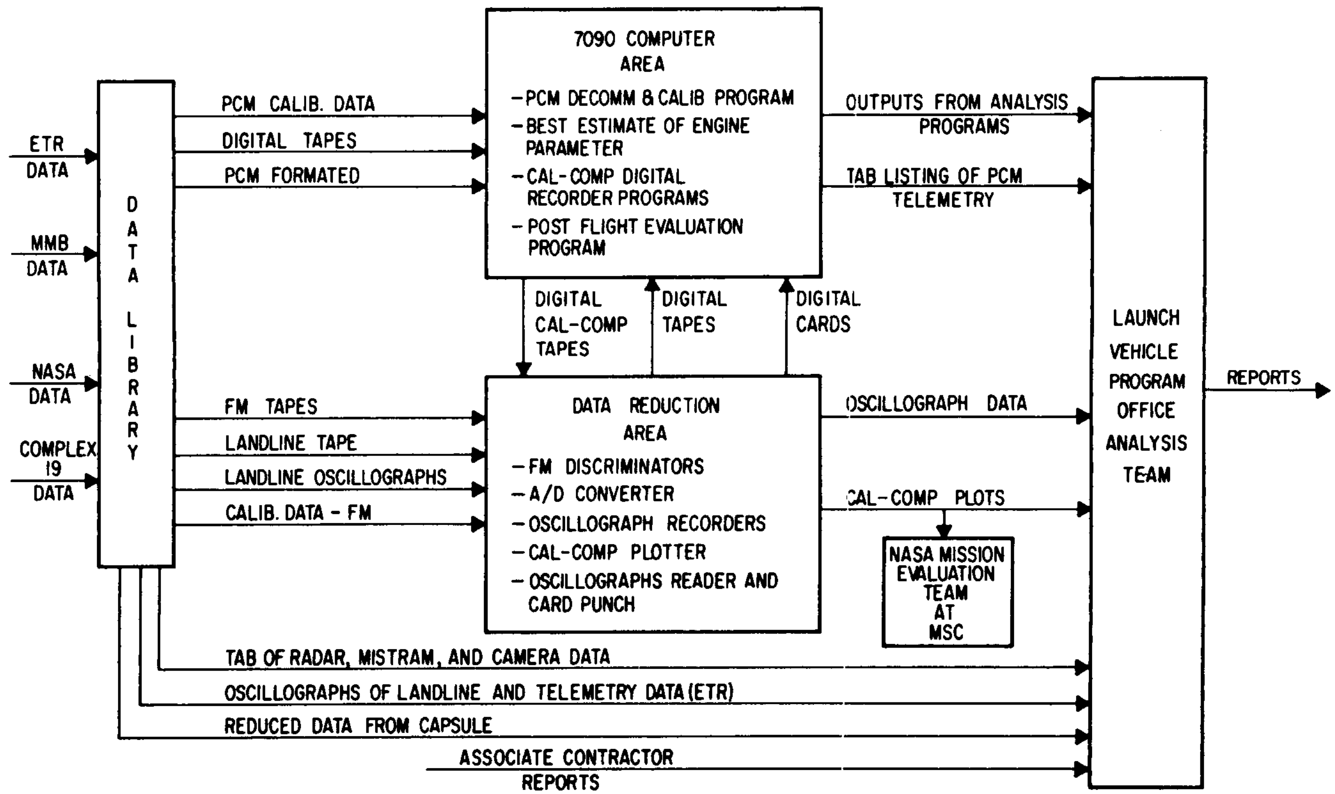


Figure II. B-12. Aerospace Data and Reports Flow

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SECTION II-B

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SECTION II-B

DEFINITIONS

AEIP	Augmented Engine Improvement Program
AFBS	Air Force Bonded Stores
AFPRO	Air Force Plant Representative Office
AFQA	Air Force Quality Assurance
AFQC	Air Force Quality Control
AGC	Aerojet General Corporation
AGE	Aerospace Ground Equipment
AMR	Atlantic Missile Range
ART	Active Review Team
ATW	Aerospace Test Wing
BEEP	Best Estimate Engine Performance
BOI	Break of Inspection
BSD	Ballistic Systems Division
CARE	Corrective Action and Repair Evaluation
CCB	Configuration Control Board
CSAT	Combined Systems Acceptance Test
DAA	Discrepancy Analysis Area
DASO	Demonstration and Shakedown Operation
DOD	Department of Defense
ECP	Engineering Change Proposal
ELT	Environmental Life Testing
ERB	Engineering Review Board
ETD	Engineering Test Directive
ETR	Eastern Test Range
FA/CA	Failure Analysis/Corrective Action
FACI	First Article Configuration Inspection
FMT	Failure Mode Testing
FWG	Facilities Working Group
GATV	Gemini Agena Target Vehicle

GCI	Gemini Configuration Index
GE	General Electric
GFRB	Gemini Flight Readiness Board
GLV	Gemini Launch Vehicle
GLVWG	Gemini Launch Vehicle Working Group
GPIS	Gemini Problem Investigation Status
GT	Gemini Titan
GTO	Ground Test Only
IR	Inspection Report
LIT	Launch Integrity Team
LTD	Launch Test Directive
LMSC	Lockheed Missiles & Space Company
MARS	Martin Automatic Reporting System
M/B	Martin Baltimore
M/C	Martin Cape
M/D	Martin Denver
MDR	Malfunction Discrepancy Reports
MMC	Martin-Marietta Corporation
MRB	Material Review Board
MSC	Manned Spacecraft Center
NASA	National Aeronautics & Space Administration
OOAMA	Ogden Air Material Area
PCM	Pulse Code Modulation
PFRP	Post Flight Reconstruction Program
PMT	Production Monitoring Test
POGO	Colloquialism - Defining longitudinal oscillation
PTSO	Patrick Test Site Office
QE	Quality Engineering
QRR	Quality Reliability Report
R & D	Research & Development

DEFINITIONS (Continued)

RE	Reliability Engineering
RGO	Radio Guidance Operation
ROPA	Resolution of Problem Area
R/QA	Reliability/Quality Assurance
SSD	Space Systems Division, United States Air Force
STL	Space Technology Laboratory
TARS	Three-Axis Reference System
TD	Technical Direction
USAF	United States Air Force
VAFB	Vandenberg Air Force Base
VAT	Vehicle Acceptance Team
VTF	Vertical Test Facility

C. LAUNCH VEHICLE DEVELOPMENT AND CONFIGURATION

1. GENERAL

At the beginning of the Gemini program, the Titan II ballistic missile was chosen as the launch vehicle to be used in conjunction with the Gemini spacecraft. The decision was made at that time to use the Titan II "as is" with only those modifications required to enable the Titan II to perform the launch vehicle function for a manned system. The general arrangement and configuration of the Gemini Launch Vehicle are shown in Figures II. C-1 and II. C-2. Modifications are shown in Figure II. C-3. Modifications were approved only if they added to the system reliability or to pilot safety. The principal modifications were:

1) Addition of a Malfunction Detection System (MDS), designed to sense problems in any of the vital booster systems and transmit this information to the astronauts.
2) Addition of a propulsion system Prelaunch Malfunction Detection System (PMDS) to assure satisfactory operation of the Stage I autogenous propellant tank pressurization system prior to vehicle release.
3) Addition of a redundant flight control system which could take over the functions of the primary system, should the primary system fail in flight.
4) Addition of redundancy in the electrical system and sequencing functions with necessary changes to provide power for such added launch vehicle equipment as the MDS.
5) Substitution of a radio guidance system, similar to that used on Mercury, for the inertial guidance system used on the Titan II ICBM, to provide weight reduction and a more responsive system during critical orbital injection and variable launch azimuth capability.
6) Elimination of retro-rockets and vernier rockets since their functions were not required for the Gemini launch vehicle mission.
7) Modification of the equipment truss in the vehicle second stage to hold much of the new flight control, MDS, and guidance equipment.
8) Addition of a new Stage II oxidizer tank forward skirt assembly to mate the launch vehicle to the spacecraft.
9) Addition of redundancy in the hydraulic system, where required for pilot safety.
10) Addition of equipment in the Stage I propellant feed lines to suppress the vehicle longitudinal oscillation (POGO) during first stage flight.
11) Replacement of the AVCO tube type range safety receivers by solid state Advance Communications, Inc., units, to allow weight savings and incorporate airborne time delays for astronaut escape in the event destruct command was transmitted.
12) Substitution of a high level telemetry encoder (0 to 5 v) for the Titan II system (0 to 40 mv) to increase signal-to-noise ratio, expecially from transducers distant from the signal conditioner.
13) Addition of the necessary logic and equipment to utilize the spacecraft inertial guidance system as the launch vehicle secondary guidance source.

Section II-B of this report described the Pilot Safety Program in terms of both philosophy and practical implementation, pointing out that the program was comprised essentially of three primary efforts:

1) Design improvements
2) Product integrity
3) Malfunction Detection and Abort capability

From a very generalized standpoint, it was meant to accomplish three basic features:

1) Through systems trade-off studies, arrive at the best practical compromise that would provide a vehicle with an acceptable "inherent" design reliability.
2) Assure that the vehicle, once designed, was manufactured and tested such that the inherent design reliability could be realized.

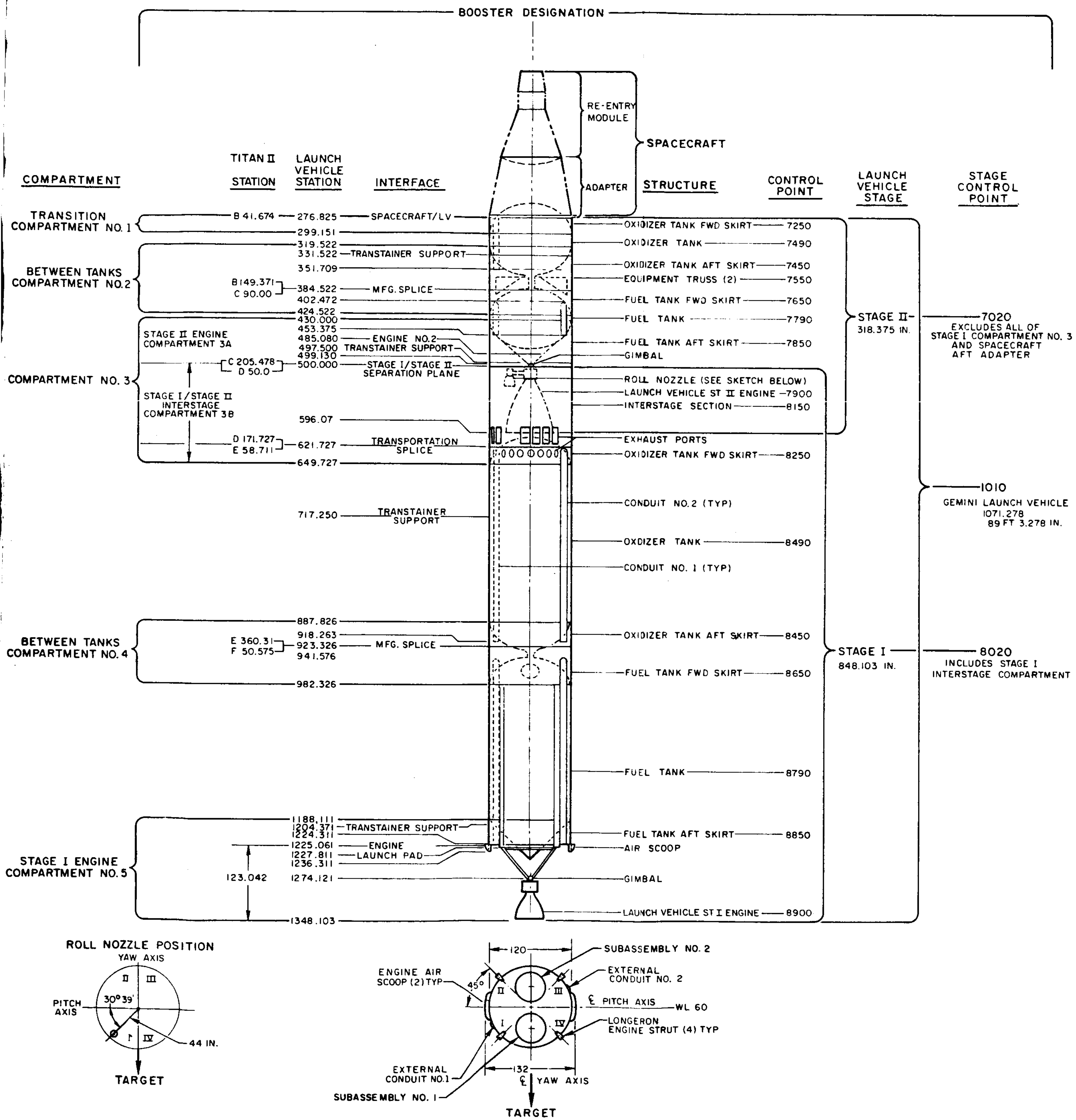
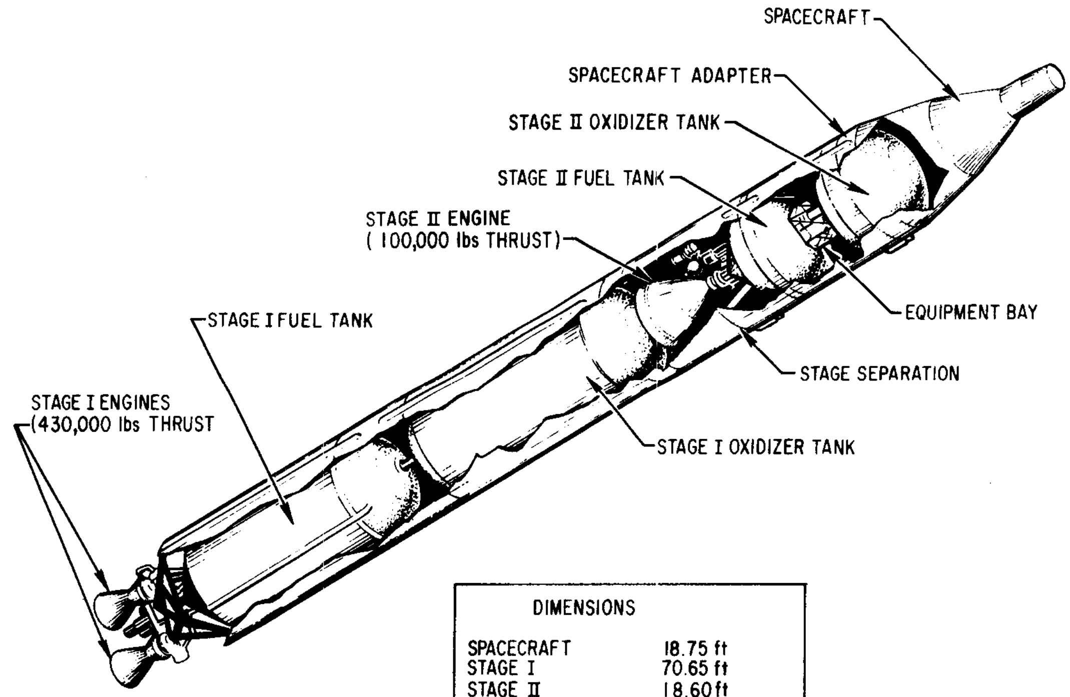


Figure II. C-1. General Arrangement - Gemini Launch Vehicle



DIMENSIONS	
SPACECRAFT	18.75 ft
STAGE I	70.65 ft
STAGE II	18.60 ft
TOTAL LENTH	108 ft
OXIDIZER	NITROGEN TETROXIDE
FUEL	AEROZINE 50

Figure II. C-2. Gemini Launch Vehicle Configuration - Simplified

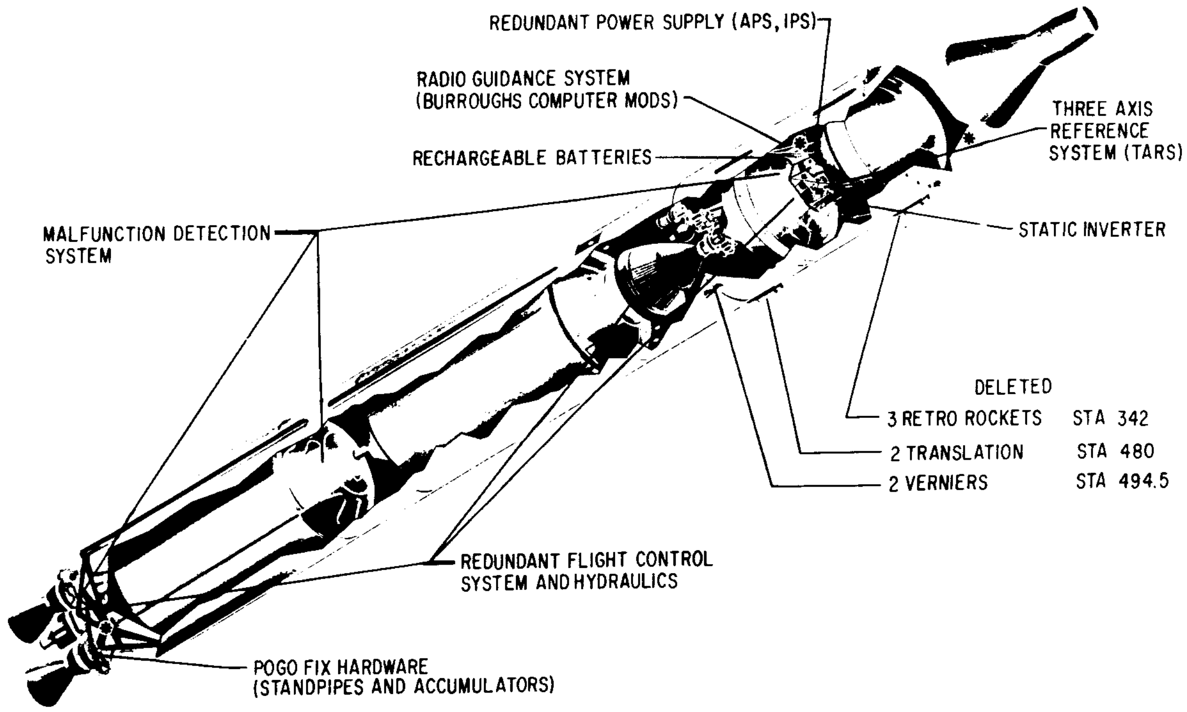


Figure II. C-3. Gemini Launch Vehicle Modifications

3) Recognizing that no system is ever 100% reliable, provide monitoring capability in real time that would permit malfunctions to be recognized and subsequent actions to be taken to 1) provide additional capability for mission success or 2) provide safe pilot abort.

The thirteen items outlined above represented the initial design improvements made to the vehicle. Other improvements were subsequently incorporated as the program progressed. These improvements will all be discussed under their respective subsystems in the following subsections. The addition of a slow malfunction monitoring capability and the safe-abort measures taken will also be discussed.

The Gemini Launch Vehicle was composed of the following primary subsystems:

Airframe/Structures

Engines

Vehicle Propulsion

Flight Controls

Electrical

Malfunction Detection System

Flight Termination

Instrumentation

Guidance

In addition, the GLV was supported by the following:

Ground Guidance Computer System

Guidance Equations

Each of these areas will be generally discussed in the following sections in terms of:

System Title

System Description and Design Concept

Configuration

Component History, including development, qualification testing, production history and final system status.

2. AIRFRAME/STRUCTURES

a. System Description and Design Concept

The GLV structural configuration was the same as the Titan II ICBM with the exception of changes necessary to meet the Gemini mission requirements. The basic construction was a semi-monocoque configuration with short sections of monocoque structure in transitional areas between propellant tank tangency and skirts.

The structure was designed to withstand ultimate load without failure at maximum anticipated temperature. Ultimate load is defined as limit, or maximum anticipated load, times the design factor of safety (1.25 for all loads except relieving pressure loads where it is 1.0). In addition, the criteria of no excessive deformations, which would restrict the launch vehicle from meeting performance requirements at limit loads, existed.

b. Configuration

The major airframe differences between GLV and Titan II are listed below. The numbers below in paranthesis refer to locations on Figure II. C-4.

1) Forward Oxidizer Tank Skirt - The skirt was an integrally stiffened cylinder, replacing the Titan II monocoque cone frustum, whose diameter was 120 inches and whose length was 22.33 inches. The spacecraft adapter was attached to the forward oxidizer skirt by twenty (20) external lugs and bolts.

2) To accommodate pre-launch firing of the spacecraft's OAMS rockets, two scuppers were located diametrically opposed to each other. Each scupper assembly consisted of an ablative-lined, fabricated aluminum body incorporating a flexible seal at one end and a mounting flange at the other. The seal-equipped end (inlet) mated with the OAMS rocket thruster face while the flanged (outlet) end was secured around an exhaust port in the forward skirt.

3) On GLV's -1, -2, and -3, the forward oxidizer skirt was coated with thermal insulating material (MMS K-438) to protect the structure from protuberance heating caused by the external lugs at the spacecraft launch vehicle interface. Analysis of temperature measurements taken on GLV-1 in flight resulted in (1) a reduction in insulation to half thickness on GLV-2 and GLV-3 and (2) elimination of the insulation entirely on GLV-4 and up. This was due to the fact that the above measurements showed that initial thermal design calculations were excessively conservative.

4) An interface seal was provided for the attachment of the spacecraft adapter to the booster forward oxidizer tank skirt in order to maintain pressure in the spacecraft adapter.

5) The Stage II equipment trusses were redesigned to (1) accommodate the MDS, RGS and other Gemini peculiar components and (2) to reduce the truss weight from Titan II (70 lbs).

6) Titan II peculiar retro-rockets and vernier engines, together with the structural provisions for these units were deleted.

7) One of the two external conduits on Stage II was deleted, since all wiring could be routed through the remaining external conduit.

8) The bolts used in the manufacturing splice at station 923 were changed on GLV-9 from AN bolts to high strength NAS bolts in order to improve the tension capability of the manufacturing splice. By improving the tension capability of station 923, breakup would have been forced at the interstage in the event of a ride-it-out abort during Stage I flight.

c. Development History

(1) Development Testing

During development testing of Titan II, a structural failure occurred between the tanks of Stage I. The initial failure was in the region of the waffle section at the skirt to tank juncture. This waffle

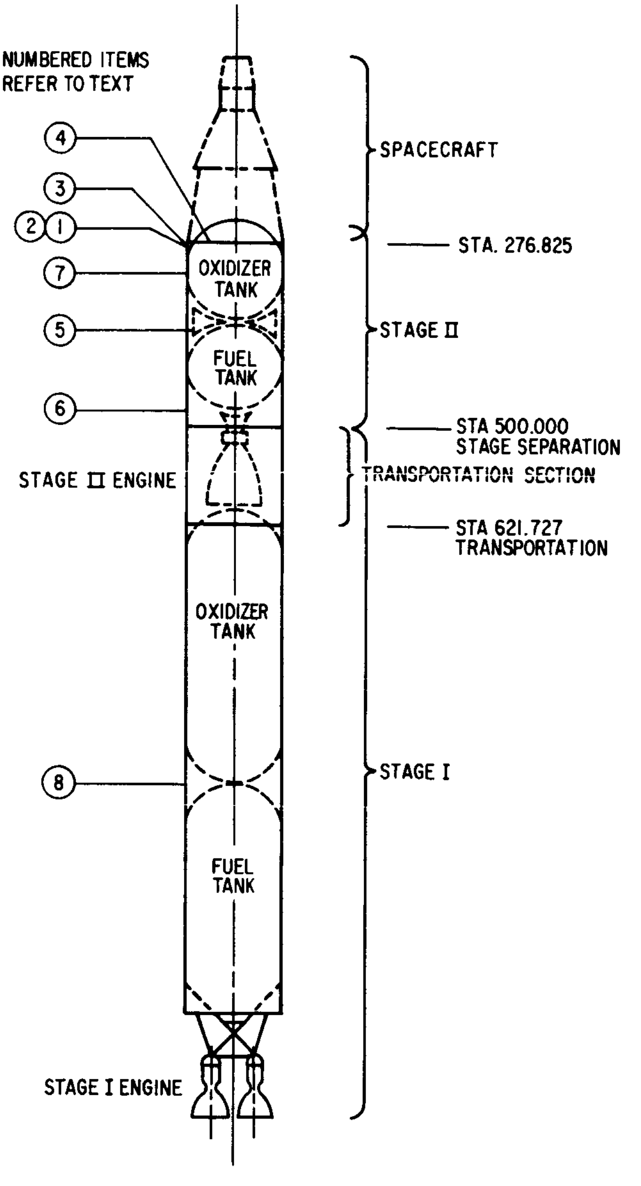


Figure II. C-4. GLV Major Structural Components section was redesigned and the structure was successfully tested to design conditions. The redesigned waffle configuration, which was flown on all but 10 Titan II's, was used on the GLV successfullv.

(a) Static Test. The calculated loads for the Gemini Launch Vehicle were equal to or less than the tested loads of the Titan II weapon system. Structural tests performed on the Titan II program satisfied the test requirements of the Gemini program for the structural elements which were identical. A comparison of tested and design loads is shown in Figure II. C-5. The Gemini peculiar structure was successfully tested for the GLV design loads, and a detailed stress analysis was conducted.

(b) Wind Induced Oscillation. A 7.5 percent scale model of the GLV, the Complete Vehicle Erector and the umbilical towers in the pre-launch configuration was tested at NASA Langley Research Center's Dynamic Tunnel to obtain air vehicle and erector bending moments due to steady ground winds. The GLV was dynamically scaled. The Complete Vehicle Erector was frequency and geometrically scaled, while the umbilical towers were geometrically scaled only. The results of the wind tunnel testing verified the adequacy of the pre-launch design loads for the launch vehicle. In addition, the data was used to establish wind restrictions for the Complete Vehicle Erector.

(c) Wind Tunnel Static Force and Pressure Tests. A 6.0 percent model of the complete Gemini Vehicle was tested in the NASA Ames Research Center Unitary Facility to determine the aerodynamic force coefficients and static pressure distribution of Mach numbers up to M=3.5. The results of these tests were incorporated into the Gemini design loads.

(d) Buffet Load Tests. Wind tunnel tests were performed to determine the effects of fluctuating pressures in the transonic flight regime. McDonnell Aircraft Corporation wind tunnel data defined the magnitude of local fluctuating pressures. Additional tests were performed in the NASA Ames Research Center Facility on a 8.5 percent model of the spacecraft and forward portion of the launch vehicle to determine the overall vehicle bending loads induced by transonic buffet fluctuating pressures. The results of these tests were incorporated into the Gemini design loads.

(2) Special Structural Analysis

(a) Interface Structural Analysis. TOR-269(4126-60)-2[1], dated 19 August 1964, was a detailed redundant analysis of the GLV forward skirt and aft end of spacecraft adapter. Results showed excellent correlation with strain gage results from interface structural tests.

(b) Structural Response of the Gemini Launch Vehicle Forward Skirt and Spacecraft Adapter to Fluctuating Buffet Pressures. TOR-269(4126-60)-3[2] dated 15 August 1964. This analysis showed high ring stresses due to buffeting pressures. Subsequent MAC analysis verified the basic contentions of the Aerospace analysis but the final MAC strength analysis showed adequate margins of safety when the loads were combined.

(c) A Gemini Malfunction Transients and Loads Study was conducted by Martin and Aerospace. This study, plus additional effort, demonstrated that excessively high loads could result from switchover. This analysis formed the basis for the change in rate switch settings and fader lag times that were incorporated on the GLV.

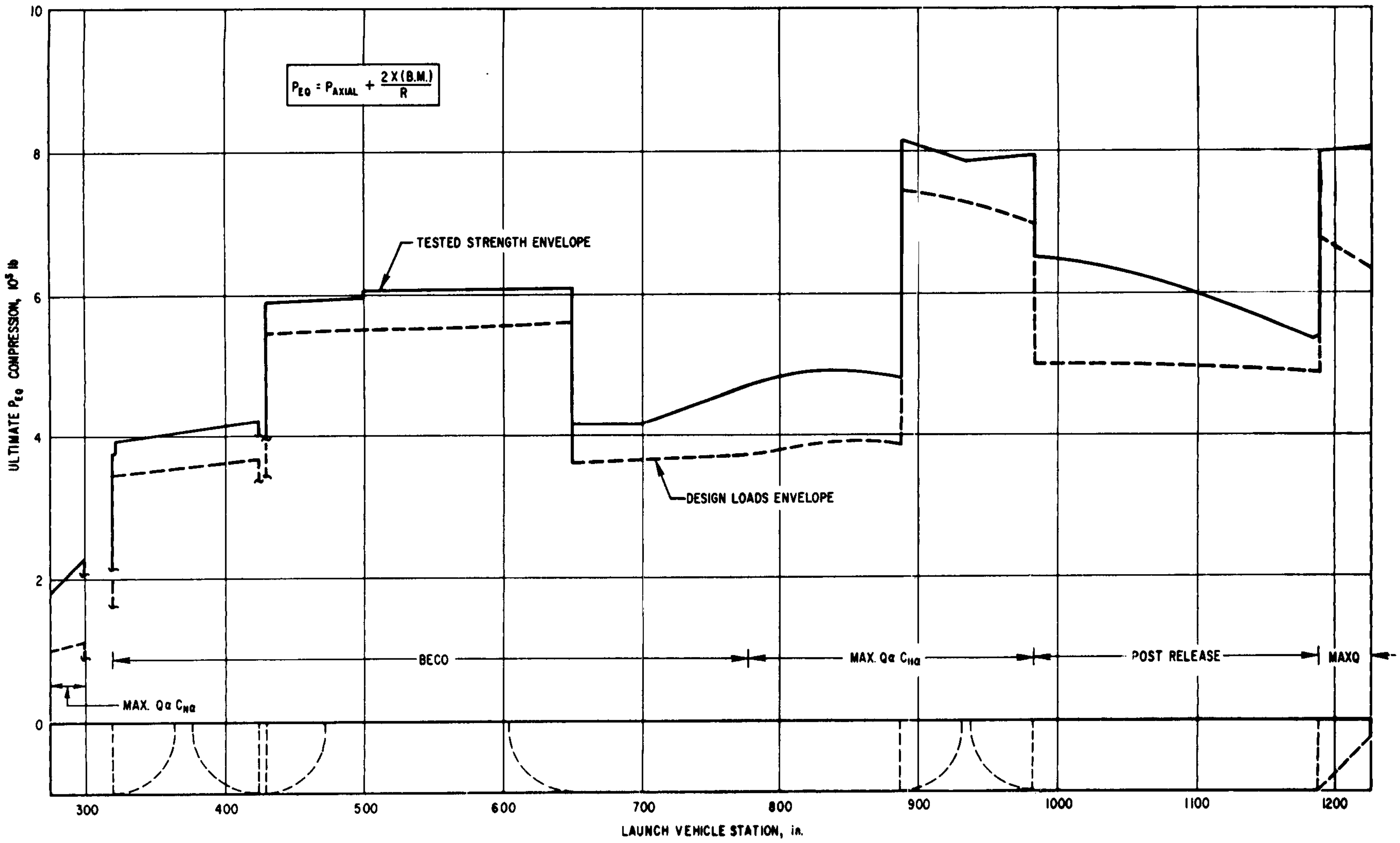


Figure II. C-5. Comparison of Tested and Design Loads

(d) GLV Detail Stress Analysis. In this effort, Martin performed a detailed stress analysis for the entire Gemini Launch Vehicle, including a re-analysis in more detail of Titan II designed structure as well as Gemini peculiar structure. Positive margins were shown for all structural elements as a result of the study.

d. Production History

(1) Evaluation of Defective Conduit Welds

Prior to the flight of GLV-2 minute transverse cracks were found in the longitudinal welds by x-ray and visual inspection on a Stage II fuel tank internal conduit detail after subassembly. Further inspection of other parts uncovered similar defects on all other Stage II conduits in stock and some cracked welds in Stage I conduit details.

An environmental life test program was initiated to investigate the structural adequacy of the defective welds. The test program consisted of a 450 cycle pressurization test and a 7.5 minute vibration test at 11.5 g's rms. After completion of the test program, the specimen was cut up and microscopically examined. These examinations, plus visual examination, x-ray examination, dye penetrant check and helium vapor emission testing, showed no evidence of crack propagation. The environmental life test program demonstrated that the conduits with these minute transverse cracks in the welds were structurally adequate. Although no problems were encountered in flight, GLV-9 and subs had improved conduits as an added factor of safety.

(2) Stage I Oxidizer Tank Contamination

To prevent the dome from reversing under pressures from Stage II Engine Start, a truss structure was installed internal to the Stage I oxidizer tank. An inspection of this truss structure during a GLV-5 tank cleaning operation revealed burrs lodged between the bolted assemblies. It was determined that this structure, which was installed internal to the tank, was not deburred after mating and drilling. Analysis of the particle sizes revealed a significant out-of-specification contamination. The corrective action decided upon was to enter the Stage I oxidizer tank of GLV-2 (at Complex 19) and perform a deburring and cleaning operation on each truss detail with subsequent cleaning of the complete tank. Identical operations were performed at Martin-Baltimore on the GLV-3 tank and subsequent assembled tanks. Furthermore, the manufacturing process plan was revised to specifically include deburring and cleaning of each detail before assembly.

(3) Acceptance Testing

Acceptance testing was performed at the individual propellant tank level and on tank conduits and feedlines. Acceptance tests for these structural components are described in the following paragraphs.

(a) Propellant Tanks

1) A low pressure helium vapor emission test was conducted prior to hydrostatic tests on GLV-1 through GLV-7. It was found to be unnecessary and was eliminated on GLV-8 and subsequent.

2) Hydrostatic test; four cycles at maximum operating pressure and two cycles at proof (1.1 x Maximum operating) levels.

3) A high pressure helium vapor emission test was conducted after hydro tests GLV-4 and GLV-5, and was replaced by a low pressure helium vapor emission test on GLV-6 and subsequent.

During hydrostatic test of the initial GLV-3 Stage I fuel tank, the tank failed at 95 percent of proof pressure. The failure analysis indicated that the start of tank burst was in an area where the skin gage was less than 50 percent of design thickness. This thinned-out area was due to excessive hand finishing by the vendor. Corrective action was taken to prohibit hand finishing by the vendor without Martin Company Engineering approval. Also, special tooling was developed to inspect the skin thickness of all Gemini fabricated tanks where standard inspection was not possible. All GLV tanks had this skin thickness inspection.

X-rays were taken of all welds before and after hydrostatic testing. Recycling of Steps 2 and 3 were required if weld repairs were accomplished after Steps 2 and 3.

(b) Conduit Assemblies

1) Five cycles of internal pressure.
2) Helium vapor emission test.

After installation of the conduits in the fuel tanks they were subjected to the tank hydrostatic tests and helium vapor emission tests.

(c) Feedlines

1) Hydrostatic test; five cycles at maximum operating pressure and one cycle at proof (1.5 x Maximum operating) level.
2) Helium vapor emission test.
3) X-rays of all welds were taken before and after hydrostatic test.

(d) **Rollout Inspection.** At tank rollout, all defective welds were reviewed and a detail stress analysis documented, when applicable. Positive margins of safety at ultimate loads had to exist for all defective weld areas.

(e) VTF and ETR. The airframe activities at VTF and ETR consisted essentially of maintenance of a +10°F inner tank dew point and external corrosion inspection. Moderate corrosion occurred on GLV-1 and GLV-2 at ETR. The corrosion was predominantly on the side of the vehicle facing north. The weather curtains on the north side of the erector had to be left open because of umbilical lines, air conditioning lines, etc., running into these areas. This condition was corrected by modifying the curtains such that they could be closed and the lines fed through them.

Corrosion inspection of the vehicle during the time it was on the launch pad was conducted periodically.

(f) Flight. There were no flight failures or airborne propellant leaks associated with the airframe during the Titan II and GLV flight history. One small oxidizer leak at a spotweld on GLV-11 was satisfactorily repaired prior to flight.

Early in the Titan II program, it was observed that the transportation section disintegrated after staging due to the Stage II Engine Start-up environment. Airborne cameras and added instrumentation were installed on five vehicles to record the event. Exhaustive studies of the motion of the debris revealed that no particles had trajectories that could impinge on Stage II, i.e., all debris moved away from the sustained stage. It was concluded that this occurrence did not present a problem.

On GLV-10 and GLV-12 a review of film coverage of the flights indicated that the Stage I oxidizer tank was venting (i.e., N_2O_4 vapor emission). On GLV-12 the fuel tank on Stage I also seemed to vent. A thorough study was conducted by Martin and Aerospace to attempt to ascertain the cause of the tank ventings. The fuel tank venting is felt to be caused by debris from the oxidizer tank; however, this is only conjecture. The most probable causes were found to be: debris impingement from transportation section; cracking of the ablative coating on the dome and subsequent failure due to weakening of the dome by excessive heat; and tilting of Stage I resulting in burn through of the tank side wall. It was ascertained that regardless of the cause, this venting proved no hazard to Stage II and thus had no detrimental effect on either mission success or astronaut safety.

Several minor anomalies occurred during the program such as the post SECO bursts of acceleration, occurrences of "green man," etc. Analysis of these anomalies was rendered almost impossible by the removal of lateral accelerometers after GLV-4 and strain gages after GLV-3.

The GLV-5 Stage I oxidizer tank, minus the aft dome, was recovered (Figures II.C-6 and II.C-7) after the GT-5 mission. The tank was in remarkably good condition, considering the reentry environment it has been subjected to. An extensive study of the tank was conducted by MB at Air Force direction and the results are summarized in Martin's Report LV 407 "Engineering Evaluation of GT-5 Recovered Stage I Oxidizer Tank."[3]

Figure II. C-6. GLV-5 Oxidizer Tank After Re-entry

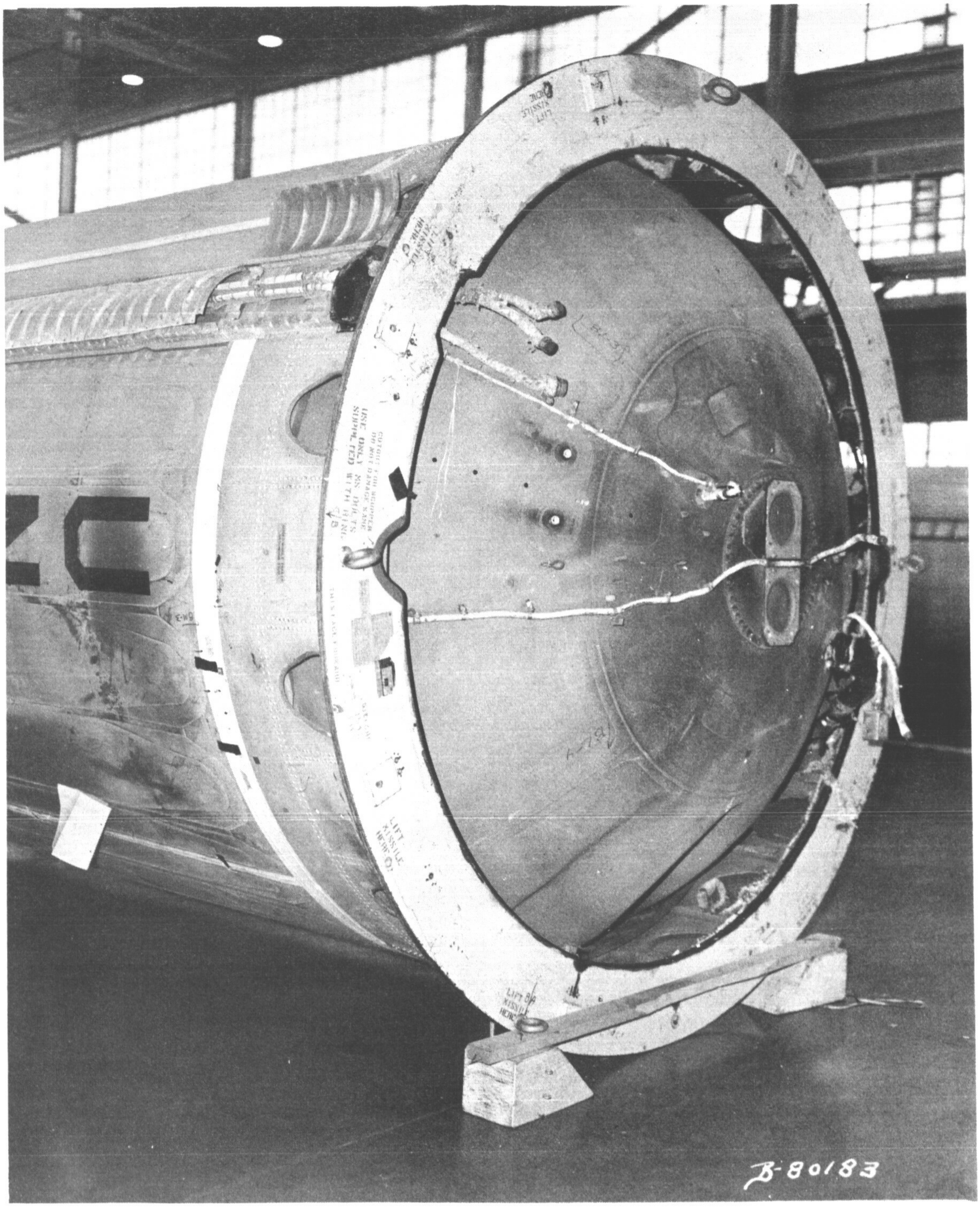


Figure II. C-7. Recovered Oxidizer Tank at Martin/Baltimore

3. ENGINES[4]

Propulsion for the GLV system was provided by the Aerojet-General Corporation manufactured YLR87-AJ-7 Stage I and YLR91-AJ-7 Stage II engines. These engines were modified Titan II engines which, in accordance with letter AFBSD to AGC on 3 December 1964, were certified to have met qualification requirements.

The basic Gemini engine system and those major features which were unique to the Gemini system are described below. (See Figures II. C-8 and II. C-9 for reference.)

a. Stage I Engine

(1) System Description

The YLR87-AJ-7 rocket engine assembly was a storable liquid bi-propellant, turbopump fed, dry-jacket-start engine rated for sea level operation. The engine was composed of two independently operating subassemblies which operated simultaneously, mounted on a single thrust takeout structure (frame). Each subassembly contained a thrust chamber assembly, turbopump assembly, gas generator assembly, starter cartridge, propellant plumbing and electrical controls harness. In addition, subassembly two (S/A 2) provided the energy source for tank pressurization. The major components of each subassembly were identical except for the pressurization system.

The design concept of this engine system was to achieve reliability through simplicity. This goal was achieved by use of hypergolic storable propellants which eliminated the need for bleed down, heaters and ignition systems. The engine required no functions during countdown beyond admission of propellant to the engine from the tankage. Thrust level was preset by sized cavitating venturis which were located in the gas generator propellant feed circuits. Cavitating venturis controlled flow to the gas generator which controlled turbine power by hydraulic equilibrium, thereby eliminating all servo thrust control mechanisms.

Descriptions of the major components follow.

(a) Thrust Chamber Assembly. The thrust chamber assembly consisted of a regeneratively fuel cooled tubular construction combustion chamber, injector, injector dome, fuel thrust chamber valve assembly, oxidizer thrust chamber valve assembly, gimbal assembly, thrust chamber pressure switch, and oxidizer and fuel propellant lines. The above components were packaged as an integral assembly.

The thrust chamber fuel valve consisted of a four inch gate butterfly valve with attached hydraulic actuator. The actuator was spring loaded in the closed position. The actuator shaft was mechanically linked to the fuel valve shaft. The fuel valve shaft in turn had a clevis for linkage to the oxidizer thrust chamber valve.

Control of the thrust chamber fuel valve actuator was achieved by use of a direct mounted pressure sequence valve. The pressure sequence valve (PSV) mechanically sensed fuel system supply pressure and opened at a level which allowed fuel pressure to overcome the spring force of the thrust chamber fuel valve actuator and open the fuel and oxidizer valve by mechanical linkage. Closure of the thrust chamber valve was achieved by an electrical solenoid which overrode the pressure sensing element causing the PSV to neutralize and admit fuel pressure to the closing side of the thrust chamber actuator to assist spring closure of the thrust chamber valve.

(b) Gas Generator Assembly. Each gas generator system consisted of an integral gas generator chamber and injector, fuel check valve, oxidizer check valve, gas generator feed lines, cavitating venturis and propellant strainers. The gas generators were mounted directly to the inlet of the turbine pump assembly turbine drive manifold.

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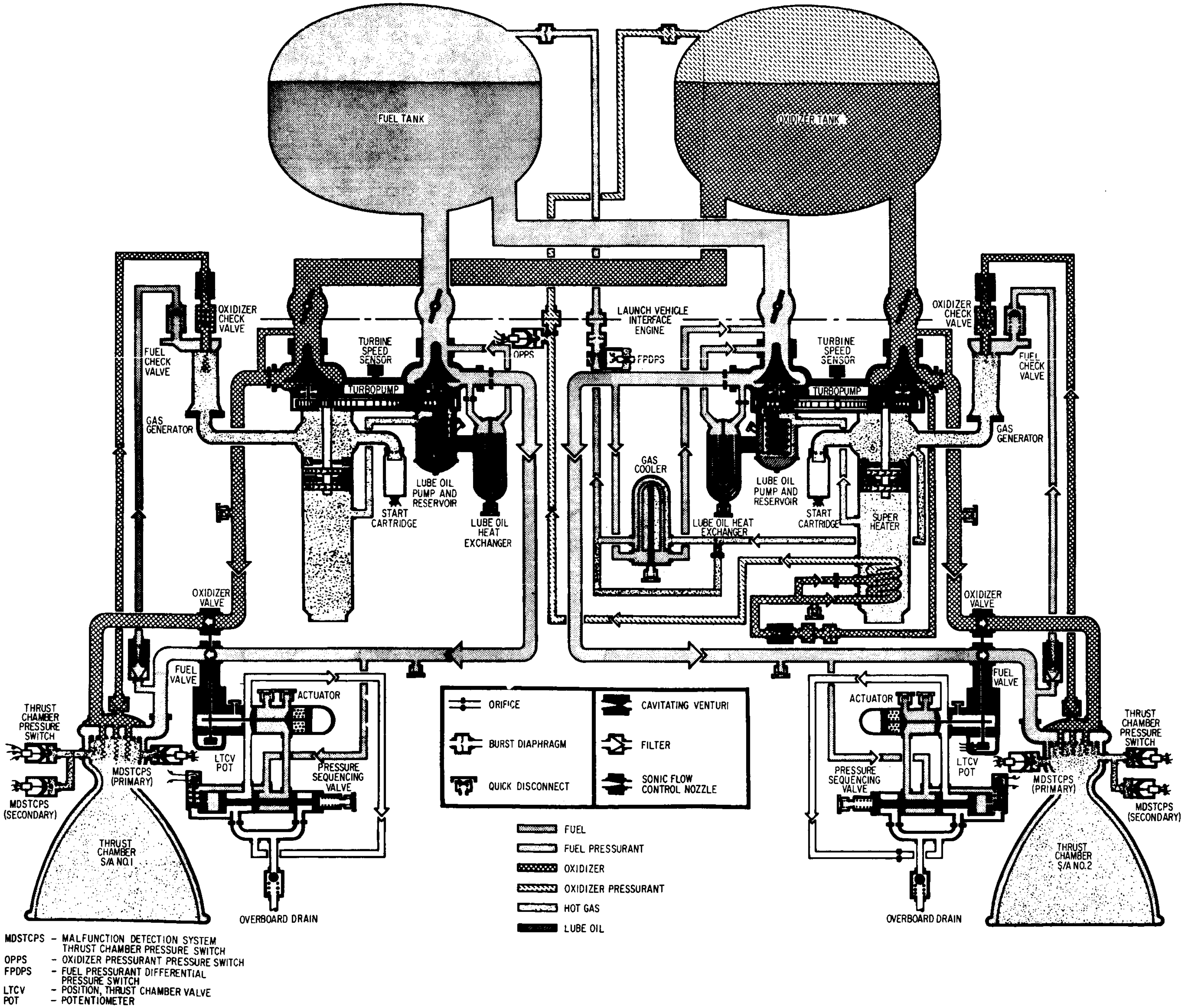


Figure II. C-8. Engine Stage I

(c) Turbopump Assembly. The turbopump assembly consisted of a fuel pump subassembly, oxidizer pump subassembly, two-stage turbine drive assembly and a gearbox assembly. The gearbox contained an integral oil pump, reservoir filtration system and lube oil cooler. These components were integrated into an assembly on the basic gearbox structure. The Gemini gearbox incorporated redesigned gears of 9310 alloy and SKF bearings exclusively.

(d) Solid Start Cartridge. The solid start cartridge was comprised of a steel combustion chamber which housed two concentric cylinders of ammonium nitrate propellant (AMR 2506). An ignition train assembly was mounted at the head end of the cartridge. Burning rate of the cartridge was controlled by a presized flow control nozzle, as well as by stringent maintenance of start cartridge temperature, through the use of cartridge thermocouples and temperature controlled air supplied to the external surfaces of the cartridge. A teflon burst diaphragm over the nozzle hermetically sealed the cartridge assembly. A pyrotechnic squib was installed in the cartridge and electrically fired to initiate the ignition train.

(e) High Pressure Propellant Lines. The high pressure fuel and oxidizer lines consisted of bolt flanges, tubing and three (3) 2-axis articulate flex joints. The flex couplings were arranged to allow for assembly tolerances and to allow thrust chamber flight control flexure obviating the need of rotary seals to achieve thrust chamber motion. All thrust chamber gimbal motion was taken between the chamber assembly and turbopump assembly. The turbopump assemblies were stationary to the frame and did not move during gimbal operation.

(f) Low Pressure Propellant Plumbing. The fuel and oxidizer lines connecting the turbopump to the tank outlets consisted of low pressure bellows assemblies, modified by POGO gear, from tank outlet to pump inlet. The bellows assemblies were for the purpose of absorbing relative deflections and to allow small misalignment of assemblies.

(g) Subassembly Two (S/A-2) Oxidizer Pressurization Super Heater Assembly. The super heater assembly was a cross flow tube-type heat exchanger located in the turbine exhaust in the subassembly two turbopump. Energy in the turbine drive exhaust gases was used to vaporize and disassociate liquid nitrogen tetroxide. The amount of pressurant gas was controlled by a cavitating venturi into the liquid side of the circuit. Energy in the gas was controlled by a back pressure nozzle in the gaseous side of the circuit. The oxidizer pressurant system included the associated plumbing.

(h) Subassembly Two (S/A-2) Fuel Tank Pressurant Heat Exchanger. The gas cooler assembly consisted of a bundle of U-shaped tubes enclosed in a cylindrical shell. Fuel was circulated on the interior side of the tubes as a heat exchange medium. Hot gas passed through the external passages around the tubing to the fuel autogenous system. The hot gas flow rate was controlled by a sonic nozzle in the plumbing circuit.

(2) Sequence of Operation

At a minimum of 30 seconds prior to engine start, propellant valves at the tank outlets were opened allowing propellant into the engines up to the thrust chamber valves (TCV's). The engines were self-bleeding by gravity head.

Each Stage I subassembly start was initiated by providing a 28-vdc signal to a redundant bridge wire squib in the solid start cartridge initiator. The initiator fired the ignition train which started burning of the solid propellant grain. Energy from the solid propellant caused acceleration of the turbopump assembly. As the turbopump assembly generated fuel pressure head, the pressure sequence valve shuttled at a pre-set pressure causing the thrust chamber fuel valve to be actuated open by fuel system pressure. The oxidizer valve was opened by mechanical linkage to the thrust chamber fuel valve. Propellant flow from the turbopump filled the volume in the thrust chamber assembly and ignition occurred upon contact of the propellants in the combustion chamber. As a result of thrust chamber combustion backpressure, propellant was supplied through gas generator feed lines downstream of the thrust chamber valve initiating the regenerative (bootstrap) cycle. As a result of the initiation of bootstrapping, the engine achieved rated thrust. Energy of the start cartridge was controlled to burn out immediately after the start of the bootstrap operation as the engine approached steady state operation. The thrust level was controlled by tuning presized cavitating venturis to achieve hydraulic equilibrium. No electrical power whatever was required by the engine during steady state operation.

During these initial moments of life of the engine, two Pilot Safety systems were monitored for engine parameters/characteristics in order to detect any abnormal operation. The first was known as the Pre-launch Malfunction Detection System and was operative until the vehicle was airborne. The second, known as the Malfunction Detection System, continued operation and surveillance throughout Stage I flight (see section II. C-11).

The PMDS was designed to monitor the Stage I autogenous system operation prior to release of the launch vehicle and to furnish go/no-go signals to launch control equipment in response to the conditions sensed in the fuel and oxidizer autogenous systems.

The MDS was a warning system that provided a visual cue to the astronauts in the spacecraft in the event of a launch vehicle subsystem malfunction which could possibly result in the failure of the flight mission. The Malfunction Detection Thrust Chamber Pressure Switch (MDTCPS), a part of the MDS package, also provided a signal to launch control equipment verifying attainment of proper thrust prior to vehicle release, as well as serving to initiate the controlled shutdown sequence whenever a pre-determined decay in thrust chamber pressure had occurred.

Engine shutdown was accomplished by either oxidizer exhaustion or fuel depletion. Although neither type of shutdown caused problems on the Gemini engines, the oxidizer exhaustion was more desirable from a hardware integrity standpoint and shall be used to further describe the shutdown sequence. As oxidizer was depleted the MDTCPS sensed a decay in thrust chamber pressure; the switch opened, causing a 28 vdc signal to be sent to the override solenoid of the PSV. The PSV override shuttled the PSV, causing fuel system pressure to be vented from the opening side and applied to the closing side of the actuator. As the thrust chamber valves closed, the propellant supply to the gas generator was terminated with a resulting thrust decay of each subassembly.

(3) Major Changes

As previously noted, the GLV engine configuration (-7 model) used the basic Titan II (-5 model) as a building block. Certain changes were made in the basic Titan II engine to adapt it to the critical uses of the Gemini Program. The changes were prompted by the need to man-rate the Titan II, and/or by experience gained in the early Gemini flights. Although many changes were incorporated during the Gemini Program, certain major changes warrant sufficient attention to be singled out for special mention.

(a) Stage I Gearbox. Failure of the idler gear which resulted in catastrophic failure of the turbopump was encountered. A detailed account may be found in Section II. E-4; however, let it suffice to relate that strengthened gears were incorporated on all Stage I gearboxes for GLV-2 through 12. A second failure mode which required resolution prior to manned flight centered around failure of the #6 bearing. Gemini's solution prescribed use of SKF bearings only and redesign of the turbine interchange labyrinth seal.

(b) Stage I Engine Frame. The Stage I Engine frame was qualified by similarity to the Titan II engine frame. However, because of certain, though minimal, modifications the capability of the Gemini engine frame under the conditions of firing and gimballing was verified during the Propulsion System Test Program.

(c) Flexible Lube Oil Cooler Coolant Lines. The fuel return line (from the oil cooler) fractured approximately 30 seconds after the start of a hot-fire test on Gemini Engine GLV-1011. The failure was attributed to excessive loads due to vibration and assembly distortion. A change to flexible inlet and outlet lines (for the fuel coolant) was incorporated to preclude further occurrences of this nature.

(d) Start Cartridges. Two problems were encountered which were resolved by incorporating non-interchangeable (between Stage I and Stage II) diaphragms and temperature conditioning of the cartridge itself. See Section II. E-4 for details.

(e) Propulsion System Test Program.[5] In Section II. E-4 the specific items under PSTP are discussed. This program was promoted to evaluate/demonstrate the satisfactory operation of Gemini unique components and requirements for the Stage I and Stage II propulsion systems.

(f) 0-5 Volt Instrumentation System. The Gemini Program utilized a 5 volt full scale telemetry system, whereas Titan II had used a 40 mv. full scale telemetry system. The higher output voltages from remote transducers on the Gemini system improved the overall signal-to-noise ratio and reduced the complexity of the multiplex/encoder. Details will be found in the Instrumentation Section of the report (reference II. C-8.).

(g) TCV Stress Corrosion. Failure of a Thrust Chamber Fuel Valve body prior to liftoff on one of the Titan II vehicles was attributed to stress corrosion. Investigation resulted in a change of heat treat from T-6 to T-73 on Gemini.

b. Stage II Engine

(1) System Description

The Stage II Engine System was in general identical in design concept to that of the Stage I Engine System. Figure II. C-9 is a block diagram schematic of the Stage II engine control system and Figure II. C-10 is a block diagram of the fuel pressurization system. The distinguishing features which differentiate the Stage II engine from the Stage I engine were the following:

1) Stage II engine components were a reduced thrust version of a Stage I subassembly.
2) Rated for altitude start and operation with a baffled injector.
3) The high expansion ratio chamber (49.2:1) was achieved by use of an ablative extension from 13:1 to 49.2:1.
4) Turbine exhaust gases were ducted through a swiveled nozzle assembly to provide roll control during Stage II powered flight.

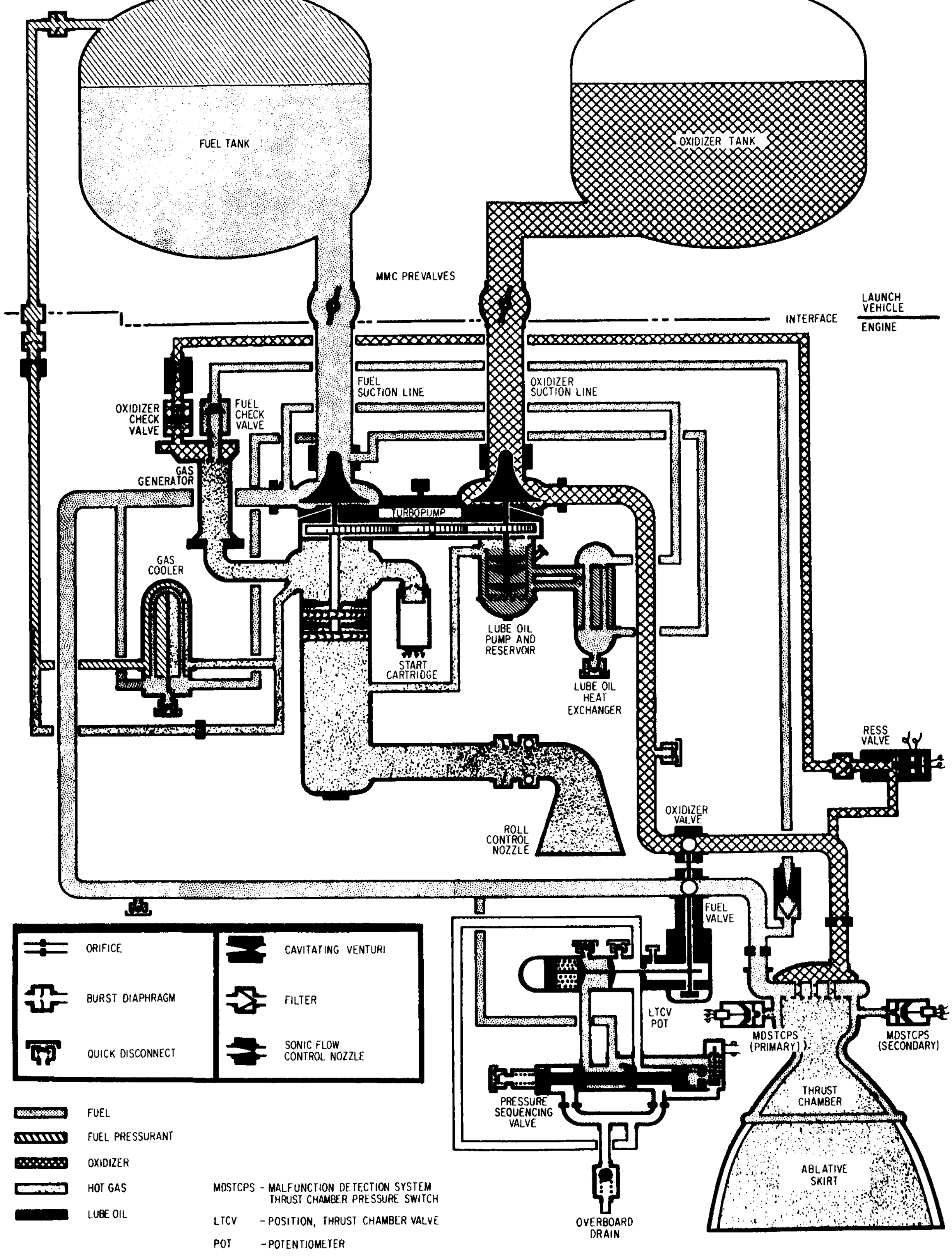


Figure II.C-9. Block Diagram Engine Control, Stage II

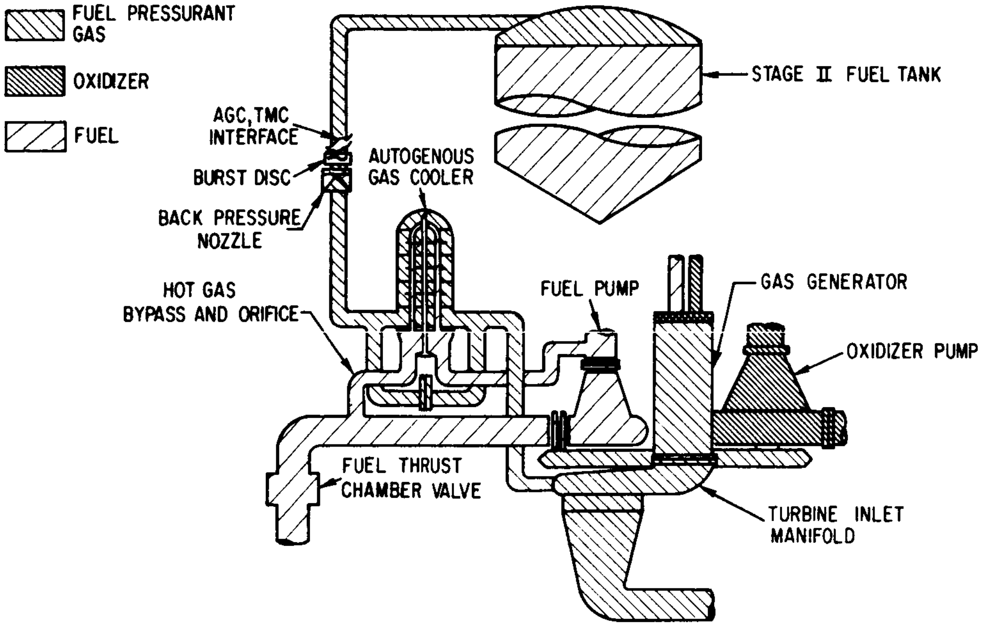


Figure II. C-10. Autogenous Pressurization System

5) Pressurization was required only on the fuel tank of the Stage II vehicle. Oxidizer tank pressurization was supplied by precharge and boiloff only.

6) The Stage II cartridge incorporated the same design features as the Stage I except that the propellant was a single, longer cylindrical section. It also included a conditioning system similar to Stage I.

7) Stage II had no thrust chamber pressure switch, TCPS.

8) Stage II incorporated a redundant shutdown system, the RESS, which is described later in this report.

9) Stage II did not have a PMDS system and the MDS switch monitored fuel injection pressure (MDFJPS) not chamber pressure as on Stage I. The MDFJPS was not a part of the shutdown circuitry as the Stage I MDTCPS.

(2) System Operation

The Stage II engine sequence was identical to the Stage I engine subassembly. The same signal which shut down the first stage, signaled the second stage ignition. Staging was accomplished by the "fire-in-the-hole" concept. Stage II thrust buildup caused separation of the vehicle stages. As will be noted further on, Gemini had a unique feature of redundant shutdown capability on the Stage II engine. This redundant shutdown capability was achieved by firing a squib actuated valve in the gas generator oxidizer feed circuit simultaneous with the signal to the thrust chamber valve.

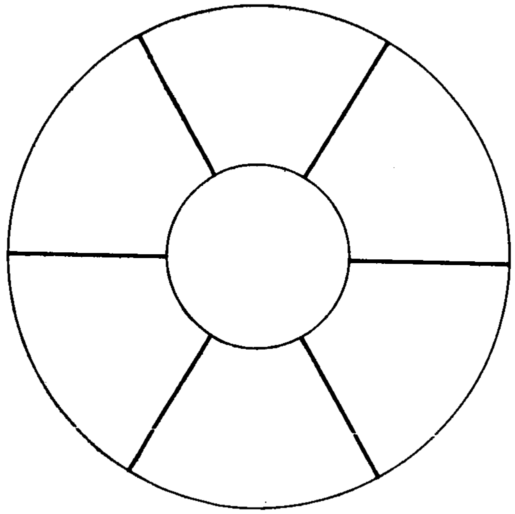
(3) Major Changes

This section reports on major components or systems which were unique to the GLV Stage II engine compared to the Titan II Stage II engine. They were also unique to GLV Stage II compared to GLV Stage I unless otherwise noted. These systems were developed and incorporated due to Gemini program requirements or as the result of problem solutions.

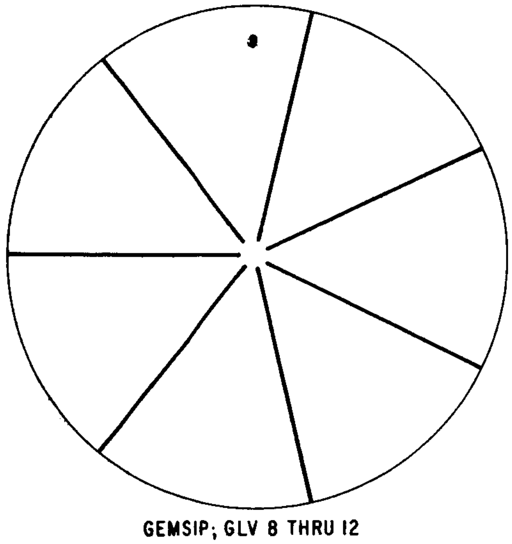
(a) Redundant Engine Shutdown System (RESS). A redundant engine shutdown capability was added to minimize the possibility of spacecraft overspeed due to a failure of the basic Thrust Chamber Valve/Pressure Sequence Valve. This redundant system was developed under the Augmented Engine Improvement Program and consisted of a squib actuated valve in the oxidizer bootstrap line which was activated by the same signal sent to the PSV. This system was incorporated on GLV-3 and subsequent vehicles.

(b) Fuel Injection Pressure Switch. The Malfunction Detection System pressure switch initially monitored chamber pressure as on the Stage I engine. However, the initial Titan III Stage II engine which utilized a similar configuration switch encountered combustion instabilities on two acceptance tests. The analysis indicated the most probable triggering mechanism was detonation in the sensing tube, and the magnitude a function of line volume. As GLV corrective action, the MDS switch was re-located to sense fuel injection pressure, which reduced sensing tube volume to a minimum with only a chamber pressure transducer on the combustion chamber. This action was taken because the standard Titan family injector (which had a higher susceptibility to instabilities) was on the first seven GLV's. The relocation was effective on GLV-2.

(c) GEMSIP Injector. A dynamically stable injector was developed to increase the combustion stability margin. The new injector, GEMSIP, was incorporated on GLV-8 and subsequent vehicles. A comparison of the Titan II and GEMSIP baffle configuration is shown on Figure II. C-11. The details of the Gemini Stability Improvement Program are delineated in the final report for that program.[6]



BASIC T-II; GLV 1 THRU 7



GEMSIP; GLV 8 THRU 12

Figure II. C-11. GLV Stage II Injector Baffle Configurations

(d) Start Cartridge Temperature Conditioning. As previously described, a solid start cartridge conditioning system was also incorporated on the Stage II engine system.

(e) 0-5 Volt Instrumentation System. The 5 volt flight instrumentation system was also incorporated on the Stage II engine system.

The same configuration ablative skirt was utilized of the GLV engine but due to longer burn durations a program was conducted to demonstrate the capability to withstand these durations.[7] To insure that the incorporation of the GEMSIP injector was also compatible with these long durations, additional testing was conducted.[8]

4. VEHICLE PROPULSION SYSTEM

a. System Description and Design Concept

The propulsion system of the GLV was a direct adaptation of the Titan II propulsion system. Some deviations or additions were made to meet specific Gemini requirements. The launch vehicle utilized the Aerojet-General YLR 87-AJ-7 liquid propellant rocket engine for Stage I and the YLR 91-AJ-7 liquid propellant rocket engine for Stage II respectively. These engines burned storable hypergolic propellants, nitrogen tetroxide and UDMH-hydrazine blend. The propulsion systems included a propellant feed system and a tank pressurization system. The propellant feed systems for first and second stages contained the tanks, feedlines and associated valving necessary to store propellants in the vehicle and supply these propellants to the rocket engines. The first stage propellant system also contained hydraulic oscillation suppression devices necessary to eliminate longitudinal oscillation instabilities (POGO) caused by closed loop coupling between the structural resonances and the propulsion system. The tank pressurization system was used to provide proper propellant pressure to the engines during start and flight.

b. Component Configuration

The propellant feed systems, transferring propellants from the propellant tanks to the engine, were designed to produce a minimum pressure loss at the design flow rates. The Stage I and Stage II propellants were stored in ten-foot diameter aluminum tanks which also formed the primary launch vehicle structure. Oxidizer was fed to the engine turbopump inlets by an aluminum feedline, passing through a conduit in the fuel tank. For Stage I, the oxidizer feedline was divided in the engine compartment to feed each engine subassembly. The fuel tank outlets were located immediately above the engine pump inlets.

The propellant tanks had anti-vortex and anti-slosh baffles. In addition, the oxidizer tank outlets were designed to prevent loss of energy in discharging the propellant from the tanks.

Propellant prevalves were utilized in the propellant feed system in a manner similar to T-II. The prevalves enabled holding propellants above the engine pumps and thrust chamber valves until opening was necessary for terminal countdown 45 seconds prior to Stage I engine start for fuel and 60 minutes prior for oxidizer to enable the charging of the POGO oxidizer standpipe. The launch vehicle was filled and drained of propellants through manual disconnects located at the prevalves.

Liquid level sensors were installed in the fuel and oxidizer tanks to facilitate propellant flow rate and outage measurement for performance calculation. The level sensors were initially redundant to assure adequate performance data. The redundant high position sensors were removed on GLV-5 and subsequent vehicles as a weight saving item. GLV's 9, 10, 11 and 12 were flown without redundant outage sensors for the same reason.

Propellant temperature probes were provided in all propellant tanks to ensure that propellant temperature at liftoff did not exceed the allowable limit under the existing engine operational limits and vehicle performance margins for each mission.

A propellant fill and drain system was used to transfer propellants from the AGE to the vehicle tanks. The ground system included transfer pumps, and instrumentation to assure that propellants were transferred to the vehicle at proper temperatures and in the proper quantities. A schematic of the propulsion feed systems is shown in Figure II.C-12 and II.C-13. A secondary propellant conditioning, storage and transfer system was developed for GLV-6 and subsequent launches in an effort to reduce vehicle recycle time in the event of a launch scrub (See Section II. C. 19, PTPS).

From the initial lock-up condition prior to launch until engine shutdown at burnout, the pressurization system was designed to provide adequate propellant tank ullage pressure to satisfy (a) minimum NPSH requirements for the engine driven propellant pumps and (b) minimum launch and inflight structural requirements of the tanks. Simultaneously, the pressurization system was not supposed to exceed pressures as defined by structural limitations.

Initial pressurization of all propellant tanks was achieved with a charge of nitrogen gas regulated to a predetermined level. In flight, tank pressurization was provided to both fuel tanks by utilizing cooled rocket engine turbine exhaust gases. The Stage I oxidizer tank was pressurized by vaporized oxidizer from the engine. The Stage II oxidizer tank was pressurized in flight by the initial nitrogen lockup pressure and supplemented by vaporization of the tanked oxidizer. Burst discs were installed in each tank pressurization line to facilitate prelaunch checkout of the vehicle tankage and propulsion system.

The inflight pressurization subsystem for Gemini was essentially the same as that used in Titan II. Exceptions were as follows: (a) The pressurant flow to the Stage I fuel tank was increased by a change in the engine flow control orifice size. (b) Stage I oxidizer pressurant flow rate was increased through an orifice change and the tank "lockup" pressure was increased.

A revised tank pressure control system known as the tank pressure topping system was designed for the GLV. This system incorporated a pressure supply and vent line which was in parallel with the main pressure supply line available to each tank. When the main lines were disconnected, pressure control was retained through the smaller pressure topping lines which remained connected until liftoff. Lanyard operated pull away couplings were used to separate the GLV from the AGE and effect a seal at the vehicle skin line. An airborne, solenoid-operated, normally open valve was located upstream of the disconnect coupling on the GLV. This valve provided seal redundancy in flight and a method for leak testing during the prelaunch countdown. The tank topping system is shown in Figure II.C-14.

One of the big differences between Titan II and the GLV in the propulsion system was the use of suppression devices to control the longitudinal oscillation that occurs during first stage flight. The POGO suppression devices (discussed further in the next section) consisted of tuned resonators inserted into the engine feed lines just upstream of the first stage pumps. These devices were tuned to provide attenuation of pressure oscillations in the frequency band of the structural first longitudinal mode. In the oxidizer lines the resonator was a standpipe, while in the fuel lines a tuned mechanical accumulator was used. (Figure II.C-15)

The oxidizer standpipe worked on the principle of using an entrapped gas bubble to provide a soft spring for the oxidizer mass in the standpipe to act upon. The energy due to pressure oscillations could be transferred to this spring mass system at the desired frequency by proper choice of the volume of the entrapped gas bubble. The fuel surge chamber or accumulator used a helical spring and piston to perform the same job. The spring together with the piston and fuel mass in the accumulator acted to provide the desired resonance.

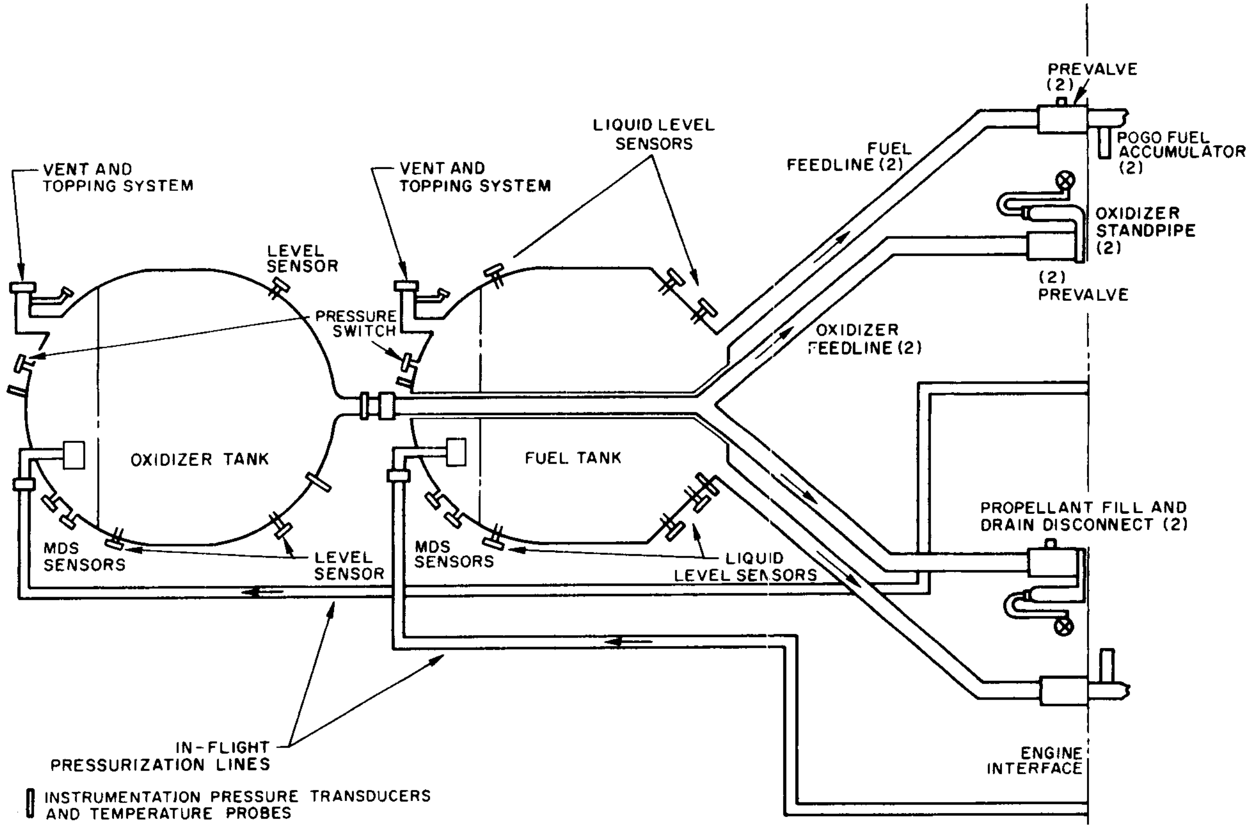


Figure II. C-12. Stage I Propulsion System

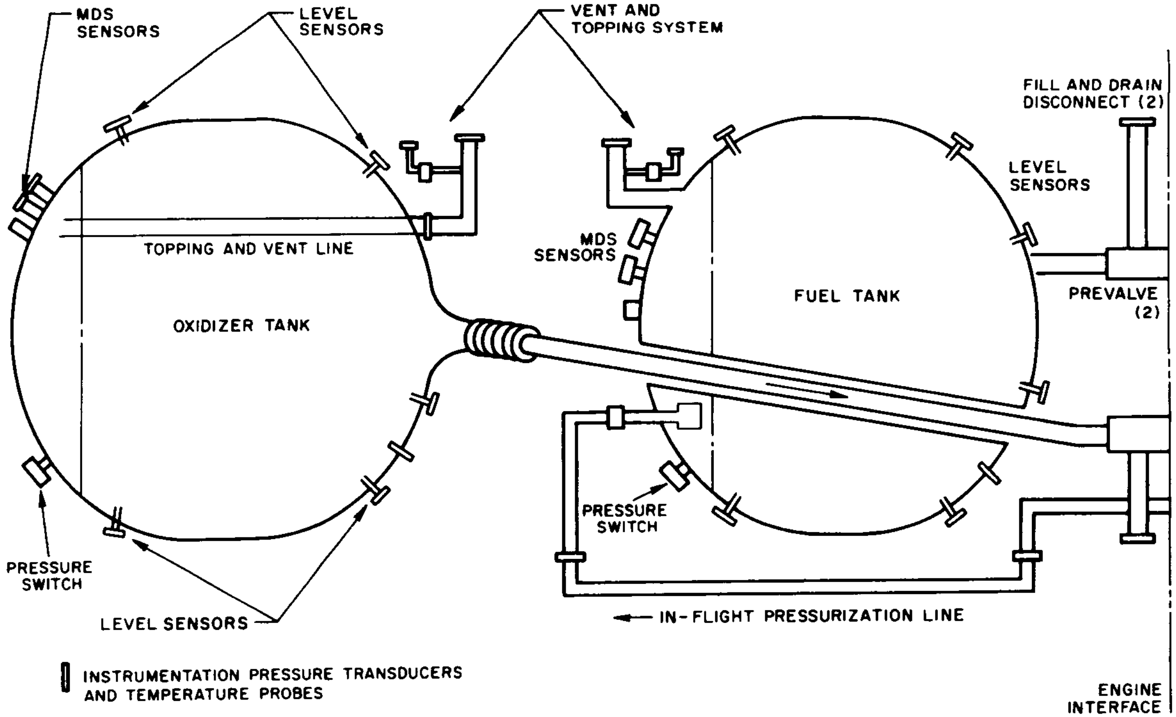


Figure II. C-13. Stage II Propulsion System

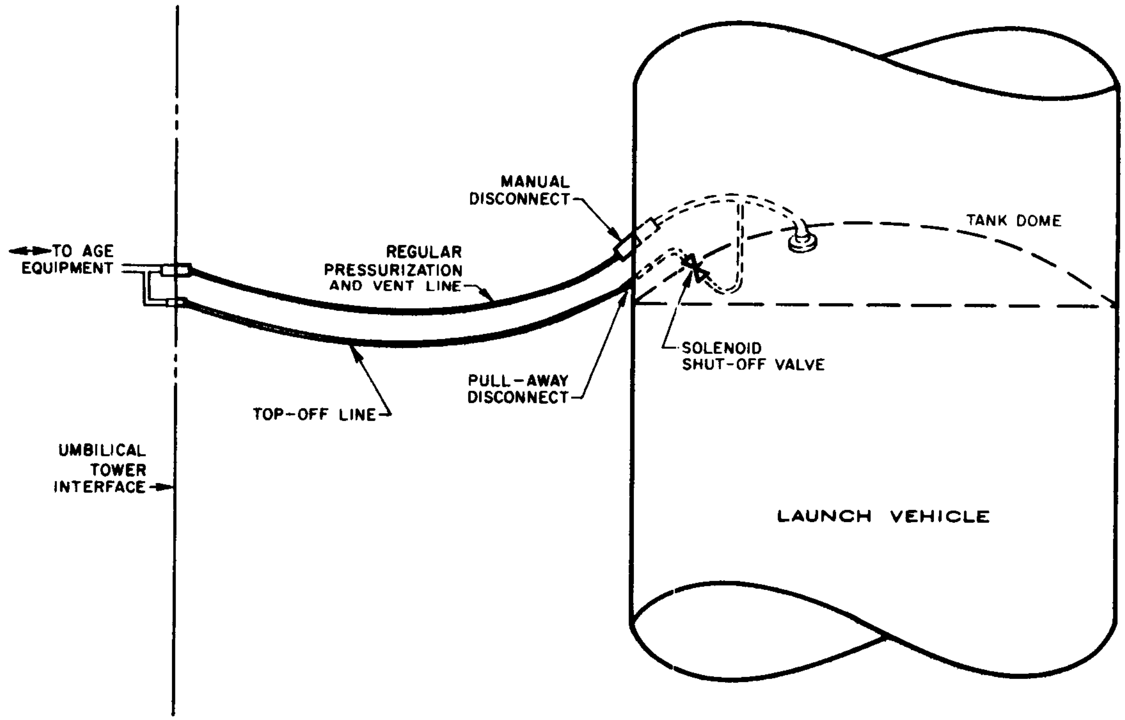


Figure II. C-14. Tank Pressurization Top-Off System - Schematic

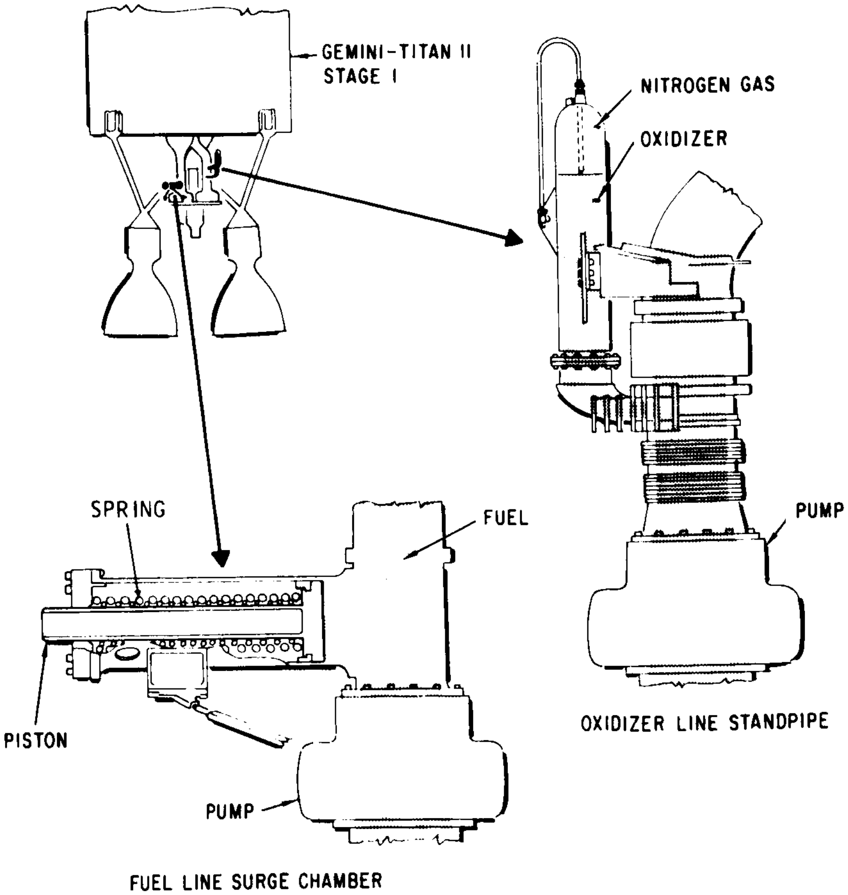


Figure II. C-15. "POGO" Suppression Equipment

c. Component History

(1) Development History

The Gemini development of the propulsion system was conducted in the Propulsion System Test Program (PSTP) at Aerojet-General, Sacramentro, 28 January 1963 to 2 March 1964. The purpose of the test program was to check and verify the operation of that portion of the Titan II/Gemini propulsion system peculiar to the GLV.

The PSTP results demonstrated the ability of the Gemini propulsion system to meet its design requirements. Tests were run to determine whether reduced fuel and oxidizer tank ullage volumes had adverse effects on engine starting characteristics. Results demonstrated that reduced ullages did not cause violation of engine net positive suction head requirements. Validity of the analytical technique to predict pressurization system performance was also satisfactorily demonstrated. The effects and calibration of engine performance with cold propellants were also demonstrated in these tests.

Because of the performance requirements imposed on the launch vehicle, level sensor performance and reliability became significant factors. The flight test failures on Titan II and the problems encountered in attempting to qualify the sensors resulted in a decision to change from the Titan II level sensor to a Bendix unit for Gemini.

"Piggy Back Testing" was used to fly GLV hardware on Titan II R&D vehicles. Nineteen Bendix level sensors were flown on Titan II flights N3A and N33. Flight uncovering times were compared with readings from the Titan II level sensors in equivalent tank locations. The agreement between all pairs of sensors was within tolerance. On these flights the first indications of fuel sensor uncovery and subsequent recovery were noticed. This condition was verified on GT-1, and subsequent corrective action shielded the fuel sensors from pressurization gas condensation.

Development history problem areas were primarily concerned with the propellant shutoff valves (prevalves), POGO hardware and modified tank pressurization hose assemblies.

Titan II experienced prevalve opening problems which prompted changes in the prevalves used for the GLV. The valve shear disc (hermetic seal) was eliminated, the butterfly release pin was shortened and modified to remove the ramp opening action and the release pin material changed. The valve butterfly seal material was changed to make it compatible with GLV storage requirements. Titan II also experienced several prevalve position indicator switch failures. However, studies showed the GLV switch to be more reliable than the Titan II switch due to different installation and checkout techniques.

Several Titan II configuration flex hoses were modified for GLV for use in the tank pressurization system. During evaluation testing of these hoses, cracks were found in the weld between the vent line boss and the hose. Gussets were added between the boss and the line to strengthen the weld area.

The first Titan II flights showed a very high "g" level of oscillation in the first longitudinal mode in the time period preceding booster engine cutoff. The "g" levels were too high for the equipment aboard the Titan II, much less a pilot. Titan II pursued the problem and found that an adequate reduction in level for a weapon system could be achieved by simply increasing fuel tank flight pressures. The reduced levels of oscillation resulting from this change were not, however, compatible with a manned requirement. A special program was established to evolve a fix that would reduce the levels to the $\pm$ .25 g requirement for a manned vehicle. The proposed fixes of detuning the feed system, the standpipe and fuel accumulator, plus providing an attenuation of pressure oscillation frequencies in the structural band, were tested and demonstrated successfully in three Titan II flights, N-25, N-29 and N-31. These flights verified the major portion of the initial Gemini redesign of the Titan II experimental hardware. The N-31 flight (flown with reduced fuel tank pressure) also verified that the hardware was effective at a reduced fuel tank pressure level, which was shown to be a strong parameter in controlling the levels of oscillation without the fixes.

Considerable effort was directed toward the development of a remote charging system (AGE) which was used to tune the launch vehicle oxidizer standpipes. The system was developed to provide greater flexibility in mission planning. The original manual charging technique involved opening prevalves and dropping propellant relatively early in the countdown thus exposing the engine to oxidizer. This exposure necessitated a greater recycle time in the event of a launch scrub after prevalve opening. The concept of the remote charging system provided increased safety in the charging operation and greater flexibility when the charging operation was performed because it was controlled from the blockhouse. It was proven through development test and subsequent flight test experience that the remote charge system provided a more consistent charge than the manual system.

d. Qualification History

Components in the GLV which were like Titan items were qualified by similarity to their respective Titan II counterpart. Qualification problem areas existed with the POGO fuel accumulator position potentiometer, the tank topping system vent disconnect, the tank topping system solenoid valve, POGO airborne ball valve (shutoff valve) and the propellant tank level sensor.

The basic POGO hardware, oxidizer standpipe and fuel accumulator, were qualified by successful flight test on Titan II flights N-25, 29 and 31. During component vibration testing of the POGO fuel accumulator rotary potentiometer, the potentiometer mounting screws loosened and the negator spring which connects the potentiometer to the accumulator piston slipped off the rim of its potentiometer attach wheel. Corrective action was taken to strengthen the potentiometer mounting provisions and these failure modes were eliminated. The potentiometer failed the propellant compatibility test due to penetration of oxidizer through the front bearing of the pot. Corrective action consisted of installing an O-ring seal external to the bearing area.

Problems were encountered with the remote operated airborne ball valve in that the ball rotated to a slightly open position under vibration environment. The problem was solved by incorporating a spring detent mechanism to positively lock the ball in the fully closed position.

Excessive connecting load problems were encountered with the tank topping system disconnect during qualification temperature testing. Corrective action included additional lubrication and tightening of valve and seal cleaning requirements. Sand and dust tests resulted in entry of contamination into the disconnect valve. Corrective action added seals to the open areas.

The tank topping system solenoid valve failed to pass qualification vibration tests by exhibiting sporadic leakage. Corrective action consisted of changing the valve design from a normally closed to a normally open valve.

The Bendix level sensors were subjected to a complete qualification test program. No significant problems were encountered with the component during this program.

e. Production History

The production history of the GLV propulsion system documents problem areas in the following components: POGO fuel accumulator position potentiometer, pressurization system pressure transducer, pressurization system pressure switch, the fill and drain disconnect, prevalves, the liquid level sensors, and tank topping system disconnects. Many of these problems were instrumentation type and thus did not affect the man-rating of the vehicle propulsion system.

(1) Fuel Accumulator

During development, the POGO fuel accumulator rotary potentiometer experienced problems with broken lead wires. Heavier gage wire and better support was provided for these lead wires and the lead wire connection to the potentiometer internal wiring was relocated inside the potentiometer housing. Improvements were made in the accumulator piston seal and in a bushing material to substantially reduce the friction involved in piston cycling. The piston shaft was hardcoated to prevent wear on the shaft from generating particles within the accumulator. During the flight of GT-2, a loss of potentiometer output was in evidence after approximately 90 seconds of flight. The problem was found to be caused by the recirculation temperature environment. A protective shield was provided around the potentiometer and the end of the fuel accumulator. The shield reduced the temperature environment at the potentiometer to acceptable levels and also improved the dynamic operation of the accumulator piston.

(2) Tank Pressure Transducer

Installation problems were discovered with the tank pressurization system pressure transducer on early launch vehicles. Damage resulted to the transducer due to improper torquing of the unit. Corrective action consisted of adding notes to installation drawings and procedures to flag the proper wrench flats for torquing the transducer.

(3) Tank Pressure Switches

The pressurization system pressure switches exhibited drift following exposure to propellants and long storage. The problem was observed beginning with GLV-5. Corrective action consisted of removing the switches from the launch vehicle subsequent to subsystems test at ETR to verify proper operation. All old switches in the field were checked for proper calibration and operating switch points.

(4) Fill and Drain Disconnect

The propellant fill and drain disconnect experienced many early production problems such as poppet leakage and connecting thread galling. These problems did not present a flight problem, but could have resulted in countdown delays. Corrective action included thread lubricant changes, thread plating process changes, polarizing ring material change, thread protector modifications, improved contamination control, and poppet design changes. Late in the program, allowable vapor leakage rates were established at levels which could be sealed by installation of the disconnect cap thus alleviating the persistent problem of small vapor leaks past the poppets.

(5) Prevalves

Prevalve butterfly seal leakage was reported on GLV-1 and 2 tanking tests. These leaks were found to be minor and caused by permeation of vapors through the seal upon initial loading. Inoperable (dummy) prevalves were installed on GLV-3 and subsequent vehicles for tanking tests. Late in the program several flight prevalves failed their gas leak checks after installation in the launch vehicle. These problems were due to deformed butterfly seals and no corrective action was taken because a sufficient number of acceptable prevalves remained to complete the program.

Many flight prevalves were found to have been delivered in 1965 to ETR with contamination evident in the valve after unpacking. As a result improved packaging and unpacking procedures were instituted for both the vendor and ETR personnel to eliminate the possibility of introducing contamination in these operations and all suspect valves were returned to the vendor for recleaning.

During flight preparations at ETR for GLV's 11 and 12, problems with the "dummy" prevalves (used for shipment) were encountered. The problem was brought to light by the discovery of aluminum chips in the turbopump feed lines. Handling and installation procedures were improved for the dummy prevalves.

(6) Level Sensors

At various times in the program, the level sensors experienced electrical and mechanical problems due to defective solder connections, defective resistors, plating imperfections and pin corrosion. The problem areas were of a quality nature and appropriate measures were taken to tighten the vendor's quality control and to improve test procedures. Connector pins were gold plated. The GLV-1 flight exhibited fuel level sensor recover and uncover characteristics similar to the level sensors installed on Titan II piggy back flights. Shields were added to the fuel sensors on GLV-2 and the problem was eliminated with the exception of the one sensor at the Stage II fuel tank high level position. This location continued to show recover characteristics on several flights, however, adequate performance data was continuously obtained. The engine shutdown capability by means of level sensors was eliminated and the sénsors were used only as instrumentation on all flights.

The presence of minute cracks was discovered in the ceramic which bonded an optical prism to the sensor housing. Because of the possiblity that the ceramic would fail and pieces be allowed to enter the propellant, a teflon retainer was designed to retain any loose particles of bond material. There was no loss of structural integrity as a result of the cracks.

(7) Pressurization System Disconnect

Problems were encountered during GT-4 and GT-10 launches with pressurization system disconnect separation. The GT-4 anomaly resulted in the disconnect halves being ripped apart while the GT-10 anomaly resulted in tearing the pressurization line loose from the umbilical tower and all parts flying with the vehicle. The disconnect lanyards were broken in both cases and both anomalies resulted in slight vehicle roll disturbances. The first occurrence was thought to have been caused by a jammed ground-half disconnect collar and corrective action resulted in removing the dust cap and other interfering cables that could cause jamming from the unit, prior to launch. The latter anomaly was considered to have been a lanyard rigging problem and this condition was remedied by improved rigging and handling procedures.

(8) Oxidizer Standpipe Charging System

During GT-5 flight, longitudinal oscillations (POGO) in excess of the 0.25 g zero to peak allowable occurred late in Stage I flight. The spectral analysis of oxidizer suction and standpipe pressure fluctuations revealed that the standpipes were charged approximately 10 percent at vehicle liftoff. The observed flight disturbances correlated with analytical stability analysis for a vehicle without standpipes, which shows that POGO effects of the character of an unmodified vehicle occurs, but with reduced severity. Studies and tests indicated the lack of capacity of the manual charging tool, which was used to charge the GT-5 standpipes. The manual tool, which was used as a backup to the remote charging system, was redesigned to provide more than sufficient charging capacity. Operational procedures for both the manual and remote charging units were revised to assure adequate charging time and regulated pressure under any operating condition. Subsequent vehicle standpipes were charged with the remote unit with no indication of problem.

f. Conclusions

Most problems concerned with the vehicle propulsion system were in the nature of handling, quality, and reliability. There was a significant downward trend in number of problems as the program progressed. GLV flight test results indicated satisfactory hardware operation with measured performance being very close to predicted.

5. HYDRAULIC SYSTEM

a. System Description and Design Concept

The launch vehicle Stage I engine had two thrust chambers, each chamber being gimballed through two degrees of freedom. Both chambers could be moved simultaneously for control. Thrust chamber motion was controlled by electrohydraulic servo-actuators responding to electrical commands from the flight control system. Each thrust chamber was controlled by two actuators, one in the pitch plane and one in the yaw plane. Figure II. C-16 illustrates a schematic of the Stage I hydraulic system which consisted of two independent power subsystems designated "primary" and "secondary." The four first stage control actuators were constructed as independent tandem cylinder units with a primary servo-actuator section and a secondary servo-actuator section. Independent primary and secondary power systems supplied fluid to their respective servo-actuator sections. The independent, redundant first stage hydraulic systems were designed specifically for the Gemini Launch Vehicle.

The launch vehicle Stage II rocket engine had a single thrust chamber, gimballed to provide control in the pitch and yaw planes. Roll control was provided by an auxiliary nozzle which utilized the hot turbine exhaust gases. This nozzle was placed off-center in relation to the vehicle center line and was swiveled about an axis perpendicular to the vehicle center line. As on Stage I, thrust chamber and roll nozzle motion were produced by electro-hydraulic servo-actuators controlled by signals from the flight control system. Figure II. C-17 shows a schematic of the Stage II hydraulic system.

b. Component Configuration

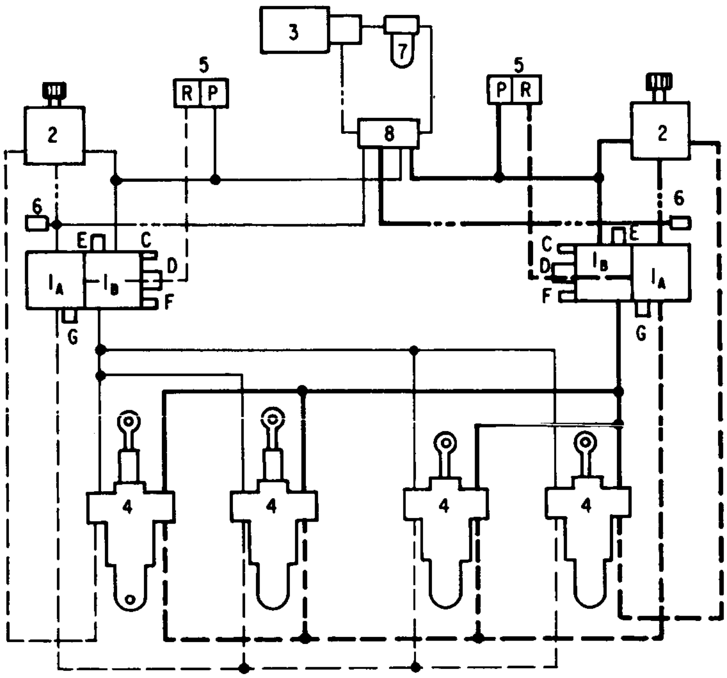
(1) Stage I System Components

The airborne power sources for the primary and secondary hydraulic subsystems were provided by two identical pressure-compensated, variable volume pumps which maintained a supply pressure of 3050 psi. The pumps were driven from accessory drive pads on the Aerojet engine. The primary pump was driven by subassembly two, the secondary by subassembly one. The output of each pump was supplemented by a piston-type gas accumulator to permit servo-actuator response to transient demands which exceeded the pump outlet flow capacity. The reservoir which supplied the system utilized "bootstrapping" for pressurization. Pump discharge pressure was applied to a small piston which was mechanically connected to the reservoir piston, and thus pressurized the pump supply.

As indicated on the schematic (Figure II. C-16), the reservoir and accumulator of each subsystem was combined into a single package along with instrumentation necessary for subsystem checkout and performance evaluation. The pumps and accumulator were procured to Gemini drawings with tighter contamination requirements than Titan II.

The hydraulic ground service disconnect on each subsystem was designed as a dual unit incorporating both pressure and return disconnects in a coaxial arrangement. This feature positively prevented application of ground hydraulic power through the pressure line unless the return line was also connected. The disconnects were the same as T-II, but procured to Gemini drawings and standards.

The primary and secondary subsystems were designed to be completely serviced, tested, and sealed in operational readiness early in the countdown preceding engine start. To enable pre-launch flight control and hydraulic system tests, an electric motor-pump was provided with a selector valve for operating either the primary or secondary system. Both the pump and selector valve were operated from the AGE. In addition to its use for ground checkout, the motor-pump was used to pressurize



COMPONENTS

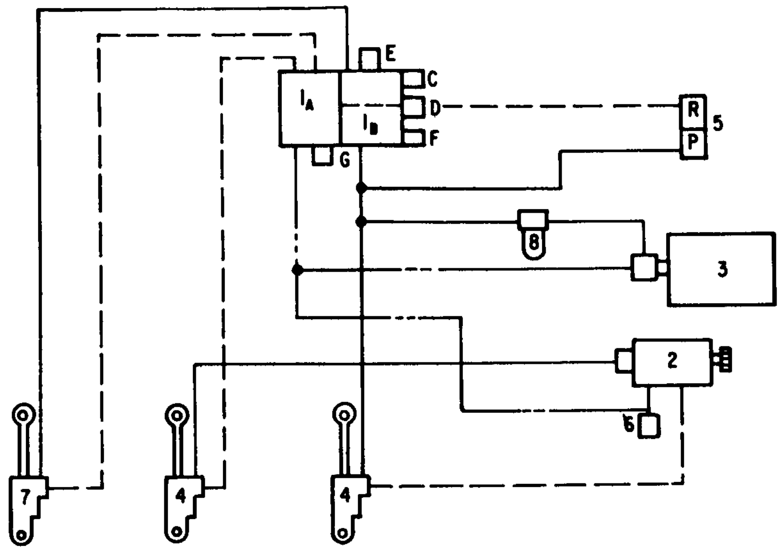
I_A – RESERVOIR
I_B – ACCUMULATOR
I_C – DUAL PRESSURE SWITCH
I_D – AUTOMATIC FILL VALVE
I_E – PRESSURE TRANSDUCER
I_F – N_2 CHARGE VALVE
I_G – TEMPERATURE SWITCH

2 – ENGINE PUMP
3 – ELECTRIC MOTOR PUMP
4 – TANDEM SERVO ACTUATORS
5 – COAXIAL DISCONNECTS
6 – TEMPERATURE TRANSDUCER
7 – FILTER
8 – SYSTEM TEST SELECTOR VALVE

LEGEND

PRIMARY PRESS.
PRIMARY RETURN
PRIMARY SUPPLY
SECOND. PRESS.
SECOND. RETURN
SECOND. SUPPLY

Figure II. C-16. Gemini Launch Vehicle Stage I Hydraulic System



COMPONENTS

1_A – RESERVOIR
1_B – ACCUMULATOR
1_C – DUAL PRESSURE SWITCH
1_D – AUTOMATIC FILL VALVE
1_E – PRESSURE TRANSDUCER
1_F – N_2 CHARGE VALVE
1_G – TEMPERATURE SWITCH

2 – ENGINE PUMP
3 – ELECTRIC MOTOR PUMP
4 – ENGINE ACTUATOR
5 – COAXIAL DISCONNECT
6 – TEMPERATURE TRANSDUCER
7 – ROLL NOZZLE ACTUATOR
8 – FILTER

LEGEND

——— PRESSURE
– – – – RETURN
—– – – – SUPPLY

Figure II. C-17. Gemini Launch Vehicle Stage II Hydraulic System the primary hydraulic subsystem during the engine start sequence, providing stabilization and control of the engine thrust chambers. The system selector valve was unique for Gemini.

Each of the primary and secondary sections of the Gemini-designed tandem servo-actuator were complete integral electro-hydraulic servo systems whose position output was proportional to their respective electrical inputs from the flight control system. Only one of the redundant subsystems on Stage I was used to control the engine thrust chambers at any given time. Each actuator contained a two-position transfer or "switch-over" valve located between the servo-valves and their respective cylinders, which selected the mode of operation. The switchover valve permitted free flow across the piston of one system and allowed the servo-valve to control the other. The switchover valve was spring-loaded toward the secondary position; however, the primary pressure was used to override the spring to maintain the valve in the primary position prior to engine ignition. With loss of primary system pressure or a signal from the MDS, the valve switched over, putting the secondary system in control. A pressure switch sensed the switchover and sent a signal back to the MDS indicating the switchover and ensuring the switchover of the rest of the actuators to the secondary system. To ensure that a valve or electrical signal failure subsequent to switchover would not result in an actuator switching back to the primary system, the pressure in the secondary system was used to lock the valve in the secondary position. The primary servo-actuator section also contained a snubbing and cylinder pressure-limiting device to absorb end-of-stroke dynamic forces and to relieve the actuator of excessive externally applied loads. Flow limiters were used to limit maximum actuator rate of travel in both primary and secondary portions for vehicle flight control stability and load control.

(2) Stage II Components

The Stage II system was similar in design and operation to the primary or secondary system, but was not redundant. The Stage II hardware was identical, except for the use of 15 micron actuator assembly filters, to that used on Titan II, but procured under Gemini drawings with some upgraded quality control requirements.

Table II. C-1 lists the GLV hydraulic system components and their similarity to Titan II hardware.

c. Component History

(1) Development History

(a) ASFTS Testing. The redundant system concept was verified in the Airborne Systems Functional Test Stand (ASFTS). Further discussion of ASFTS appears in the next section and under Section II. C-16. All the systems components and interconnecting plumbing were installed as they would be in the vehicle with the exception of the engine driven pumps which were mounted on Varidrives. Frequency response, switchover performance, thermal rise rates, pressure response, the flight controls and MDS interfaces and other pertinent factors were evaluated on the hydraulic system. The ASFTS was also useful in developing proper servicing techniques and solving test equipment and procedural problems.

The tandem actuator was installed on the Gemini test engines at Aerojet during the final series of Stage I development tests. Only the actuators were installed with the hydraulic supply system simulated by test stand equipment. The purpose of the testing on the engine was to gain experience under realistic vibration conditions, to demonstrate the switchover capability under firing loads and to verify the adequacy of the force limiters at engine start. No problems were experienced with the actuators during the engine runs. The actuators showed similar behavior to the T-II actuators during engine start and performed their switchover function properly.

One of the facets explored during development of the Gemini hydraulic system was the control of built-in contamination in component parts. Contamination requirements were more stringent than for Titan II or general system requirements. In the vendor's facilities, special procedures and tooling were instituted to achieve these requirements. The achievement of cleanliness within the system gave a large improvement in its reliability.

Only two components in the GLV hydraulic system were developed exclusively for the Gemini program. These were the Stage I tandem actuator and Stage I test-selector valve. The development history of "like Titan II" hardware showed no problem areas of special concern to the GLV. Tighter Gemini cleaning procedures were instituted on all Titan II qualified components.

The system test-selector valve was designed and manufactured without engineering evaluation testing. The qualification testing, which was designed to include evaluation tests, produced no major failure modes or performance discrepancies.

Several tandem actuator failure modes were demonstrated during component development. The actuator assembly force limiters exhibited oscillation and subsequent structural failure. The force limiter was redesigned from a spring-loaded ball valve to a sliding sleeve poppet design. The secondary system pressure switch showed premature life failures due to bellows failures. Subsequent changes were made to reduce pressure surges to the bellows and obtain a better bellows material. Changes were made in the actuator piston velocity requirements which resulted in modifications in the flow limiters. The actuator initially exhibited poor dynamic characteristics in the form of excessive phase lag. The gain of the second stage valve was increased by softening the spool centering springs, and this corrected the dynamic characteristics. The position switch used for ground checkout of the flight control system exhibited premature wearout. This was caused by self-generated contaminant in the switch case. Corrective action consisted of installing Pyrex beads in the air gaps between active and inactive switch elements and revisions in the acceptance test methods.

(2) Qualification History

Qualification tests were performed on the GLV tandem actuators and test-selector valve. The "like Titan II" components were qualified by similarity to the Titan II hardware.

The test selector valve had a modification made in the gasket material during qualification testing, but experienced no design failure.

During qualification tests, the tandem actuator experienced a major problem. The actuator forward body cracked at its mounting flange where it mates with the main actuator assembly housing. The forward body flange was strengthened and the seal between the forward body and the actuator assembly housing was modified to eliminate this problem. The life tests were successfully completed. The actuator position switch experienced vibration failures which were corrected by strengthening the switch radius guide-way. Final testing of the modified position switch was successfully completed during the fourth trial of the qualification test program. The final force limiter configuration was qualified during trial III of the program. This configuration contained a teflon "O" ring cap on the "O" ring, separating cylinder pressure from return pressure and inserting a damping orifice to eliminate the oscillation problem. Non-linearities in the flow limiter were experienced with the actuator under loaded conditions. Small bleed holes were added in the valve to modify regulation characteristics and thus improve the linearity.

Table II. C-1. Component Comparison

Component	Vendor	Gemini vs. Titan II Identical	Different	Remarks
Reservoir-Accumulator	Cadillac Gage	X		Instrumentation systems different - Titan 40 mv - Gemini 5V full scale
Tandem Actuator	MOOG Controls		X	Titan II has no redundant system
Electric Motor Pump	Vickers	X		Gemini has a more stringent contamination control
Engine Driven Pump Stage I	New York Airbrake		X	Gemini uses standard differential compensator plus more stringent contamination control & Titan uses short differential compensator
Disconnect	Aeroquip	X		Gemini has a more stringent contamination control
Filter	Aircraft Porus	X		Gemini has a more stringent contamination control
Test Selector Valve	Sterer		X	Titan II has no redundant system
Sustainer Actuator	MOOG Controls	X		Instrumentation system different - Titan 40 mv, Gemini 5V full scale
Roll Nozzle Actuator	MOOG Controls	X		Instrumentation system different - Titan 40 mv, Gemini 5V full scale
Engine Driven Pump Stage II	New York Airbrake	X		Gemini has a more stringent contamination control

An impulse load test was added to the tandem actuator qualification test program subsequent to the actuator problem associated with the GT-2 launch attempt. Test criteria were set up using Titan II engine start transient loads. The modified actuator successfully completed the impulse tests and was flown successfully on GLV-2, and subsequent GLV flights.

(3) Production History

Production problems were encountered on the tandem actuator, sustainer and roll actuators, reservoir-accumulator, Stage I engine driven pump, the electric motor-pump, the disconnect and system test selector valve.

(a) Stage I Tandem Actuators. The initial launch attempt of GLV-2 resulted in a tandem actuator failure. The primary servo-valve body separated from the actuator housing when the servo mounting lug fractured during the engine start-up. The GLV-2 launch attempt actuator failure was simulated in the laboratory using an impulse shock test rig. Data was obtained on the resulting behavior of components and on the internally generated pressures with impact loads. A thorough investigation of T-II and GLV flight data revealed that the actuator loads seen during these engine starts were greater than those measured in static firings at Aerojet early in the T-II program. This new data was used to establish design requirements. Analysis of all facts revealed some marginal design conditions, and with the predicted loads from flight data, confirmed that such a failure could be experienced, particularly in the case where the valve body material was marginal, as was the case in the GT-2 launch attempt. Modifications were made in the servo-actuator design to strengthen the servo valve body and mounting lugs, and to reduce resulting internal pressure loading on the valve. The results, as verified by comparative tests on the impulse shock machine, showed the modified GLV actuator to be more compliant (able to absorb more energy) than either the Titan II or the original GLV tandem actuator.

Tandem actuator quality problems such as lead wire breakage, switch noise, and transducer drift were corrected by improved inspection and process control procedures at the vendor.

Minor actuator overshoot-oscillation problems were experienced on GLV-2. This problem was found to be caused by an overcharged torque motor which created valve overshoot-oscillation response characteristics to step input signals. The problem was corrected by improvements in the valve setup procedures at the vendor. Actuators in the field were subjected to special step input response tests after installation in a launch vehicle while new actuators were given similar tests during their acceptance testing.

Tandem actuator position transducer failures prior to GT-3 launch and during the GT-6A launch, and discrepancies in transducer outputs during GT-11 flight resulted in electronic component changes and special component checks to assure flightworthy hardware. GLV-6 and subsequent vehicles incorporated higher reliability transistors in the position transducer and all electronic pieceparts in the unit were subjected to more stringent acceptance tests. Component checks designed to show transducer degradation in the field were incorporated.

(b) Stage II Sustainer and Roll Actuator. Minor quality problems such as broken lead wires and damaged seals existed with the Stage II actuators. These problems were corrected by improved handling and inspection procedures at the vendor.

(c) Reservoir - Accumulator. Rigid acceptance tests were introduced for acceptance of accumulator-reservoir "O" ring seals. Tests were designed to eliminate seals with internal voids which tended to allow permeation of N_2 from the accumulator into the hydraulic fluid in the reservoir.

Excessive contamination was found to exist in the accumulator-reservoir fluid chambers prior to the launch of GLV-5. This condition was found during a failure analysis which was attempting to isolate the source of fluid in the gas chamber of the accumulator. The fluid was found to be residual hydraulic fluid which was used to proof test the accumulator chamber during acceptance test. Concerning the contamination, it was discovered that the vendor was not adhering to acceptance test requirements in the method used to check the cleanliness of the unit. The vendor's facilities were not designed to provide a high enough fluid flow to flush the accumulator-reservoir chambers free of contaminants. All units in the field in addition to new components were cycled through the Martin-Baltimore laboratory for recleaning to GLV requirements.

During GLV-7 checkout at ETR, a reservoir was overpressurized causing the endcaps to separate from the main body of the reservoir-accumulator. The overpressurization was caused by two violations of procedural steps in bringing up pressure in the Stage I hydraulic system. Further clarification of procedural steps and cautioning of personnel to closely follow established procedures was undertaken in an effort to reduce human errors of this type.

(d) Engine-Driven Pump. The engine-driven pump compensators on Stage I were changed from a "short-differential" to a "standard-differential" compensator to eliminate the excessive secondary pressure overshoot experienced on GT-1 at engine start. The standard-differential compensator was also incorporated to eliminate the long period of pressure oscillations subsequent to engine start in both the primary and secondary systems.

GT-2 flight data indicated an abnormal primary pressure dip at engine start. This pressure level approached that required to cause switch-over from the primary to secondary system. When it occurred, the pressure rise characteristic was as expected for a normal pump. Subsequent performance in the flight was satisfactory. The cause of the anomaly appeared to be a momentary pump compensator hangup in the no-flow position. It was felt that the compensator hangup was caused either by burrs, shaft distortion, contamination or a phenomenon termed hydraulic lock. All engine driven pumps were completely disassembled at the manufacturer's plant, were subsequently cleaned with particular attention to the elimination of any contamination and were then carefully reassembled. All pumps for GLV-3 and up were installed after the major portion of the vehicle checkout had been completed. Delayed pump installation allowed a minimum exposure to test conditions and to possible contamination buildup. A gauss meter check was devised to record compensator stem motion per a given pressure drop. This check was conducted after pump installation and during launch vehicle countdown to assure that the compensator was free to move. No subsequent problems were encountered with the pump compensator.

On later launch vehicles many pumps showed evidence of intergranular corrosion on compensator caps and pump bodies. All suspect units were cycled through the vendor for replacement of affected parts.

(e) Disconnect. The disconnect housing weldment fabrication techniques were improved early in the program to eliminate cracking in the welds which was discovered on early launch vehicles. Disconnect mating surface scoring was eliminated by improved component inspection, through the vehicle testing sequence, and the relocation of the AGE fluid filter away from the disconnect end of the ground servicing hose.

(f) Electric Motor-Pump. A cracking problem in the motor case of the electric motor-pump was discovered on Titan II operational vehicles. The problem occurred due to tolerance buildup causing an excessive press fit between the motor housing and internal parts. Component tolerances were revised to eliminate the problem and new pumps purchased. Since the pump was not operated during flight, minute cracks were allowed in nonload bearing areas of the pump case for existing units in the field.

A check valve was installed in the Stage I pump discharge line for GLV-6 and subsequent vehicles to reduce the pump case pressure trapped in the motor-pump circuit after operation in parallel with a second pressure source, either ground power or engine driven pump. The trapped pressure in the common lines resulted in high pump case pressures which distorted seals and made external leakage paths. This source of leakage problems was discovered as a result of a failure analysis investigating the cause of hydraulic fluid seepage at the motor-pump interface. The condition had been observed on pumps installed on GLV's 5, 6, and 7 thus initiating the detailed investigation.

(g) System Test Selector Valve. The system test selector valve showed a tendency to hangup in the secondary position during switchback from secondary to primary systems during pre-countdown testing for GT-8, 9, and 10.

Failure analysis indicated the possibility of misalignment of the stacked elements in the selector valve pilot valve assembly. It was felt that the misalignment allowed cocking of the pilot valve solenoid creating high friction forces during solenoid operation. To correct the situation shims were added behind the forward valve seat to eliminate adverse tolerance conditions. This modification was made for GLV-9.

In an effort to further understand the selector valve problem, tests were run in the Martin-Baltimore ASFTS area to look at valve internal pressure conditions during operation. It was found that the driving pressure fell off rapidly with valve operation due to a drop in the electric motor-pump discharge pressure. The drop in pressure was caused by a pump starvation condition due to return flow path restrictions in the selector valve. As corrective action, the main valve spool return land was cut down to open up the return flow path to the pump. The change was incorporated for GLV's 10, 11, and 12.

d. Conclusions

Problem trends showed a slight negative slope throughout the program with early design problems being replaced later in the program with quality and reliability problems. The ASFTS test program, qualification test program, T-II flight test program and Gemini flight test program demonstrated the high performance level and the basic reliability of the hydraulic system.

6. FLIGHT CONTROL SYSTEM (Primary and Secondary)

a. System Description and Design Concept

The flight control system was made up of two independent subsystems, the primary and the secondary system. The purpose of each of these systems was to provide: (1) open loop trajectory and vehicle stability control during Stage I flight, (2) ground control (radio guidance) in Stage II flight in the case of the primary subsystem, and (3) inertial guidance control in the event of switchover to the secondary subsystem during Stage II flight. Both of these subsystems were conventional attitude and rate feedback control systems utilizing gain switching. The control moment was produced by engine thrust vector control which was provided by hydraulic actuators attached to the thrust chamber assembly gimbals. The secondary subsystem was used as a backup in case of serious primary system malfunctions.

(1) Primary Flight Control Subsystem

This subsystem was composed of a three axis reference system (TARS), a Stage I rate gyro package, an autopilot control package, and an adapter package. Pitch, roll, and yaw attitude stabilization signals were originated by the TARS and transmitted by the adapter package to the autopilot control package. Steering control was accomplished by precessing the TARS attitude gyros. During Stage I flight, the trajectory was controlled by programmed attitude changes in roll and pitch developed in the TARS package. During Stage II flight, the TARS pitch and yaw attitude gyros were torqued according to input commands from the radio guidance system.

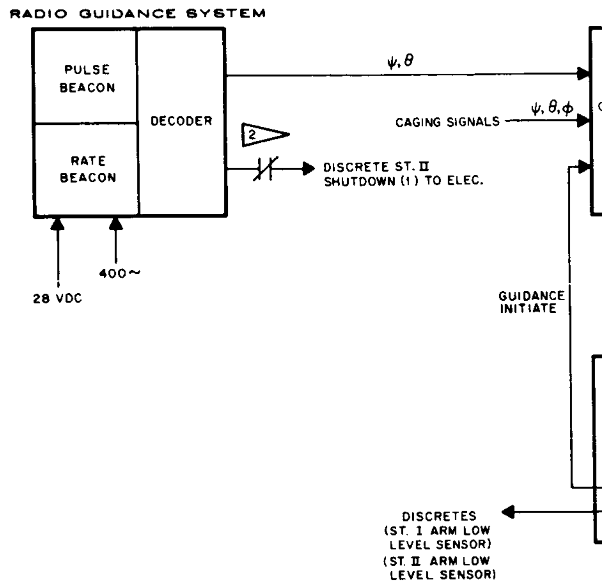
(2) Secondary Flight Control Subsystem

This subsystem differed from the primary flight control subsystem in that an inertial guidance system (IGS), located in the spacecraft, was used for attitude reference providing open-loop programming of Stage I flight and closed-loop guidance commands in Stage II flight. The secondary system was redundant to the primary system through the tandem actuators in Stage I flight and redundant to the primary system except for actuators in Stage II flight. The IGS sent attitude and guidance commands through the control system at all times. When a switchover was commanded, hydraulic pressure was applied to the secondary actuators for Stage I flight, while the secondary control signals were simply switched to the actuators for Stage II. Simultaneously, the switchover signal was sent to the IGS. Upon receipt of a switchover signal, the IGS reduced its output close to zero and then restored the attitude error signal to the system according to an exponential law. For GT-3 and subsequent flights, the capability was provided for switching back to the primary system during the Stage II flight so that the radio guidance could be used if the failure requiring switchover was in the Stage I hydraulic system. A block diagram for the flight control system is presented in Figure II. C-18 and for the switchover mechanization in Figure II. C-19.

b. Component Configuration

(1) Three Axis Reference System (TARS)

The three axis reference system contained three strapped-down integrating rate gyros (pitch, yaw, and roll) and a digital solid state timer for programmed gyro torquing in pitch and roll during Stage I flight. The programmer also provided discrete signals for other vehicle functions, e.g., arm Stage I shutdown sensor, Stage I gain change, arm Stage II shutdown, and guidance initiate.



NOTES:

1. [1] SWITCHOVER SYSTEM CONTAINS REDUNDANT POWER AMPLIFIERS LOCATED IN THE MDP AND FCS SWITCHOVER RELAYS LOCATED IN THE ADAPTER PACKAGE
2. [2] DENOTES CONTACTS IN FCS SWITCHOVER RELAY NO. 2
3. —·———— DENOTES PRIMARY SYSTEM
4. ——— — — ——— DENOTES SECONDARY SYSTEM
5. IN STAGE I TANDEM ACTUATORS, "S" DENOTES SOLENOID; "PS" DENOTES PRESSURE SWITCH; V1, VALVE NO. 1. V2 DENOTES VALVE NO. 2, AND "P" DENOTES POSITION SWITCH (HARDOVER SENSOR)
6. $1_1, 2_1, \ldots\ldots 6_2, 7_2$, DENOTE ACTUATORS

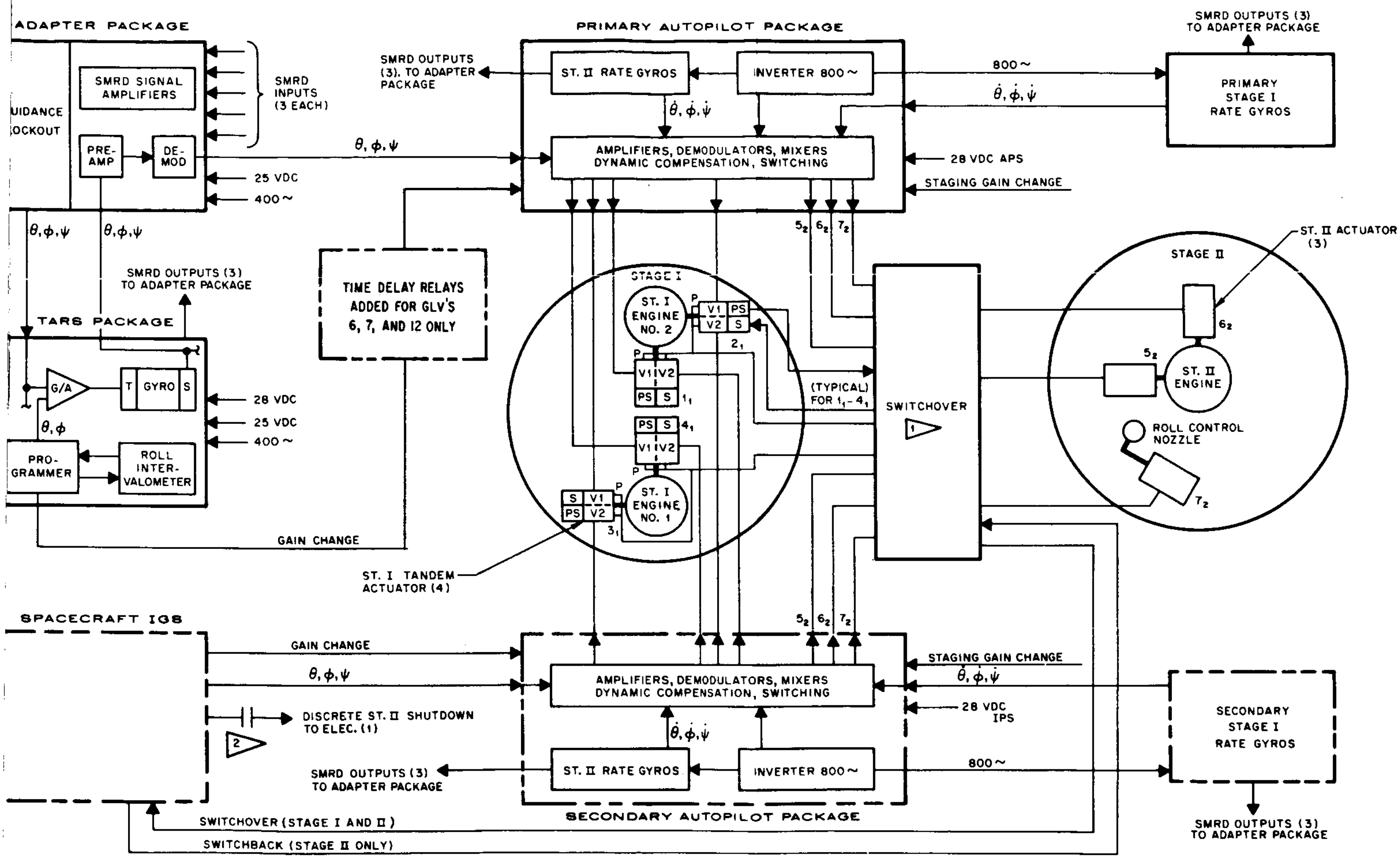
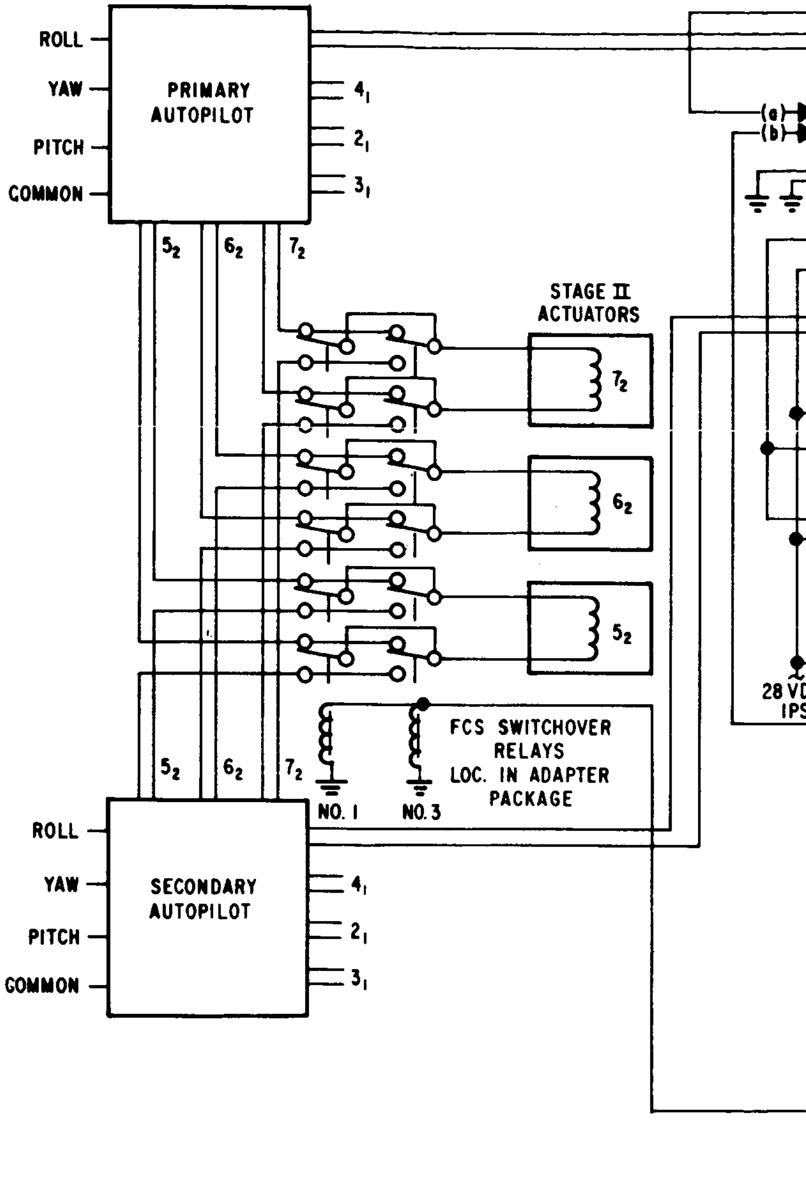


Figure II. C-18. Flight Control System


ROLL
YAW
PITCH
COMMON
PRIMARY AUTOPILOT
4_1
2_1
3_1
5_2
6_2
7_2
STAGE II ACTUATORS
7_2
6_2
5_2
FCS SWITCHOVER RELAYS LOC. IN ADAPTER PACKAGE
NO. 1
NO. 3
ROLL
YAW
PITCH
COMMON
SECONDARY AUTOPILOT
4_1
2_1
3_1
(a)
(b)
28 VD
IPS

II. C-52

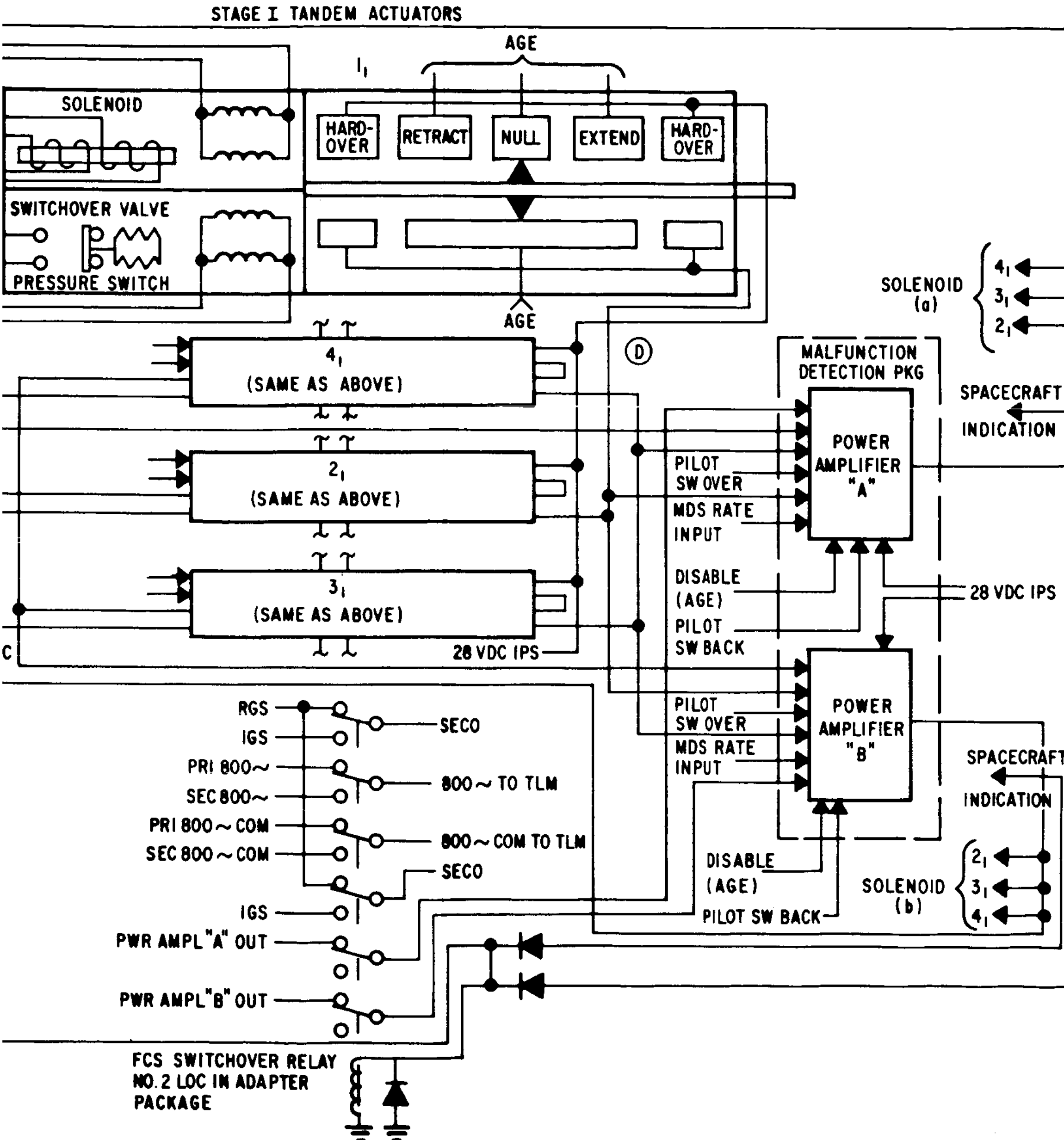


Figure II. C-19. Switchover Mechanism

The TARS received signals from the MOD III radio guidance decoder during Stage II flight. The basic TARS package was used on the Titan I Program. The major change that was incorporated for the Gemini Program was the reactivation of the roll programmer.

(2) Stage I and Stage II Rate Gyro Packages

These packages contained pitch, yaw, and roll rate gyros for sensing vehicle angular rates. Both of these packages were previously used on Titan II. The Stage II rate gyros were contained within the autopilot package and the Stage I rate gyros were in a separate package.

(3) Adapter Package

The adapter package contained signal conditioning electronics for the TARS attitude signals. The attitude signals were amplified and demodulated so as to be compatible with the input requirements of the autopilot control package. The switching relays for the switchover circuitry were located in this package. Included in the adapter were the gyro spin motor rotation detectors for ground testing. This adapter assembly was unique to the Gemini Program.

(4) Autopilot Control Package

This package contained electronic (transistor and magnetic amplifier) circuitry that amplified and provided dynamic modification of the signals controlling electro-hydraulic actuator position. The controlling signals to the actuators were inititated by the inputs of the Stage I and Stage II rate gyro signals and the TARS attitude signals. During Stage I flight, inputs from both rate gyros were used. The package also contained an 800 cycle static inverter power supply and the Stage II rate gyro package.

The autopilot assembly was previously used on the Titan II Program. Rate and attitude signal filters and loop gains in the autopilot were changed, where required, to provide adequate margins for GLV flight stability.

c. Component History

(1) Development History

Basic Titan I and Titan II flight control system components were adapted to the Gemini flight control system with minimum changes. Complete systems development and compatibility testing was accomplished in the Airborne Systems Functional Test Stand (ASFTS) at Martin-Baltimore. This special facility included a complete set of vehicle systems hardware; all flight controls components, a complete hydraulic system, dummy engines, analog computer-simulated vehicle dynamics, a spacecraft simulator, and standard vehicle test AGE. The ASFTS testing evaluated component compatibility and closed-loop system performance for a wide range of flight conditions. It also proved to be extremely valuable for problem resolutions and design change evaluations. The ASFTS proved to be especially useful for the resolution of hardware problems throughout the program. Several AGE problems at ETR were resolved through the use of this facility. TARS temperature malfunction indications at Complex 19 were resolved during many of the prelaunch sequences by sending the AGE chasses to Baltimore for functional analysis in the ASFTS setup. ASFTS is described further in Section II. C-16.

The secondary flight control system problems on GLV-12, the Stage I and Stage II Spin Motor Rotation Detection (SMRD) "no-go" indications along with the 26 VAC malfunction indication, were successfully isolated in ASFTS. The rapid isolation of the problem to the static inverter in the autopilot cleared the flight hardware in the launch vehicle for what proved to be a successful final flight.

The Three Axis Reference System (TARS), which incorporated a solid state electronic timer, had been modified since its initial development for Titan I. A significant change was the reactivation of the roll programmer so the TARS would be compatible with the MOD III guidance system, which did not transmit roll program information from the ground.[9]

A bigger temperature compensation bellows was incorporated in the HIG-4 gyros, effective GLV-1 and up, to prevent a negative pressure from developing in a cold gyro, causing bubble formation in the flotation fluid. The overall radio guidance gain was corrected by changing some TARS gain resistors, effective GLV-1 and up.

The Stage I and Stage II Rate Packages were manufactured by the Giannini Controls Corporation. The Stage I rate package was a subassembly for the Stage II autopilot package. Because these units were used "as-is" from Titan II, no special Gemini development effort was applied.

The Autopilot Adapter Package was designed by Martin-Marietta Baltimore specifically for use in the GLV primary flight control system. Its function was to change the TARS output signals from modulated 400 cps to d. c. signals suitable for inputs to the Titan II Autopilot Package. There were no significant changes incorporated in this unit as a result of the development period.

(2) Qualification History

In general, the qualification requirements for Gemini were the same as for Titan II. However, the temperature requirements were more stringent for the Gemini Program. All of the components, except the adapter, were originally qualified for either Titan I or Titan II. In the case of the adapter, development and qualification were performed exclusively for Gemini.

TARS - Since this unit was basically the same as that used on Titan II, it was tested only for the Gemini temperature requirements and passed.

Autopilot Adapter Package - Vibration problems were encountered during initial testing. A vibration damping rubber pad was installed between the modules and case to eliminate noise in the SMRD circuits. The case cover was also stiffened to prevent bulging at altitude. Design changes were made and the unit met the test requirements. This unit failed to meet the specification on electrical interference requirements, but evaluation showed that a deviation was acceptable.

Rate Gyros - The gyro packages were qualified for Titan II usage. Due to Gemini unique environment conditions, the unit was retested for a wider temperature band and passed with no problems.

(3) Production History

A review of the production history of each of the flight control components for significant problems, failures, and corrective action is presented below:

TARS - Throughout the production and testing history, only one design deficiency was found, and it was not an airborne problem. At ETR, on GLV-2, it was found that under certain conditions, the timer roll intervalometer in the TARS (S/N F15) could malfunction. This was found to be a result of a Schmitt Trigger generating extra pulses, and could be traced to a marginal transistor selection within the TARS. The corrective action consisted of a special test to screen all units prior to use.

Due to the predicted performance characteristics of GLV's 6, 7 and 12, stability considerations required that the Stage I gain change time be revised to LO + 110.0 seconds from LO + 104.96 seconds. Parallel time delay relays were added in series with the TARS gain change discrete to accomplish this modification.

A series of failures (4) involving the Gyro Spin Motor Rotation Detector (SMRD) coils occurred in mid 1965. The defect was screenable and had no inflight effect since they were only used for ground checkout. The problem was found to involve manufacturing methods and test techniques. Corrective action was implemented by Minneapolis Honeywell.

Diodes manufactured by the gold paste process were found to be defective in that gold flakes which formed during manufacturing had caused diode malfunctions. It was decided to replace the suspect diodes in the affected TARS assemblies. This change was effective for GLV-3 and up.

Due to higher than expected failure rates and life limitations on existing units, it was decided to build five additional TARS units. Additional process and piece-part controls were included in the new build based on the failure history at that time. Resistors, relays, capacitors and transistors were x-rayed for defects where schedule and cost implications were not limiting factors. The use of existing subassemblies was reviewed based on their particular data histories.

A slight shift in resistance in a TCA (Temperature Control Amplifier) precision resistor resulted in an out-of-spec condition for gyro gain in one of the TARS packages at M/B. The extensive investigation that resulted from this failure revealed that the failed resistor did not include in its construction a nickel slug used in the welding process for the attachment of leads. A large sample of this type of resistor was dissected and none had the nickel slug missing. It was decided that no changeout would be attempted in existing units because of the uniqueness of the failure and the possible degradation due to changeout. The new build packages did include an x-ray examination of these particular piece parts for this type of defect.

The failure history of a resistor used in the TARS guidance amplifiers was found to be unacceptable. The problem involved an incomplete cure process for the protective coating of the resistor and the presence of aluminum traces on the ceramic resistor body. Extensive investigation determined that the defect was isolated to specific sizes and particular build times. A special directive was issued whereby all subassemblies would be physically inspected for the presence of this resistor. All other possible applications of this resistor on Gemini were reviewed and corrective action taken where necessary.

There was one failure at ETR concerning a transformer in the TARS heater circuitry. The unit was being used as a test tool to check out the test set when it was found that the airborne heaters were inoperative. Failure analysis revealed that the primary winding was burned out and that the failure was probably caused by shorted windings. The cause was not determined, but thorough investigation indicated that the problem was most probably caused by a necked-down or nicked wire in the primary winding, or the misapplication of an over-voltage. Review of the failure history did not reveal any similar failures or problems with this component. It was decided that there was no effective corrective action that could be taken for the flight of GLV-12 and that the probability of another failure was remote.

Autopilot - Most of the problems involved with this package concerned the internal rate gyro package. These problems are covered in the rate gyro section.

As the result of production test failures and subsequent failure analysis, it was determined that diodes containing loose gold flakes were installed in the autopilot assemblies. These were the same kind of diodes that were found in the TARS assembly. Since a malfunction of these diodes could cause an unsuccessful countdown or loss of vehicle stability control during flight, it was decided to replace these diodes. Effectivity was GLV-3 and up.

Adapter - A failure occurred at ETR in the adapter attitude signal conditioner pitch preamplifier module on GLV-1. Failure analysis and investigation of the module and components used indicated an inferior transistor, USA 2N328A (manufactured with an alloy junction and acid etch process) which was susceptible to thermal shock and vibration after aging. Corrective action dictated removal of all electronic modules manufactured with this type transistor and replacement with units using transistors selected under strict controlled requirements. Part of the new requirements included using JAN specification parts. Although the new transistors eliminated the original deficiency, the 100% x-ray inspection imposed did detect metallic contaminants in the JAN 2N328A units. No problems had been experienced with the JAN units, and it was determined that the silicon oxide passivation and the transistor physical construction were such that it was highly unlikely that the contaminant could cause a transistor failure.

The adapter switchover relay module was redesigned because of a failure during manufacturing module tests, in which a relay plunger bound up. New relays were procured to specifications which controlled cleanliness, required component logs, and instituted life cycling of samples. The module with the new relays was subjected to qualification level vibration tests. During the vibration testing, it was found that the levels were being amplified by the mounting bracket and causing contact chatter. This was solved by the addition of vibration isolators.[10]

Rate Gyro - The following significant problems were encountered at the vendor's plant and in the field, and resulted in several major production changes. Due to a gimbal hang-up failure which occurred on GLV-1 at VTF and to the failure analysis which followed, extensive changes were made to upgrade process controls and acceptance criteria for manufacturing clean rate gyros. Contamination control was incorporated.

Several failures due to open torquer coils occurred at Martin/Baltimore. The torquer function was used for ground checkout only and not during flight. Process changes were implemented early in the program to better control the manufacturing and handling of the torquer coils.

Early in the program one of the most significant failure modes of the gyros in the field was the failure of motors to run up to synchronous speed within the specified time. This requirement was a carry-over from Titan I and Titan II performance restrictions and provided a measure of motor bearing quality on Gemini. Special tests were incorporated (effective GLV-2) to find the marginal units. Failure analysis of failed gyros revealed serious bearing degradation and lack of lubricant. It was determined that bearing preload level and control were contributing to these failures and new process control methods were incorporated for gyro subsystems effective GLV-6 and up. No further failures of this type were encountered for the remainder of the program.

It should be noted that the reduced preload did appear to cause an increase in gyro output noise. ASFTS evaluation determined that system performance was not adversely affected by this characteristic noise.

It was originally expounded that once a rate gyro reached synchronous speed it would not run subsynchronously. There was a unique case in which a gyro did run subsynchronously intermittently. Subsequent failure analysis was inconclusive. On GLV-12 at the launch complex, AGE indicated that a secondary rate gyro ran subsynchronously intermittently. The gyro was removed and the failure did not repeat during extensive testing at Baltimore. Therefore, there was only one substantiated case of a gyro which did run subsynchronously.

Another significant failure mode of the gyros involved insulation resistance. Several failures occurred with a high impedance path appearing from an insulated standoff to the gyro stationary top bearing plate (both internal to the gyro). Extensive evaluation, including special ASFTS tests, determined that the failure mode would not affect the gyro or flight control system performance. Production was already terminated at the time the problem was isolated, so no corrective action was implemented concerning production processes. Special tests were incorporated in the field to screen out the defective units. These tests were also run as close to launch time as possible to provide maximum confidence in the flight hardware.

As in the case of the TARS, there were several instances of recycling of gyro units because of test equipment malfunctions. Differences in test procedures and techniques also contributed to rejection and subsequent recycling of good flight hardware.

There were several gyro null failures in the field. The inherent temperature sensitivity characteristics of the gyro resulted in null shifts as the gyros were temperature-cycled. In most of the cases, the gyros were stabilized by recycling through the vendor's facility and readjusting the null.

(4) Flight Test History

The flight control system performed satisfactorily in GLV's 1 through 12. The secondary system was never used in flight; however, telemetry data revealed no significant abnormalities. Some items were found which are considered important and are mentioned below for historical purposes. They should not be considered as significant problems.

1) The roll liftoff transient was large and resulted from thrust and rigging misalignments. The Stage I roll actuator rigging was biased to compensate for this on GLV-9 and up. Subsequent liftoff transients were significantly reduced.
2) Slosh modes were excited in Stage I and Stage II as expected and were of small amplitude.
3) Staging transients were large and were caused byengine thrust decay characteristics in conjunction with staging gain changes.
4) Post-SECO transients were evident on every flight. The cause of this phenomenon (Green Man) was never determined; however, its effect on performance was not significant.
5) A small bias was noted in the TARS timer performance subsequent to BECO. Exact cause was not determined; however, it had no significant effect on performance.

d. Conclusions

The flight controls component development was based on flight-proven hardware for three of its four components. The design changes required for the system were minimal. The fact that there were no in-flight failures throughout the program has demonstrated that the controls that were imposed and the corrective procedures that were incorporated were sufficient to ensure satisfactory performance of the flight control system.

7. ELECTRICAL SYSTEM

a. System Description

The Gemini Launch Vehicle electrical system consisted of the power distribution system and electrical sequencing system.

The power distribution system was composed of the accessory power system (APS) and the instrumentation power system (IPS). The accessory power system consisted of the APS Battery, APS Inverter and distribution circuitry, and supplied the power required by the primary guidance and flight control systems, the electrical sequencing system and APS Range Safety System.

Similarly, the instrumentation power system consisted of the IPS battery and supplied the power required by the secondary guidance and flight control systems, the redundant electrical sequencing system, and the IPS range safety system. A block diagram of the electrical power distribution system is shown in Figure II. C-20.

b. System Operation

The electrical sequencing system consisted of relay and motor-driven switch logic to provide discrete signals to the vehicle system. Figure II. C-21 presents a block diagram of the electrical sequencing system and the manner in which it is integrated with the vehicle. As the vehicle lifted off from the launch pad, disconnect plugs removed a ground from program initiate relays 1 and 2. This action energized the three axis reference system (TARS) timer and 145-second time delay relay. After approximately 145 seconds the time delay relay and TARS timer armed the Staging Control Relays 1 and 2 by applying +28 vdc to their coils. Upon depletion of Stage I propellant, Stage I chamber pressure switches sensed the decaying pressure, closed and provided the required ground to operate the staging switches. Operation of either staging switch was sufficient to effect staging, (Stage I/II separation). As shown in Figure II. C-21 operation of either staging switch provided autopilot gain changes, shut down the Stage I engine, started the Stage II engine and armed the Stage II shutdown. Explosive nuts on each side of the bolts joining Stages I and II were activated to effect separation. It was only necessary for either Stage I or Stage II side nuts to operate. The normal mode of Stage II engine shutdown was via the radio guidance system, however, the spacecraft inertial guidance system would command Stage II shutdown when the vehicle was under inertial guidance system control. Therefore, the electrical sequencing system was fully redundant.

c. Configuration

The basic difference between Gemini and other Titan electrical systems was the use of a completely redundant electrical system to support the secondary guidance system on the GLV. This was accomplished by reworking what was solely an instrumentation power system.

The Gemini electrical system was designed and integrated with the other launch vehicle systems in such a way that a single inflight electrical failure would not cause a catastrophic failure or prevent completion of the mission.

The Titan II APS and IPS batteries were silver zinc, nonrechargeable, and remotely activated while in the booster. To achieve higher reliability on Gemini, rechargeable silver zinc batteries were used and were identical in size and capacity. For a Gemini launch mission, the batteries were activated discharged and charged prior to installation in the launch vehicle.

The 750 va 400 cycle static inverter was a new development for Gemini since Titan II does not use a central a-c power system. The inverter supplied a-c power for the TARS package which was not used on Titan II. The 25 vdc power supply also supplied power to the Gemini TARS system and therefore was not on Titan II; however, this power supply was flown successfully on Titan I.

As a result of a test procedure error in the VTF it was apparent that a short or a very low voltage of the APS bus could provide a ground path to the staging initiate circuit. To eliminate this possibility, the APS power feed to the TCPS was looped through an umbilical connector so that this path would be eliminated in flight. This change was effective on GLV-2. (Reference ECP 372.)

Special fire protection was provided in the Gemini Titan II Stage I engine area wiring by wrapping the wire bundles with insulation and aluminum glass tapes. So protected, the wire could withstand a $3640^{\circ}F$ flame for 104 seconds. This change was effective on GLV-3. (Reference ECP 373.)

To insure positive cutoff of the Stage II engine at optimum time to place the spacecraft in correct orbit, and enable safe return of the astronauts, a redundant shutdown system was incorporated in the GLV for all manned flights. The redundant engine shutdown system used a squib operated shutoff valve in the gas generator oxidizer feed line of the Stage II engine. This valve was initiated from the same signals that operate the shutdown solenoid in the fuel line of the Stage II engine. (Reference ECP 382.)

The Thrust Chamber Pressure Switches (TCPS) Monitoring Circuit was also revised prior to the first manned flight. The change made was to use an existing spare set of prevalve switch contacts to lock out the operation of AGE TCPS monitor relays in the event of a TARS malfunction. This was incorporated to eliminate the possibility of a failure in the TARS package applying +28 vdc to the staging control relays. This single failure could cause staging on the pad during the time interval of T-34 seconds and T-0. (Reference ECP 470.)

Figure II. C-22 represents the configuration of the electrical sequencing system effective on vehicles 9 through 12. The essential difference between this system and the previous electrical sequencing system was the deletion of the Stage I engines thrust chamber pressure switches (TCPS) and incorporation of their function into the malfunction detection thrust chamber pressure switches (MDTCPS).

This change led to the revision of the staging control circuitry in order to protect the MDS system. Electrical wiring changes were made so that one side of the staging control relay coils 1 and 2 were permanently connected to ground and the energizing voltage to these relays was now supplied through MDTCPS contacts. The function of the Program Initiate Relays 1 and 2, the (TARS) timer and the 145-second time delay relay were unchanged. Thus the sequence remained the same; upon depletion of Stage I propellant and loss of thrust chamber pressure, closure of either MDTCPS shutdown sensor would supply APS/IPS voltage to energize the staging control relays. The remainder of the electrical sequencing system was unchanged. This modification overcame the deficiencies in the system which were corrected by the ECP's 372 and 470 and, in addition, increased reliability in that the redundant switch feature of the MDTCPS system added redundant APS/IPS power for the Stage I AGE engine function. (Reference ECP 526.)

d. Major Problems

(1) Motor-Driven Switches

During Combined Systems Acceptance Test (CSAT) for GLV-2, out-of-tolerance conditions were observed for the APS and IPS 8-pole staging switches. A failure analysis performed on these components revealed mechanical defects in one and electrical defects in the other. These defects, however, did not prevent the switches from functioning, but resulted in erratic out-of-tolerance conditions.

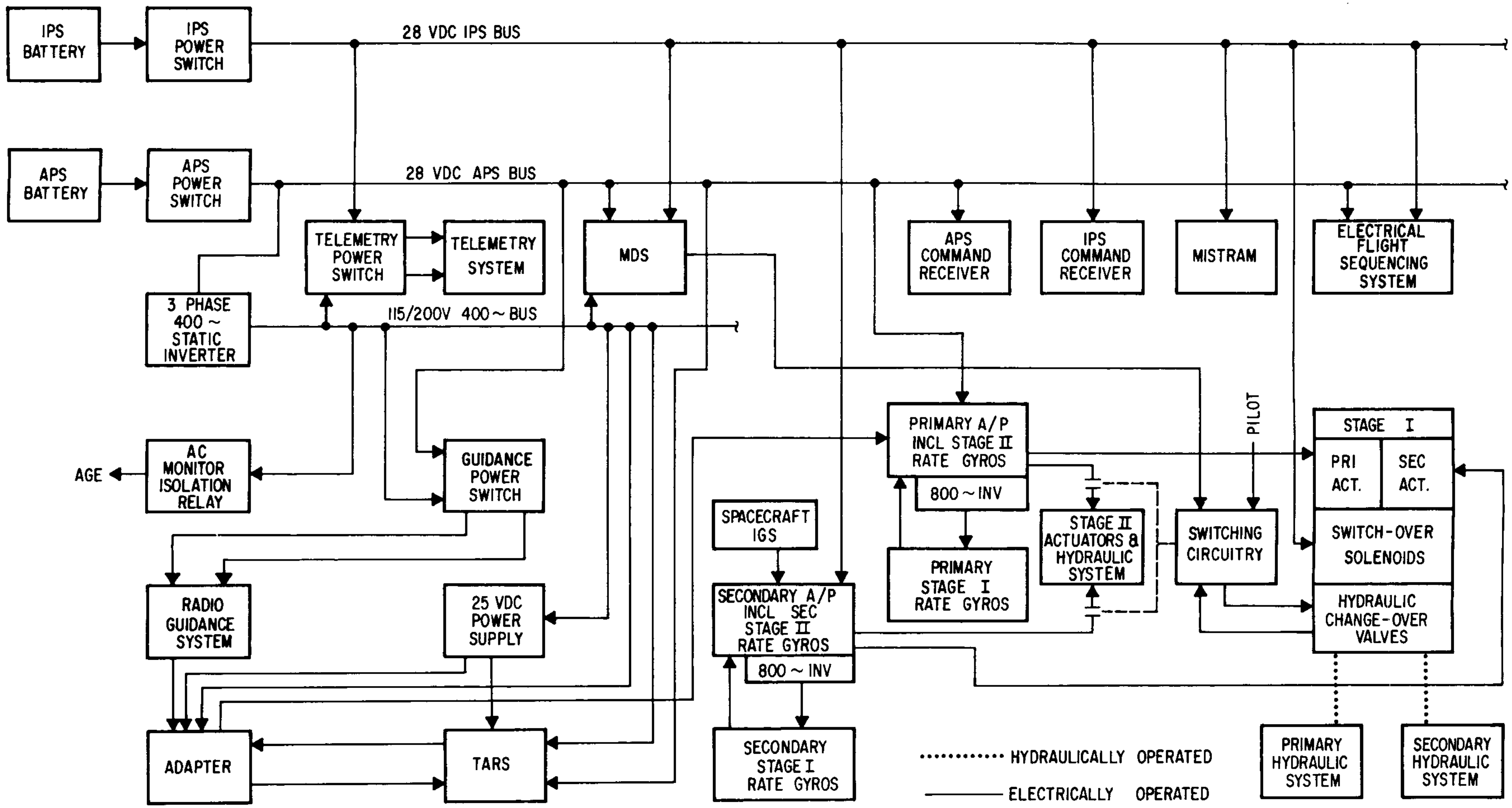


Figure II. C-20. Electrical Power System

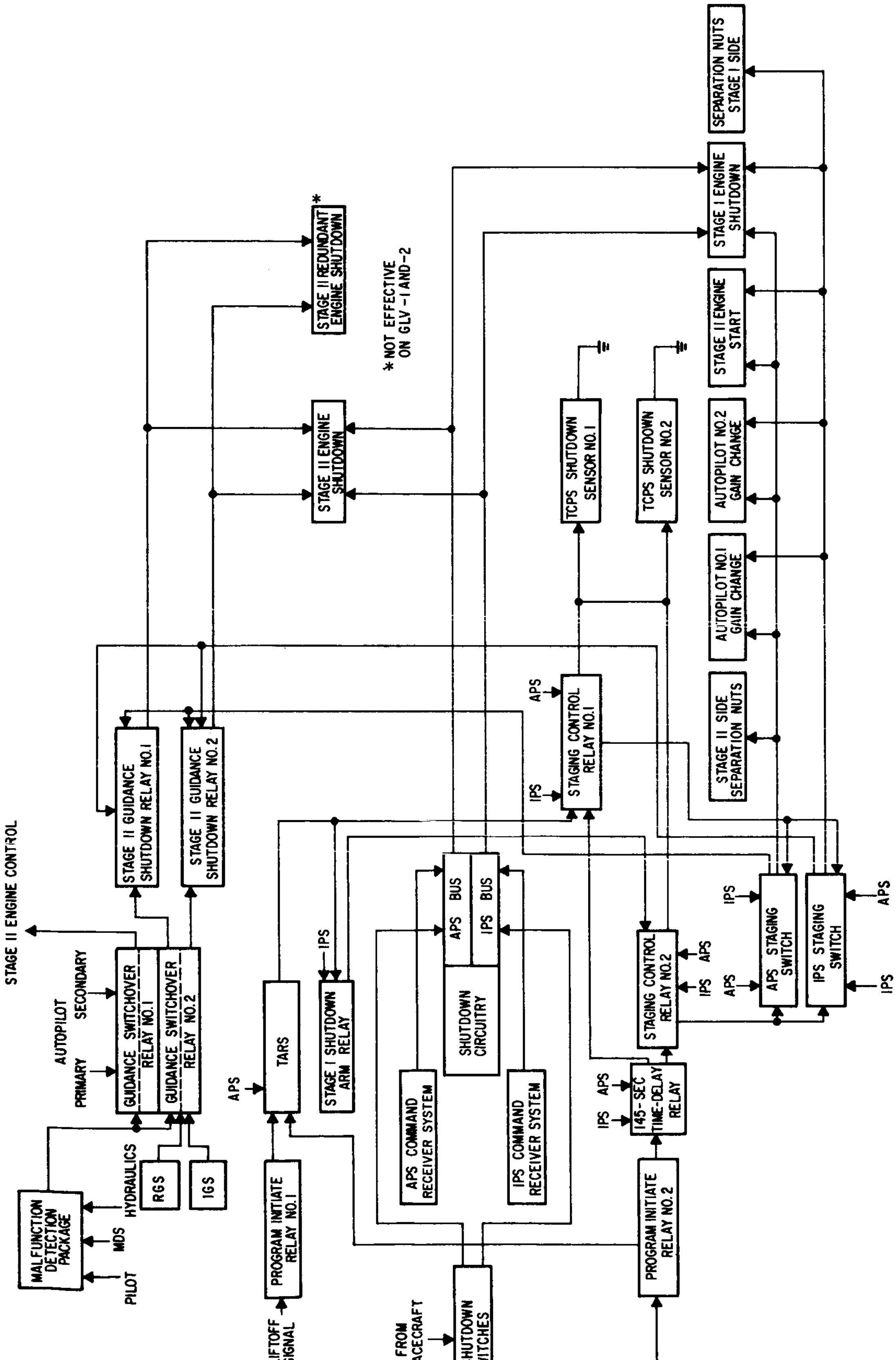


Figure II. C-21. Electrical Sequencing System - Effective GLV-3 through GLV-8

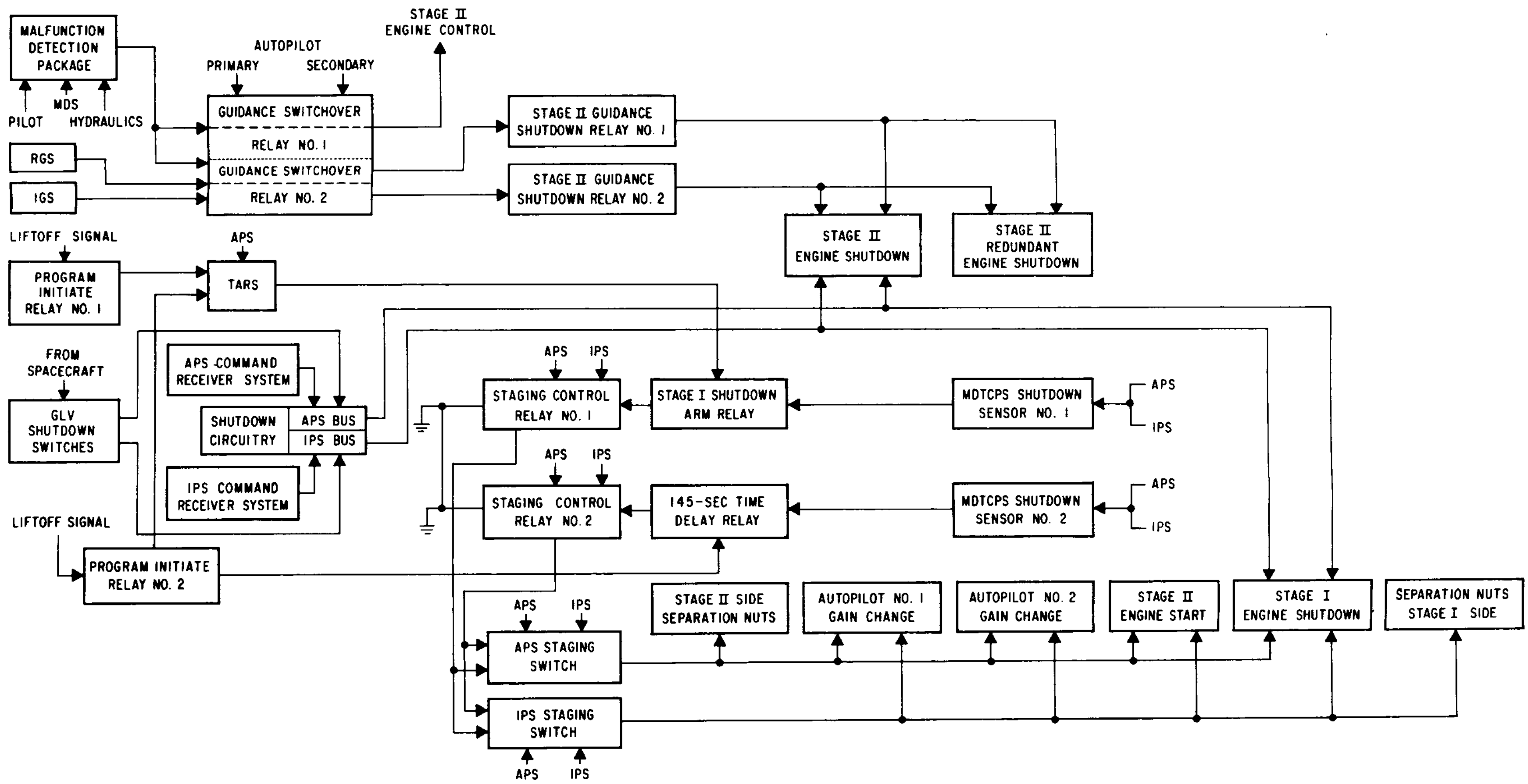


Figure II. C-22. Electrical Sequencing System - Effective GLV-9 through GLV-12

As a result, a special confidence test was performed on each switch in receiving inspection. During checkout test on GLV-5 at the Cape a 6-pole motor driven switch used as a telemetry power switch (TPS) failed to give a normal opening. Further tests indicated erratic motor current and long operating times. Failure analysis performed by the vendor revealed trouble caused by improper cleaning of wax off the commutator following operation for potting of the armature. All suspected units were identified with a manufacturing lot of 17 units. Each unit was located or accounted for, subjected to special retest, returned to stock or scrapped. Manufacturing procedures were changed to eliminate using wax to protect the commutator during potting of the armature.

(2) Time Delay Relays

One production problem arose from the process of solder-sealing the case. Sufficient manufacturing checks were added to ascertain that no damage occurred from the sealing process.

Other problems encountered include potentiometer adjustment difficulties due to galled lead screw threads (a 25-cycle screening test was instituted), and relay chatter (relay position was optimized to reduce shock). A vibration screening was also set up at the relay vendor's facility to catch inferior units.

Subsequent to resolving these problems, a time delay relay timed out erratically in a M/B receiving inspection test. At first, the problem was attributed to solder flux contamination and weld spatter contamination in the switching relay. New time delay relays were built with the new improved and more closely controlled relays. These relays timed out erratically in vibration and failed the qualification test. Further failure analysis revealed the vendor of the tantalum (timing) capacitor had recently made a process change. The problem was resolved by having a special lot of capacitors made to the old manufacturing process with close quality control inspection. The newly built timers with new lot capacitors, and improved switching relays successfully completed requalification tests. These re-identified time delay relays were used on GLV-4 and subsequent vehicles with no further problems.

(3) Microfuses

Numerous microfuses evidenced fractured glass caps after prolonged storage periods in VTF and ETR. The fractures were the result of different coefficients of expansion between the glass cap and epoxy used. The microfuse was redesigned with a metallic cap and the post redesigned to accommodate the new microfuse. (Reference ECP 419)

(4) Disconnect Plugs

During the launch attempt of GLV 6A, a sequencer shutdown was sent at approximately 1.5 seconds after Stage I engine start. Investigation revealed that umbilical 3D1M inadvertently released prior to liftoff. Visual inspection of both the airborne and ground half showed nothing abnormal. Pull tests were performed, using both old and new ground half disconnects. All pull tests were within specification. Vibration tests were also performed and were within specification. It was concluded that most probable cause was improper installation of the plug. Contributing to the cause was a vehicle fairing covering the receptacle which made it difficult to mate the connectors and visually observe the connection. Corrective action was taken to provide safety wiring to help keep connectors mated until pulled by the lanyard, to provide inspection cutouts in the receptacle support bracket, and add keying and alignment markings on the plug, receptacle and installation area. (Reference ECP's 560 and 560R1)

8. INSTRUMENTATION SYSTEM

a. System Description and Design Concept

The purpose of the instrumentation system was the collection and transmission of data to ground-based receiving stations. Evaluation of the vehicle and subsystems' performance was made from data measuring points properly selected. The first four vehicles were heavily instrumented to provide environmental data for correlation with predicted performance. Vehicles 2 and 3 carried protuberance heating sensors to evaluate maximum temperature and heat transfer rates. High frequency vibration measurements were also acquired on these vehicles to determine environment experienced by the G. E. guidance packages. Additionally, high frequency response measurements, as well as numerous engine performance measurements, verified design criteria, and having done so, were eliminated from the GLV-5 and subsequent vehicles.

The instrumentation system consisted of two frequency modulated r-f links, one of these pulse code modulated, and one employing frequency modulated subcarriers, signal conditioners, transducers, magnetic tape recorder/reproducer and necessary power supplies for transducers.

Figure II. C-23 shows the significant elements in the airborne instrumentation system. All major components were located on a truss in the compartment between the Stage II tanks. The antenna system also was installed in this area. Harnessing, independent of the electrical wiring, was utilized for all instrumentation functions. The PCM/FM telemetry was a time-multiplexed data system and the primary data transmitter. The major unit consisted of a multiplexer/encoder with a capability of 196 analog channels and 48 bi-level channels. The encoder output for each channel was an 8-bit binary code providing a resolution of 0.4 percent. The encoder output modulated an FM transmitter with a minimum power output of 25 watts. The overall accuracy, from transducer through data printout, was within $\pm$ 4 percent.

The FM/FM telemeter link transmitted seven channels of continuous high frequency data during first stage flight. Additionally, the tape recorder collected data during a portion of first stage flight and during the interval of radio frequency blackout at staging. The recorded data was reproduced and transmitted over the FM/FM link after first stage separation.

The FM/FM telemeter, the tape recorder, and the associated wiring and control circuitry were removed on vehicle GLV-5 and subsequent vehicles. These are shown within the dotted portion of Figure II. C-23. Wiring kits were provided and hardware was available to utilize the subcarrier system should it have been necessary for problem resolution. However, such a need was not evident for the remainder of the program.

b. Configuration

The basic difference between GLV and Titan II instrumentation was the change from the 40 mv full scale telemetry system on Titan II to a 5-volt full scale telemetry system on GLV. Higher output voltages from remote transducers improved the overall signal-to-noise ratio and reduced the complexity of the multiplexer/encoder. As a result of the higher level system, it was necessary to develop a new line of transducers such as accelerometers, pressure indicators, temperature bridges, etc. Although the basic encoder circuitry from the Titan II was retained with the exception of the low level amplifiers, a new package was developed by the same vendor and was fully qualified. Also, the transducer power supplies and temperature bridges were all of new design and were qualified to GLV environments.

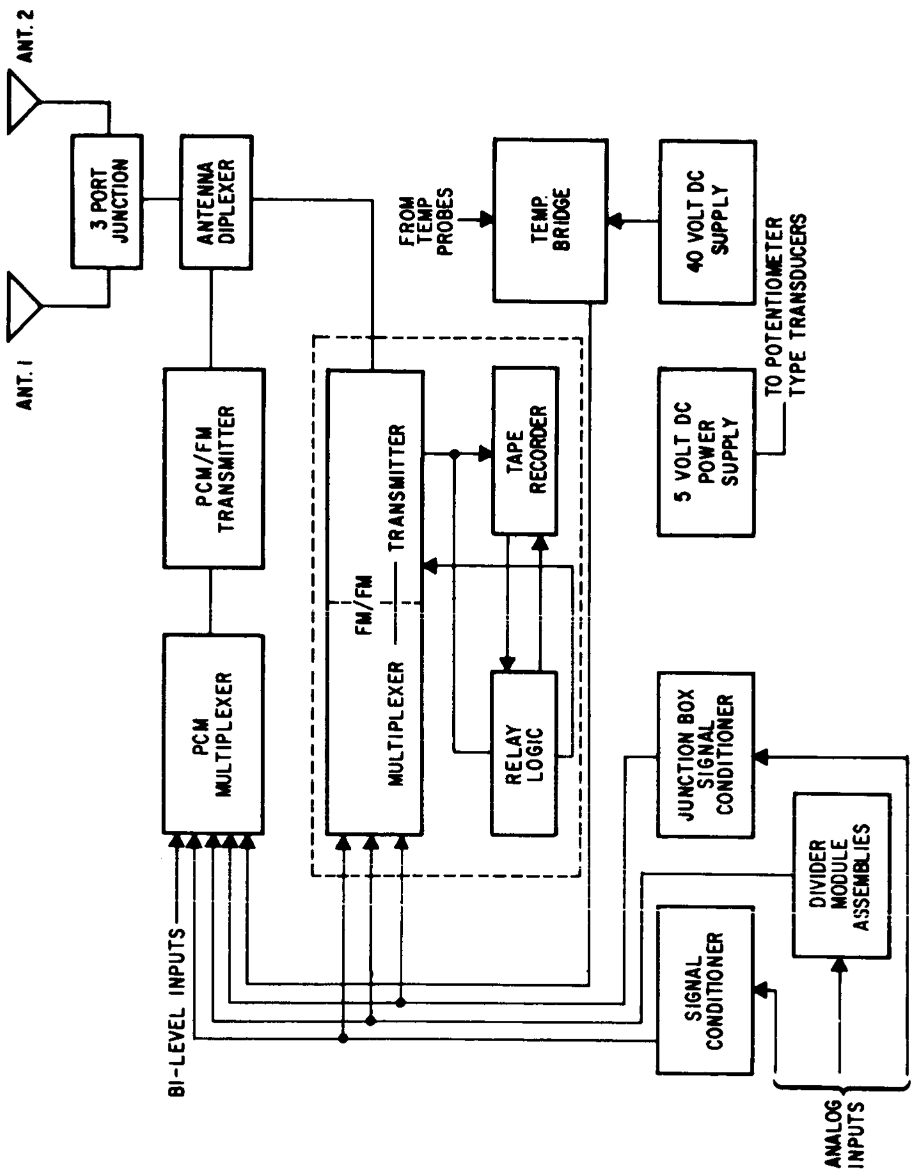


Figure II. C-23. Instrumentation System

The transmitters used for both the PCM and FM systems were nearly identical to those used on Titan II with the exception of an input modification to accept the higher level signals. Minor output circuit components were replaced and a ruggedized tube was substituted in the power amplifier. The transmitter frequencies were changed in accordance with GLV requirements.

The antenna configuration of two elements, displaced 180°, was used on GLV-2 and subsequent units rather than the four element system used on Titan II and GLV-1. This modification resulted in a weight reduction and a simplification of the coupling network.

800 cps modulators were used in the signal conditioner for rate gyro functions which were not used on Titan II. The junction box was modified to add signal conditioners for the TARS, the redundant flight control system and the MDS.

The plug-in patch board, permitting the reassignment of functions to different encoder channels, was removed for three reasons: elimination of unsoldered contacts, assurance that field changes would not affect configuration shown on the measurement schedules, and weight savings. Measurement changes, as required from time to time, were accomplished by means of solder splices in Compartment 2. These were authorized and identified only by ECP action.

Three items were classified "critical components." These were the telemeter (r-f package), signal conditioner, and multiplexer/encoder. Complete histories of these packages from production monitoring tests through the vehicle acceptance test were presented and reviewed prior to vehicle acceptance. Questionable hardware, from a standpoint of excessive repairs, or recurrent type failures would have been identified at this point and replaced on the vehicle.

c. Performance

No system flight failures occurred during the entire flight test program and fewer than six measurements were not completely recovered. Of these, two were intermittent transducers.

However, pressure transducers had a rather high failure rate during vehicle tests with either the application or removal of power. Examination of the failed units proved conclusively that they were somehow subjected to over-voltage transients although in nearly all instances the voltage monitors and alarm circuits detected no such over-voltage condition. It is believed that age and the number of cycles had an effect in reducing the over-voltage capability since each unit was tested at 40 volts during factory tests. New units could withstand the 40 volt level for hours without degradation.

No other item in the telemetry system had a failure pattern and the data recovered from the flights was in excess of 99.5 percent.

9. RADIO GUIDANCE SYSTEM

a. System Description

The General Electric radio guidance system was chosen as the primary guidance system for the Gemini Launch Vehicle. The Mod III-G missileborne guidance set operated in conjunction with the the Mod III-A ground system and the Mod III (A-1) guidance computer (supplied and operated by Burroughs Corp.). A block diagram of the Mod III-G system is shown in Figure II. C-24.

The Mod III-A ground system consisted of the rate measuring subsystem, the position measuring subsystem, and the flight data recording subsystem. The Gemini Mod III-G missileborne set consisted of a pulse beacon, rate beacon, decoder, primary antenna, secondary antenna, and interconnecting microwave components. The design of the Mod III ground system was completed early in 1957 and the first guidance mission from ETR was performed in December 1958. The transistorized version of the missileborne guidance set (Mod III-G) which was designed in 1960 has successfully supported all the Gemini missions.

b. System Operation

The position measuring subsystem tracked the missileborne pulse beacon extracting slant range and angle (azimuth and elevation) information. The rate measuring subsystem, locked in frequency to the missileborne rate beacon, measured doppler frequency shift. The doppler shift at the central rate receiver is a measure of range rate, and the difference in doppler of the two outlying rate receivers from the central receivers is a measure of range rate differences called P-dot and Q-dot. These six measured parameters were used in the computer for the computation of pitch and yaw steering commands which were transmitted on the position measuring subsystem data link to the missile. The commands were received by the pulse beacon, decoded by the decoder, and transmitted to the flight controls system. Discrete commands could be generated in the computer and transmitted over the data link. One discrete command was used on the Gemini Launch Vehicle to initiate SECO. Pertinent performance data was recorded during operation of the guidance system for post-flight analysis by the flight data recording subsystem.

c. System Configuration

The Mod III-G missileborne equipment was the guidance version of the Mod III-F instrumentation system. The basic difference was the addition of the decoder. The Mod III-F beacons supporting the Titan program were an earlier version of the equipment which did not include the vibration isolated baseplate modification, diodes screened 100 percent for internal foreign particles, and "Phase I" reliability improvements. The secondary antenna used on Gemini vehicles was flown successfully on the Titan N-series missiles. Three Mod III-G decoders (without isolated baseplates) were flown successfully on three N-series missiles as part of the Gemini "Piggyback" program.

The interface between the Mod III-G missileborne equipment and the Martin autopilot, shutdown circuits, and instrumentation systems was defined in SSD Exhibit 62-194 Revision B. Vehicle installation, power requirements, signal levels and applicable specifications were defined by the Martin Company and General Electric and approved by SSD/Aerospace.

There were no system configuration changes after the GLV-2 flight. The differences in configuration which occurred between GLV-1 and GLV-2 were as a result of GE ECP-019 and ECP-022. In March 1964, ECP-019 was approved deleting the intermediate mount from the missileborne configuration as a weight saving measure as shown in Figure II. C-25. The ECP required modification of the microwave components only, and those components were qualification tested by July 1964. In July 1964

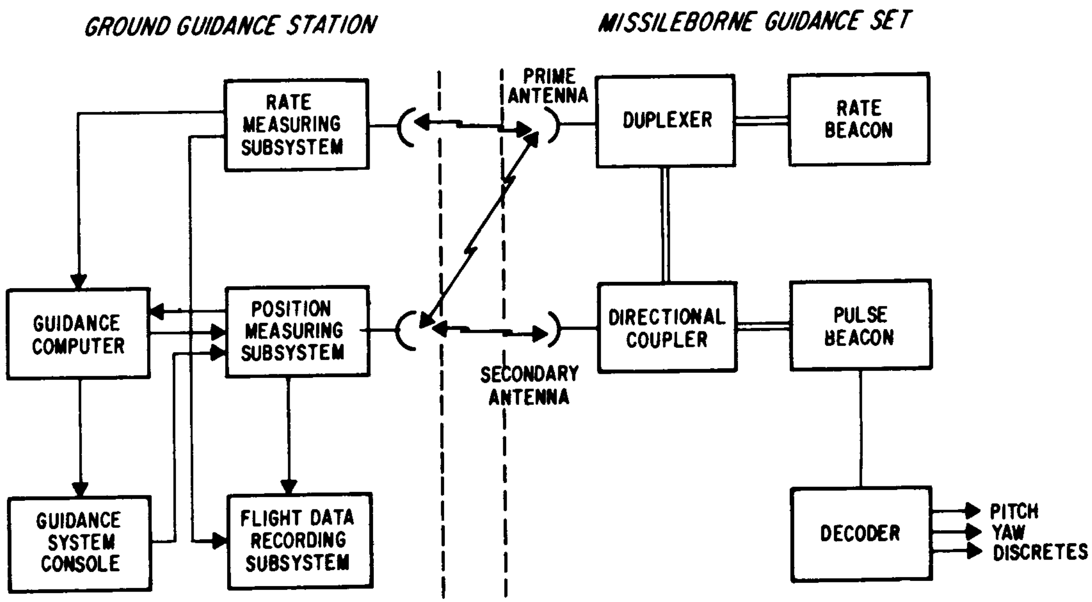


Figure II. C-24. GLV Mod III Radio Guidance System

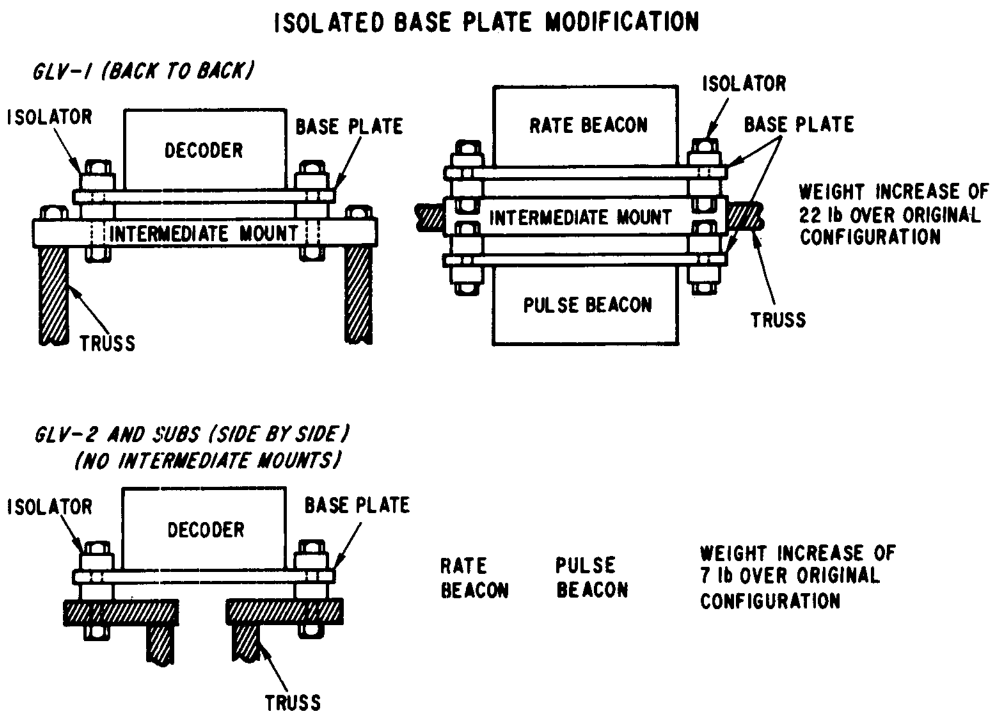


Figure II. C-25. Gemini Mod III G Radio Guidance a reliability improvement consisting of a change in vendors for capacitors and relays in the decoder (ECP-022) resulted in a group number change of the decoder. ECP-019 and ECP-022 were incorporated and flown on GLV-2 through GLV-12.

d. Major Problems

During the vibration phase of qualification tests (1963) of the first equipment produced under the AF04(695)-100 production contract, several problems were encountered such as broken wires and broken component leads. During this period, problems with "gold flaking" diodes were experienced in the factory and field. In July 1963, it was determined that the missileborne equipment, which was originally designed for the Atlas Weapon System vibration environment of 6 g's should be modified to perform in the Titan II vibration environment of 12 g's. ECP-012 was incorporated to correct these problems by adding vibration isolated baseplates, screened diodes, and the "Phase I" reliability improvements as outlined under the Task Force effort (Section II-B. 5. d). This change was flown on all Gemini launches.

In July 1964, a reliability improvement consisting of a change in vendors for capacitors with poor failure history and the replacement of contaminated relays with a more reliable type was implemented. These changes were incorporated in all Gemini decoders under ECP-022 and flown on GLV-2 and all subsequent vehicles.

ECP-036 purged the decoders of silicon-controlled rectifiers which had a history of contamination. An improved silicon controlled rectifier was incorporated and was flown on GLV-7 and subsequent vehicles.

The mica insulator in the Rate Beacon was changed to a kapton insulator as a result of failure history and altitude tests. This was incorporated by ECP-038 for GLV-8 and subsequent vehicles.

e. Special Studies

As a result of special studies, the GLV primary antenna was redesigned from a 6-inch to a 4-inch slotted configuration, and the guidance system noise model was extended to 2 degrees in elevation angle. These studies are explained further in Section II. E-3.

10. RANGE SAFETY SYSTEM

a. System Description

The Gemini Launch Vehicle Range Safety System was designed to meet the range safety requirements established by Air Force Eastern Test Range regulations. These requirements were met by use of a combination electronic and ordnance system consisting of command control, tracking and impact prediction, and destruct subsystems

The range safety system provided the emergency means of confining the airborne launch vehicle within predetermined boundaries of the Eastern Test Range. To accomplish this control, an airborne system was provided to operate in conjunction with the ground-based range safety system. These systems operating together had the capability of (1) determining real-time track and impact prediction data, (2) accepting and acting on the Range Safety Officer's (RSO) command to terminate launch vehicle thrust, and (3) accepting and executing the Range Safety Officer's command to destroy the Launch Vehicle. In case of an abort, the Gemini Program incorporated a 3.5 second time delay between thrust termination and destruct that would have allowed the Astronauts time to escape from the launch vehicle. In addition, a destruct system was provided which would automatically terminate thrust and destruct Stage I should it inadvertently have separated from Stage II during the boost phase. A 5.5 second delay between separation and destruct was provided to allow time for Astronaut escape. The Range Safety System is shown in Figure II. C-26.

The command control receivers were located in the second stage of the GLV and were used in conjunction with the ground based FRW-2 transmitters and associated equipment that made up the command control system. Two command receivers were used to increase reliability, and were both connected to four antennas through a six-port junction. Each antenna was placed 90° apart around the circumference of the vehicle to give a full 360° coverage. This system provided the Range Safety Officer means of issuing flight termination commands. The receivers would decode these commands from the FRW-2 transmitter and produce signals which would enable the appropriate vehicle circuitry to accomplish transmitted commands such as thrust termination, destruct and Stage II shutdown (ASCO).

The MISTRAM System was the primary system for providing real-time tracking and impact prediction data. A MISTRAM transponder and antenna system were installed in Stage II of the launch vehicle and this airborne system functioned with the ground stations at Valkaria and Eleuthera. MISTRAM operated independently of other range systems to acquire the launched vehicle, track its flight, and continuously measure and record its position and velocity. Simultaneously, the system transmitted real-time trajectory data to external computer facilities for Range Safety and Instantaneous Impact Prediction purposes. Secondary tracking data for back-up information to the MISTRAM system was obtained from the C-band beacon in the spacecraft functioning with the ground based Radar tracking network and the MOD III-G radio guidance system.

b. Configuration

Range safety requirements for destruct system arming before launch, coupled with Pilot Safety requirements, resulted in the design of a new destruct initiator that would provide independent action for both the arming of electrical firing circuits and the moving of the destruct explosives in-line.

On GLV-1, a Gemini modified Titan II type destruct initiator was used with no resulting problems. On GLV-2, the new three position Gemini-developed destruct initiator was employed and no problems were experienced (Reference ECP 70).

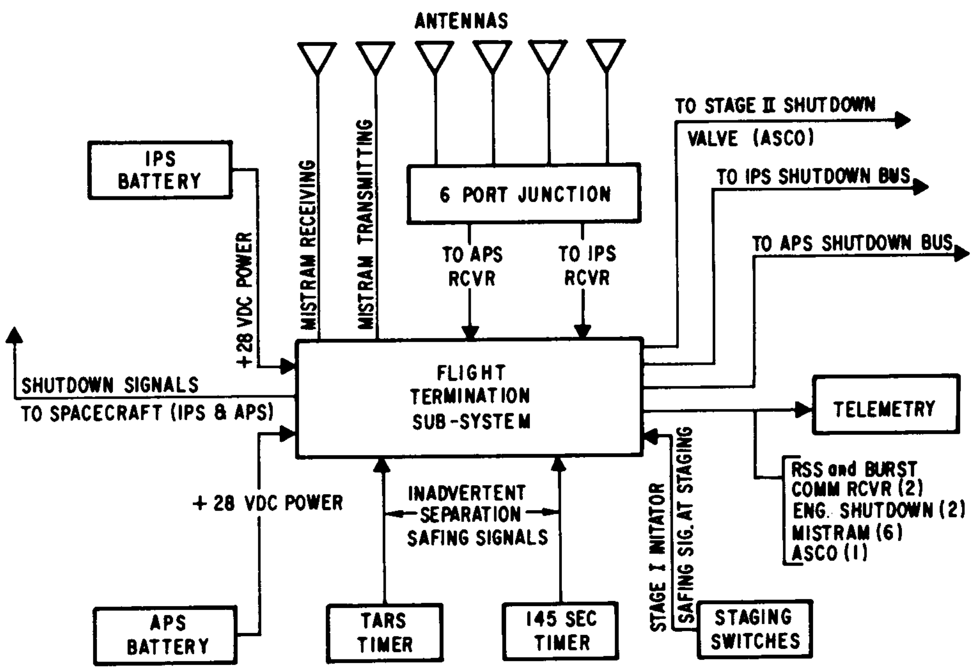


Figure II. C-26. Range Safety System

Except for the destruct initiators used in GLV-2 and subsequent, the ordnance destruct system design concept was identical to the one used on Titan II. Adjacent tank domes in each stage were ruptured by bi-directional charges to obtain propellant intermixing. The destruct initiators, destruct charges and associated primacord harness and boosters, were mounted in or adjacent to the between tank areas of each vehicle stage. The Gemini Ordnance installation was evaluated[11] and tested in accordance with the "General Range Safety Plan AFMTCP 80-2, Volume I, Appendix A."

The Titan II AVCO Mark III Command Destruct Receivers were flown on GLV-1. The command destruct receivers, Model 423A, developed by Advance Communications, Inc., were included in the system primarily for weight reduction, and were flown on GLV-2 and subsequent (Reference ECP 122).

Effective for GLV-2 and subsequent vehicles the T + 40 second time delay lockout of spacecraft initiation of GLV engine shutdown was removed from the launch vehicle circuitry. Incorporation of this change removed a failure mode that could lock out spacecraft engine shutdown through the flight region (Reference ECP 393).

On GLV-2, the Stage I destruct initiator was moved from an external fuel tank conduit to a location inside Compartment IV (between the Stage I tanks). This relocation provided a more sheltered location for the Stage I destruct initiator to increase protection against Stage II engine exhaust environment after stage separation. The change was a result of an investigation of Stage I destruct incidents that occurred on the Titan II Program (Reference ECP 282). For the first manned flight of Gemini, additional steps were taken to prevent possible actuation of the Stage I destruct system by the Stage II engine exhaust. The Stage I destruct initiator, the primacord leads, the primacord conduits and the two destruct charges were wrapped with thermal protecting silicone and aluminum glass tapes (Reference ECP 287).

Effective on GLV-3 current limiting resistors were installed in the abort warning circuit to the spacecraft to prevent possible damage to the command receivers. During a shutdown/abort sequence, the 28-vdc abort signal to the spacecraft could be shorted by the spacecraft guillotine. This could have short circuited the outputs of the receivers and prevented the initiation of the destruct system. The addition of current limiting resistors in the abort warning circuits was designed to keep the current within safe limits (Reference ECP 479).

c. Major Problems

The Gemini Program's newly developed destruct initiator was a three position unit (Safe-Shortoff-Arm) and had to be qualified after the development program. During qualification vibration testing, the destruct initiator was found to be deficient in the motor brake area. The vendor modified the brake disc anchor by replacing wedge cutouts in the disc with vertical cutouts. The brake air gap area was also decreased. Four modified destruct initiators subsequently passed qualification tests satisfactorily. As a result of these delays in this development program, a modified Titan II destruct initiator had to be used on GLV-1.

Four destruct initiators of the second production lot of the new design failed to pass acceptance tests at the vendor's plant due to excessive current demand and switch hang-up in the safe position. The difficulty was isolated to the motor operation. Corrective action was to provide additional motor support to prevent loosening under vibration. Tolerances were tightened on the disc in the motor brake area and improved quality source inspection procedures were implemented. In addition, all of the odd serial number units in the second production lot incorporated motors that were built from hand selected parts, had special burn-in tests and oscillograph pictures taken of armature current while cycling from one position to another. Later, one of the even numbered initiators from this lot failed because of excessive cycling time in a subsystem test at ETR. The three remaining even numbered destruct initiators from the second lot were returned to the vendor to be rebuilt with screened motors.

There were other problems with the initiators, such as, incorrect adjustment of micro-switches and incorrect positioning of the status wafer switch, but these were relatively minor in nature. The other major problem with initiators was found as a result of the failure of serial number 53 to pass a preinstallation test at ETR. In this test the unit failed to cycle out of the safe position to the shortoff position. At first it was suspected to be a microswitch adjustment problem. Further tests at the vendor's facility revealed that insufficient friction in the rocker arm assembly was the cause of failure. Relaxation of pressure between the rocker arm and the microswitch actuating plunger allowed the microswitch to open and thus prevented the initiator from cycling out of the safe position. Rework of the destruct initiators was authorized by ECP 577, effective for GLV-10 and subsequent. In this rework the pivot pin and bearing disc of the rocker arm friction mechanism were machined to a finer finish.

Two studies concerned with range safety ordnance are worth noting. Assessment[12] was made of the r-f hazards to such devices and stray voltage susceptibility was evaluated.[13]

11. MALFUNCTION DETECTION SYSTEM[14, 15, 16, 17, 18]

a. System Description and Design Concept

The Malfunction Detection System encompassed the major inflight launch vehicle malfunction sensing and warning provisions available for crew safety. The function of the MDS was to monitor critical parameters indicative of launch vehicle subsystems performance and, in turn, to supply signals to the spacecraft to energize warning displays and initiate switchover to the redundant flight control system if required.

The launch vehicle functions displayed to the astronauts were:

Launch vehicle overrate (pitch, yaw, and roll)
Stage I engine thrust chamber underpressure (SA-1 and SA-2 separately)
Stage II engine fuel injection underpressure
Stage I propellant tank pressures
Stage II propellant tank pressures
Switchover to the secondary flight control system

The crew had three manual switching functions associated with the MDS. These were: switchover to the secondary flight control system, switchback to the primary flight control system, and launch vehicle shutdown (abort arm).

The implementation of the MDS in the launch vehicle required redundancy of sensors and circuits and independent installation of redundant elements. To further minimize the possiblity of a single failure disabling the system, probable failure modes were considered in component design and selection, and circuit connection to provide the MDS with a greater reliability than that of the launch vehicle subsystems being monitored. The MDS was operational throughout the boost phase of flight. The launch vehicle malfunction sensing and warning provisions and their interrelationships are shown on Figure II. C-27.

b. Functional Description

The makeup of the MDS is shown by the block diagram, Figure II. C-28. The MDS was a composite of signal circuits originating in monitoring sensors, routed through the launch vehicle and the interface, and terminating in the spacecraft warning-abort system. For redundancy, the electrical power for MDS operation was derived from the launch vehicle accessory power supply (APS) bus, 400 cycle bus and the instrumentation power supply (IPS) bus.

The Stage I MDS engine thrust chamber pressure sensors, supplied with the Gemini Launch Vehicle engines, were provided in redundant pairs for each engine subassembly. The warning signal circuits were connected to individual Stage I engine warning lights (red) in the spacecraft. Upon decrease or loss of thrust chamber pressure, the redundant sensor switches closed and initiated a 28-volt d-c warning signal. A similar circuit was provided for the Stage II engine MDS monitoring fuel injection pressure. Performance of the Stage I engine was monitored through the MDS to prevent launch with a low thrust subassembly.

All MDS propellant tank pressure sensors and signal circuits were identical with the exception of the pressure operating range. A redundant pair of sensors was provided for each propellant tank. Each sensor of the pair supplied an analog output signal, proportional to the sensed pressure, through (individual) signal circuits to (individual) pressure indicating meters in the spacecraft.

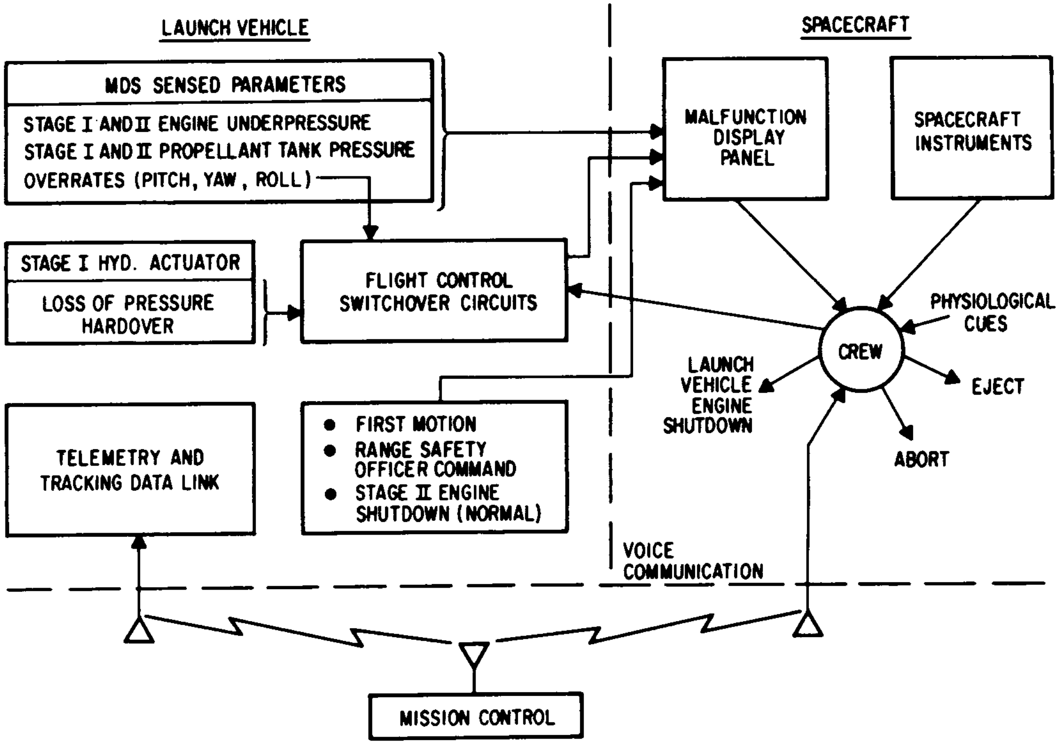


Figure II. C-27. Launch Vehicle Malfunctions - Sensing and Warning Provisions

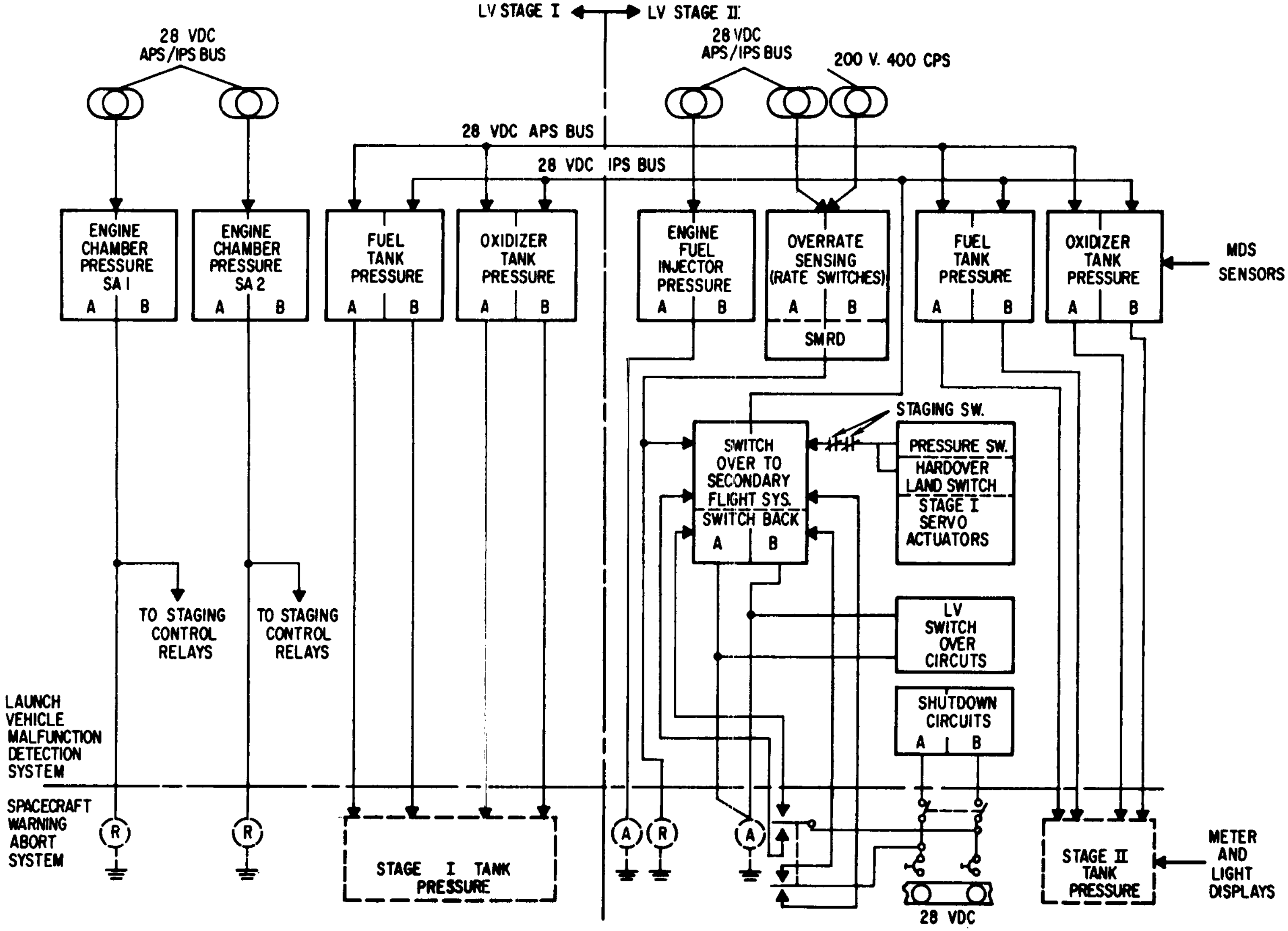


Figure II. C-28. Malfunction Detection System, Block Diagram

A Prelaunch Malfunction Detection System (PMDS) was associated with the Stage I engine's propellant tank pressurization system to monitor autogenous system performance during the engine start period. Performance of both fuel and oxidizer tank pressurant gas supplies was monitored so as to prevent liftoff in the event of an autogenous system malfunction.

Launch vehicle turning rates about all three axes were monitored by the MDS. In the event of excessive turning rate (overrate), a red warning light in the spacecraft was energized. Simultaneously, a signal was provided to initiate switchover to the secondary flight control system. Six such gyros were used - a redundant pair for each of the vehicle body axes (pitch, yaw and roll). Two rate levels of switch actuations were provided: a low rate level for Stage I boost flight, and a high rate level for Stage II flight. In the MDS circuits, the redundant (rate) switches were series-connected (low to low; high to high). This required closure of both switches in the redundant chain to initiate switchover. The low rate switch circuits were disabled at staging. Since the predominant failure mode of the rate switch gyro was failure of the gyro spin motor, a failed gyro bypass circuit (spin motor rate detector, (SMRD)) was associated with each gyro to permit continued MDS rate sensing and overrate signalling using the redundant (operable) rate switch gyro.

The dual switchover power amplifiers were self-latching, solid state switching modules used to initiate a switchover from the primary to the secondary guidance and flight control system. On the input side, signals were supplied from the MDS overrate circuits, the Stage I hydraulic actuators (low pressure or hardover) or from the astronaut in the event of a malfunction. An unlatching capability was provided for the switchover power amplifiers to permit switchback from the secondary to the primary guidance and flight control system during Stage II boost flight.

Launch vehicle shutdown could be manually initiated by the astronauts in the event of a determined or recommended mission abort or escape requirement.

c. Configuration

(1) Launch Vehicle MDS Components

The Gemini Launch Vehicle MDS contained the following major components:

Rate switch package - (overrate sensor)

Malfunction detection package - (SMRD monitor, switchover and switchback circuits)

Fuel tank pressure sensors (Stage I and II)

Oxidizer tank pressure sensors (Stage I and II)

Engine thrust chamber pressure sensors (chamber pressure for Stage I and fuel injection pressure for Stage II engines)

Stage I and II separation disconnects

All of the above MDS components were necessarily peculiar to the Gemini Launch Vehicle. The particular designs or modifications to existing Titan II designs were explicitly chosen for the MDS on the basis of functional characteristics, simplicity, reliability, failure modes, and survival in the GLV flight environments. Location of the MDS components in the GLV is shown on Figure II. C-29.

(2) Comparison to Titan II

The MDS was peculiar to the Gemini Launch Vehicle and was designed and installed as a subsystem for crew safety - a consideration not applicable to the Titan II booster. As an additional subsystem to the booster, the entire MDS was identified as a Titan II deviation. There were no like components between the Gemini Launch Vehicle MDS and the standard Titan II boosters. However, the component experience from Titan was applied to the selection and design of the MDS components.

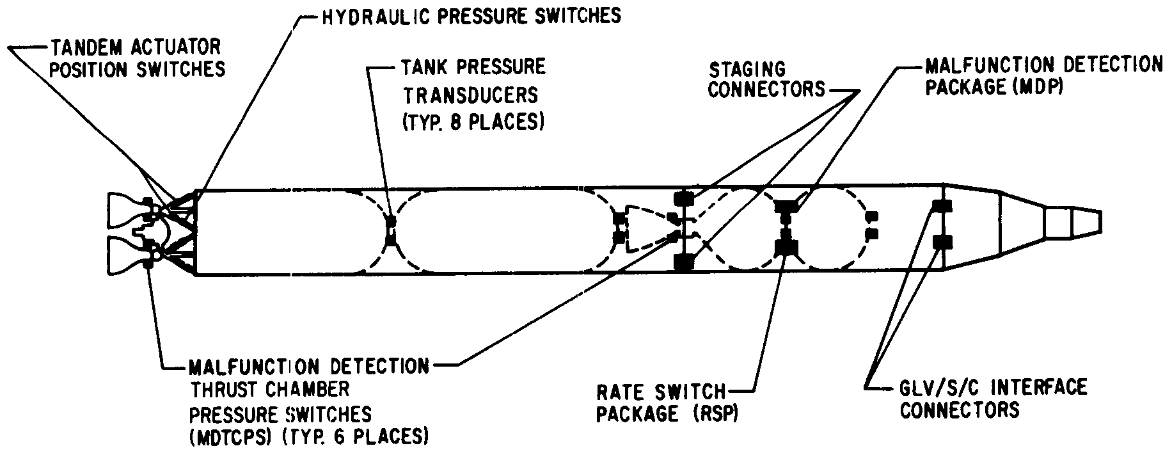


Figure II. C-29. Launch Vehicle MDS Sensor and Equipment Location

d. Component History

(1) Evaluation and Pre-Qualification

All major system components were subjected to extensive engineering evaluation tests prior to selection and preparatory to qualification attempts. For example, a SMRD module relay was replaced with a part designed to withstand higher levels of vibration. This procedure permitted selection of the best available components and thereby minimized qualification problems.

(2) Airborne Systems Functional Test Stand (ASFTS) Testing

ASFTS testing was directed toward integration of the MDS with the other vehicle systems and evaluation of their performance and compatibility. These tests revealed problems with the rate switch package (RSP) such as variation of setting with time and electrical interference with other systems. These situations were corrected by design changes to the RSP. No problems of any consequence were uncovered with the rest of the MDS components during this test phase.

(3) Qualification

The rate switch package experienced the greatest difficulty during qualification testing. The most severe problems were those of rate switch contact resistance, gyro rotor slowdown, SMRD voltage drift and variation of rate switch set point. Considerable redesign effort was involved in solving all of these problems. For example, elimination of gyro rotor slowdown required the addition of vibration isolation "O" rings within the gyro housing. The upgrading of quality control and inspection, improvement of assembly and test procedures and introduction of stringent cleanliness control also contributed toward resolution of these problems permitting rate switch qualification.

Conditions like noisy signal output during vibration and excessive calibration drift over the operating temperature range were corrected during qualification test of the propellant tank pressure transducers. Units incorporating all of the corrective actions subsequently completed all qualification tests successfully.

Problems of minor significance were resolved during qualification of the malfunction detection package and other MDS components. Timely solution of all these problems resulted in the entire Malfunction Detection System being qualified for manned flight.

(4) Piggyback Flight Tests

The Malfunction Detection System components were "piggyback" flight tested on five Titan II missiles. This permitted early performance evaluation of the system in the actual flight environment. These tests satisfactorily demonstrated the component operational configuration.

(5) Vendor Acceptance Testing

Numerous failures, particularly of the rate switch package have occurred during component acceptance test. All such failures were reported and reviewed in detail during periodic reliability reviews for impact on existing hardware and corrective action was implemented. Typical of this type of failure was a problem uncovered with the MDP during PMT vibration testing. A power amplifier transistor exhibited intermittent shorting due to internal contamination by loose weld flash particles. A recurrence of this failure was minimized by procurement control of the transistor. The controlled part was incorporated in all flight malfunction detection packages. The reliability reporting and corrective action system was effective in maintaining a high level of component integrity.

(6) VTF and ETR Testing

VTF and ETR testing was accomplished without major difficulty on the MDS for GLV's 1 through 12. Minor launch vehicle problems like electrical interference of the rate switches with the telemetry system were discovered and resolved in VTF. A compatibility problem between the MDS engine sensor circuit and the MDS test set (AGE) was also discovered in VTF and the test set modified prior to GLV-2 launch.

Three engine MDS under-pressure sensor switches failed due to current overload during test of GLV-1 and -2. Investigation revealed that the failures were associated with inadvertent interconnection of the launch vehicle MDS circuits and the (AGE) engine test set while electrical power was applied to the launch vehicle. A similar situation occurred on GLV-5 when 4 MDS switches received a current overload as a result of work being performed in the spacecraft with vehicle power on. Caution notes were added to the test procedures to preclude recurrence of the situation. Test personnel were cautioned to follow the applicable checkout procedures precisely as written.

A number of MDS engine pressure switches and tank pressure transducers were removed from GLV during the course of checkout testing at VTF and ETR. Most of the failures were associated with long term corrosion or an increase in contact resistance from contamination or due to switch inactivity. Although no positive corrective action was taken, the checkout testing was effective in eliminating questionable hardware so that no component failures were incurred in flight.

(7) Reliability-Testing

Failure mode testing was conducted on the propellant tank pressure transducers, MDP, RSP and MDS stage separation disconnects. In addition, an extended life test program was conducted on the RSP and MDP. As a result of the extended life test program, the allowable operating time on the rate switch package was increased from 500 hours to 1000 hours effective GLV-7 and subsequent.

(8) Battleship Tank Tests

This program included 135 successful propellant tank pressure sensor cycles in the propellant environment under actual engine operating conditions.

No problems were encountered during 101 Engine Under Pressure Sensor actuations.

Twenty-one satisfactory tests were run on the PMDS

(9) Flight Testing

No major MDS problems or failures were incurred in flight; however, review of the flight data from GLV-1 through -5 revealed a potential PMDS problem with the oxidizer pressurant pressure switch (OPPS) resulting from a previous design change made in the Titan II autogenous pressurization system. The GLV flight data indicated that the OPPS set point limits were not compatible with engine pressure climbout characteristics and could result in an on the pad shutdown. A change was made in the pressure oxidizer pressurant orifice inlet (POPOI) back pressure orifice to correct the condition for GLV-6 and subsequent.

As a result of a completely successful flight program, operation of the MDS warning and switch-over capability were never demonstrated during powered flight. A review of flight data from each flight, however, confirmed that the MDS systems were in proper operating condition. During many flights, excessive vehicle turning rates after spacecraft separation indicated that the RSP was functioning within design tolerances.

e. Crew Safety Systems

In addition to the GLV Malfunction Detection System, the following functions were integrated into a comprehensive operational plan designed to optimize crew safety during prelaunch activity and powered flight. These functions supplement the MDS by providing early detection and warning of any unusual condition.

(1) Pad Egress

GLV test conductor procedures were established to facilitate astronaut egress or seat ejection in the event of a preliftoff incident necessitating evacuation from the spacecraft. A voice communication link was maintained with the astronauts to keep them informed of the situation and its status. This capability was demonstrated by the securing operation following the GLV-6 on-pad shutdown.

(2) Test Conductor Abort

Visual monitoring was established to detect any anomaly occurring during the early portion of powered flight. The astronauts would have been advised accordingly by voice communication and spacecraft abort light illumination in the event of an impending catastrophe.

(3) Slow Malfunction Monitoring

Slow malfunction (guidance) monitors continually monitored trajectory, guidance and flight control information using displays and real time recordings. This function was effective throughout powered flight to preclude an abort or mission anomaly by recommending switchover to the redundant guidance and flight control system.

(4) Booster System Monitoring

Various booster parameters (e. g., propulsion functions, tank pressures, voltages, etc.) were monitored on real time displays to anticipate the requirement for a possible abort. Voice communication was maintained with the spacecraft to advise of booster status.

f. Conclusions

Of significant importance were the five MDS piggyback flights which provided early information of MDS behavior under actual flight conditions. The minor anomalies which were encountered were investigated and corrected and the fixes tested during subsequent flights.

MDS performance during all flights indicated a progression toward attaining a faultless MDS installation through stringent qualification and reliability testing and numerous checkout tests at VTF and ETR.

The MDS, though never utilized in a catastrophic flight environment, achieved a 100% flight reliability record for component performance with every indication that the total system was satisfactory for the assigned crew safety function during powered flight.

As a result of the problems encountered during component and system development combined with the close attention given the examination and evaluation of the involved areas, a first hand and intimate knowledge of the MDS was gained by its designers. This relationship to the actual components and installation provided the highest degree of assurance that the MDS would function as intended.

Since the MDS was designed primarily as a warning system for use in conjunction with the manual abort technique, its effectiveness depended largely on the training and conditioning of the crew. The excellent and thorough crew training prior to each flight was demonstrated during the initial launch attempt of GT-6A when a false liftoff signal was generated by inadvertent disconnect of a tail-plug connector. Engine start and shutdown was evident on the MDS display panel in the cockpit. An unnecessary and potentially dangerous Mode I abort was avoided by the command pilot's "seat of the pants" realization that liftoff had not occurred.

The entire MDS was visual in nature, utilizing gauges and lights to indicate a system's proper function or impending failure. It was therefore critical to solve the POGO problem in order for the displays to be clearly visible. The POGO levels of vibration were reduced to an acceptable level by the use of propellant feed system damping devices.

The PMDS (Pre-Launch Malfunction Detection System) was limited in effectiveness by the nature of the design placement of sensors. The sensors for determining sufficient autogenous gas flow to the propellant tanks were placed at the engine interface. As a result of this, a line failure downstream of the engine interface might not have been detected.

12. CREW SAFETY

a. Slow Malfunction Detection

The redundant guidance and flight control systems within the Gemini Launch Vehicle presented a unique crew safety problem. Many failure modes in the primary launch vehicle systems were detected by the airborne Malfunction Detection System which would initiate an automatic switchover to the redundant system. However, for some failure modes and during specific times of flight, the resulting vehicle disturbance would have been insufficient to initiate a switchover. Failure mode analysis indicated that approximately 15% of all launch vehicle failures would cause a slow malfunction, resulting in a departure from a nominal trajectory.[19,20,21] To effectively retain the benefits of systems redundancy for this type of malfunction, provisions for an astronaut-initiated manual switchover capability were incorporated. As an aid to making the manual switchover decision, a mission control slow malfunction detection monitoring function which would analyze real time telemetry and trajectory data was established. To attain this objective, the launch vehicle contractor was directed to perform studies to identify the causes and effects of slow malfunctions with respect to mission objectives, perform analyses to establish a monitoring capability, aid in the establishment of a monitoring facility, establish a training facility for training monitors and provide suitably trained monitor personnel for simulations and launch operations.[22]

The analysis of slow malfunction causes and effects in relation to hardware and performance limitations led to constraint identification.[23] Constraints could be classified into two categories - mission-related and hardware-related constraints. A minimum orbital abort requirement of an inertial velocity equal to that required for 1-1/2 orbits less 100 fps was used to establish the mission-related performance constraints. For the rendezvous missions, wedge angle and out-of-plane velocity also became considerations for mission-related constraints. Range safety corridors likewise were defined as trajectory constraining factors. Hardware limitations such as structural, thermal, radar look angles, spacecraft aerodynamics abort limitations developed into hardware-related constraints.[24]

Slow malfunction computer simulations were used extensively in the actual construction of the wedge angle and performance constraint lines. In these cases, by simulating a specific type of slow malfunction, switching over and evaluating the SECO +20 sec (insertion) conditions, a locus of switchover points having the same SECO +20 sec conditions was obtained for different switchover times. This locus then became a constraint line bounding the nominal unbiased trajectory. In order to create a continuous envelope of constraints, procedure lines were constructed between the various constraint segments. This precluded cases where a diverging trajectory would miss a specific constraint, but might violate another constraint at a later time. Thus a network of multi-dimensional constraint requirements was resolved with respect to the same parameters for ease of operational handling and to facilitate display. Figures II. C-30 and II. C-31 typify recent constraint displays for Stages I and II in the pitch (V-Γ) and lateral planes.

Performance-oriented constraints were developed about the nominal predicted launch vehicle trajectory for each mission. This trajectory was then biased using launch wind predictions, pitch programmer characteristics, and Stage I engine thrust data. This biased trajectory was provided to the slow malfunction detection monitors as the launch reference trajectory. Corresponding wind-biased analog traces of monitored telemetered functions were also provided for reference.

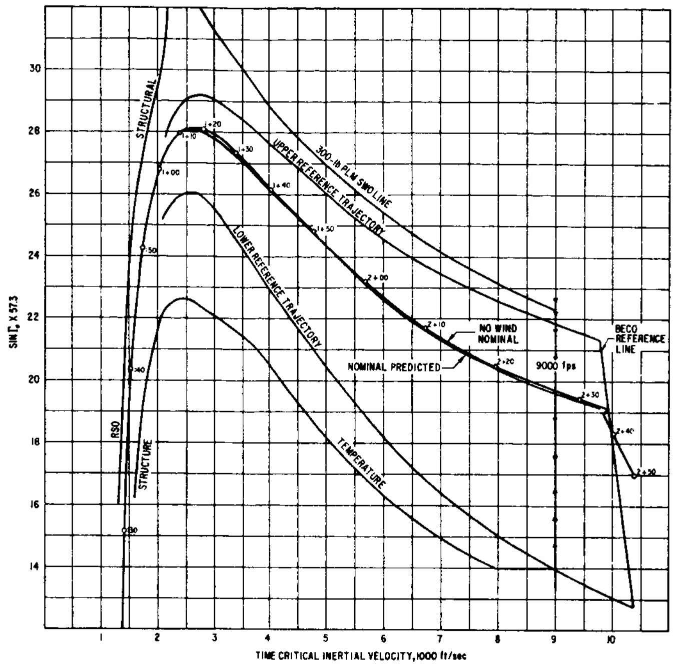


Figure II. C-30a. Stage I Pitch Plane Slow Malfunction Detection Constraint Plotboard

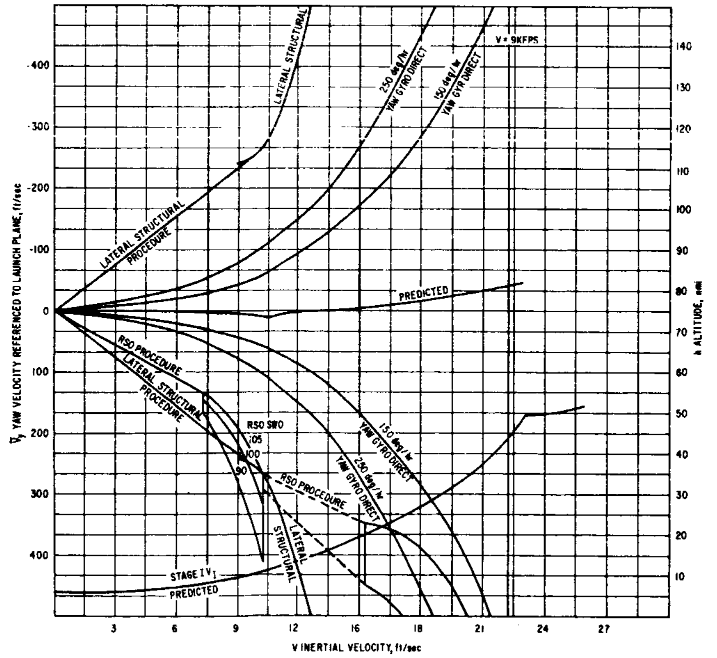


Figure II. C-30b. Stage I Lateral Plane Slow Malfunction Detection Constraint Plotboard

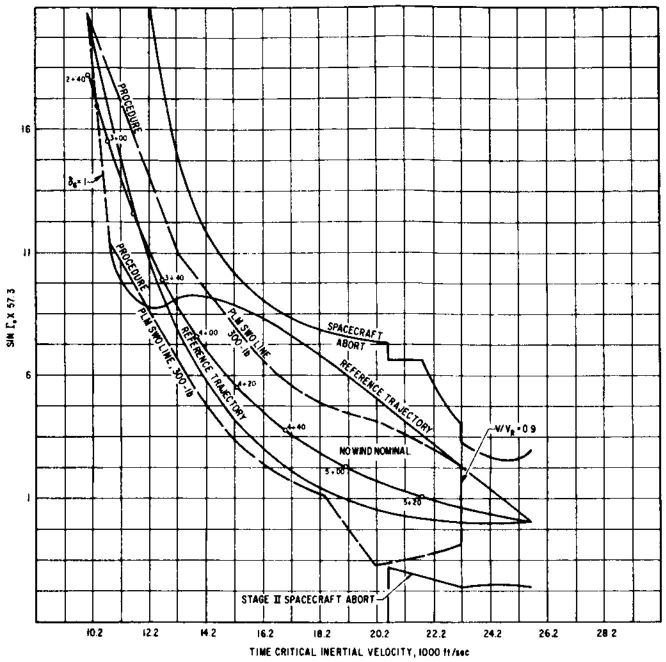


Figure II. C-31a. Stage II Pitch Plane Slow Malfunction Detection Constraint Plotboard

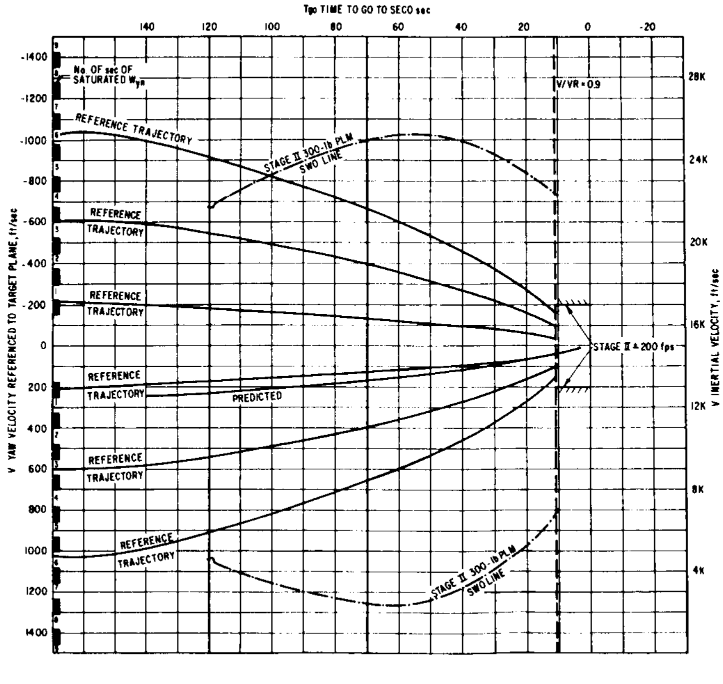


Figure II. C-31b. Stage II Lateral Plane Slow Malfunction Detection Constraint Plotboard

In addition to the actual trajectory on the constraints display, real time display of discretes and telemetered parameters were made available to the monitors via two strip chart recorders, digital displays and flag indicators located on the monitors' consoles (see Figure II. C-32). The telemetry and Burroughs quantities monitored are listed below:

Real-Time Telemetry Readouts -- Guidance Monitor Console (MCC)

Stage I pitch deflection (primary)
Stage I yaw deflection (primary)
Stage I yaw deflection (secondary)
Stage I pitch deflection (secondary)
Stage II yaw deflection
Stage II pitch deflection
Stage II roll deflection
TARS guidance initiate
Primary gain change
Primary pitch rate program
Primary roll rate program
IGS pitch error output
IGS yaw error output
IGS roll error output

Decoder pitch rate output
Decoder yaw rate output
Adapter pitch error output
Adapter yaw error output
Adapter roll error output
Secondary yaw autopilot output
Secondary pitch autopilot output
Secondary pitch autopilot output
Secondary yaw autopilot output
Secondary gain change
Stage I-II separation
MDS amp "A" switchover
MDS amp "B" switchover
TCVPSV Stage II shutdown

Real-Time RGS Ground Computer Readouts -- Guidance Monitor Console (MCC)

Radar position tracking flag

Radar range rate flag

Radar lateral rate flag

Liftoff (LO) (First motion)

Guidance initiate steering

SECO transmit

Radar quality flag

Magnitude of inertial velocity of missile, extrapolated linearly 0.789 seconds ahead in time

Sine of inertial flight path angle multiplied by 57.3 & extrapolated linearly 0.789 seconds ahead in time

Sine of inertial flight path angle multiplied by 57.3

Magnitude of inertial velocity of missile, extrapolated linearly 0.789 seconds ahead in time

Inertial velocity normal to an inertial plane containing the launch point vertical and headed along the launch azimuth minus the same velocity prior to LO

Component of inertial velocity normal to target plane plus a bias to compensate for lateral cg effect on Stage II guidance inaccuracies

Total pitch torquing rate being transmitted from ground to missile

Injection altitude component of W_{PN}

Pitch attitude error component of W_{PN}

Total yaw torquing rate being transmitted from ground to missile

Elapsed time after LO

Time-to-go to SECO

These functions were denoted as either mandatory or highly desirable for launch operations in the Systems Test Objectives document and NASA Mission Rules.

(1) Constraint Validation

Prior to the finalization of mission constraints, an independent validation procedure was initiated by Aerospace Corporation to assure constraint validity. Using the contractor-prepared constraints with Aerospace's computer routine, updated for the current secondary system ascent trajectory program, slow malfunctions were injected, switchover performed and resultant insertion conditions compared with the contractor's. This validation procedure continued as required for each mission.

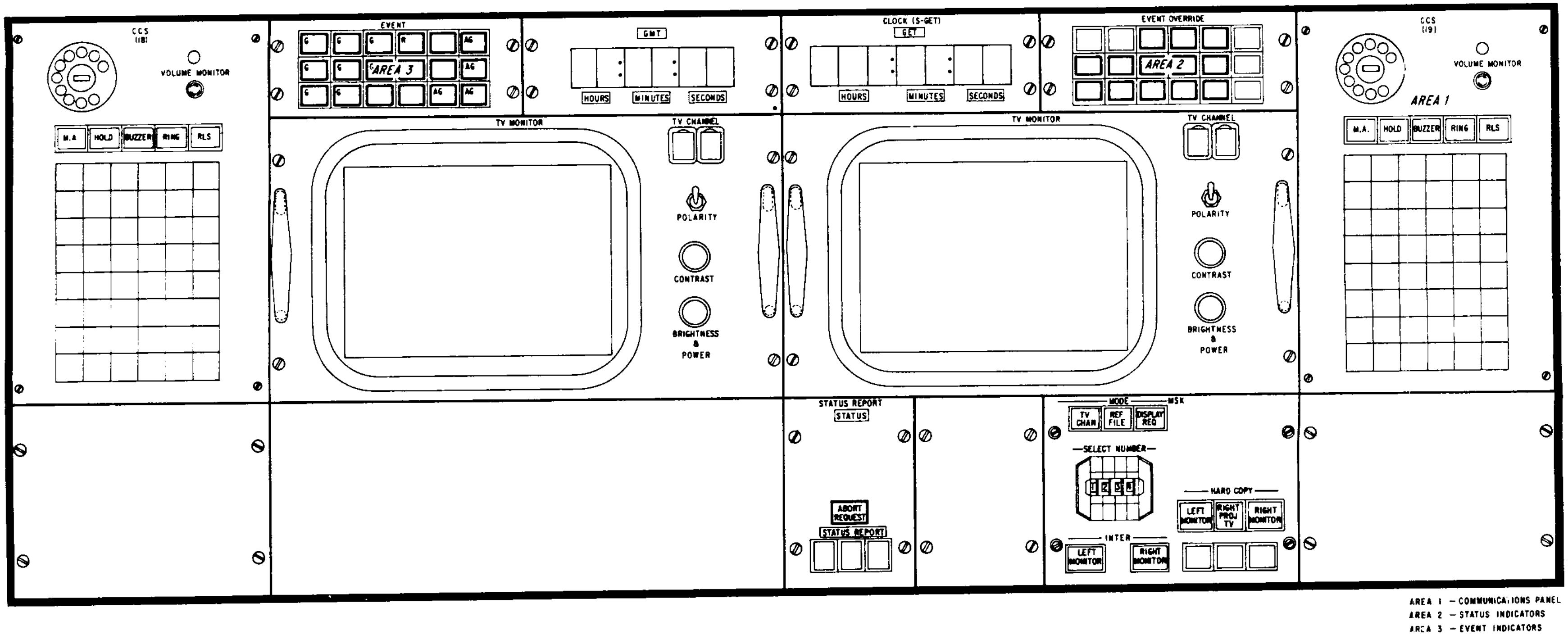


Figure II. C-32. Guidance Monitor Console, MCC - Houston

(2) Switchback

Effective on the first manned flight (GT-3), a modification to the flight control system was initiated which provided the spacecraft crew with the capability of switching back to the launch vehicle's primary guidance and control system after staging, prior to RGS closed loop guidance initiate. It was agreed that switchback would be attempted only if the switchover were due to a failure of Stage I primary hydraulics. This innovation further enhanced mission success in the event of this malfunction mode.

(3) Monitors

The slow malfunction detection monitors (Guidance Monitors) observed selected discrete and analog parameters that described the operating and performance characteristics of the primary and secondary guidance system during powered flight. They analyzed real time telemetry, trajectories and ground computer signals; informed other Mission Control Flight Controllers and the Flight Director of guidance and flight control systems' status; recommended guidance switchover or switchback when warranted; and acted as required to maximize mission success.

(4) Monitor Training [25, 26, 27]

M/B provided trained monitor personnel for active participation during all Gemini launches. To this end, a monitor training program and training facility, complete with constraint plotboards and recorders similar to the Mission Control Guidance Monitor's console, were established at Baltimore. Training tapes simulating telemetry and radar data of normal and abnormal ascent trajectories, were developed to provide inputs to this console. Contractor personnel were trained utilizing this simulator as were Aerospace and NASA monitors. This training was supplemented by active participation in all exercises at Mission Control Center - Houston. The utility of the Martin Mission Simulator cannot be overemphasized. It provided, within the contractor's facility, a practical means of confirming monitor proficiency and constraint practicality.

(5) Allied Monitored Functions

In addition to the slow malfunction detection monitoring activity, two additional monitoring activities relating to booster systems were operationally implemented at Mission Control. These functions (booster systems engineer and tank pressure monitor) were staffed by NASA FOD personnel and technically supported by the Martin Company.

The booster systems engineer and tank pressure monitor provided real time evaluation during powered flight of the primary and secondary d-c power sources, primary and secondary hydraulic systems pressure, engine(s) thrust chamber pressure and propellant tank pressure as well as other telemetered functions relative to launch vehicle operation. Predicated upon evaluation of these booster systems performance, the monitors would communicate to the Flight Director, other controllers, and to the astornauts via Capsule Communicator information relating to an abort condition, hydraulic switchover, a "no-go" switchover condition or engine malfunction.

Abort criteria, vehicle information, and recommendations relating to these monitored functions were prepared by the Martin Company evolving from Abort Panel and Guidance Constraint Meetings. Specific abort criteria was established and approved procedures delineated in and controlled by Martin's LV 376 Part B, Booster Abort Criteria, [24] and NASA's FOD Mission Rules. Telemetry requirements for support of these functions were defined and priority controlled by the STO and Mission Rules.

Specific launch information relative to these functions such as predicted engine thrust characteristics and predicted wind influenced tank pressure curves were provided to the monitors prior to each launch.

(b) Conclusions and Recommendations

Since, throughout the course of the Gemini Program, no malfunction occurred which necessitated or precipitated an in flight switchover or abort, it is extremely difficult to evaluate the net worth of the slow malfunction detection monitoring effort. However, even as the program progressed and the probability of a malfunction occurring decreased, the possibility still remained. Operator training was never relaxed. The system and techniques were constantly improved and refined. In the event a malfunction had occurred, the system and personnel would have detected the condition, evaluated it, and initiated appropriate action to insure the astronauts' safety and mission success.

In retrospect, there are several areas which could have been explored and evaluated in simplifying the slow malfunction detection effort. It would have been beneficial to define the slow malfunction detection requirements early in the program and influence the "on board" instrumentation and telemetry. Displays also were evolved around available range data sources.

Secondly, the monitors were required to observe several events simultaneously during powered flight to evaluate slow malfunction conditions. It would have been desirable to engineer the system so that the information displayed was reduced, better grouped and simplified for evaluation purposes.

13. TECHNICAL INTERFACE MANAGEMENT

This subject is discussed in general terms from the Management standpoint in Section II.A-2. The discussion which follows provides a more detailed description of the functioning of the Interface Groups and recommendations for future implementation of such groups.

Through the establishment of the Interface Panels and Working Groups, a common technical ground was created to develop the Interface Specifications used in the Gemini Program. Appropriate responsible representation by the contractors and the managing agencies provided a method for initially defining interface parameters. Each contractor was assigned the task of defining his technical interfaces and outlining the requirements for technical information required from other contractors. Frequently this data would emanate from other subsystem panel or working group areas, such as Structures, Electrical, AGE, Guidance. Rough draft specifications were prepared incorporating technical information on mechanical, electrical, software and events timing. Interface drawings were prepared and procedures established for interface control purposes. In addition to defining all airborne and AGE interfaces, the specification established the requirement for and control of simulation equipment and specialized tooling each contractor was to provide.

Draft interface specifications were reviewed, corrected, re-reviewed, negotiated, arbitrated and in some cases contractors were directed to comply with provisions of the draft. This effort ultimately resulted in contractually and technically acceptable documents. Once this had been accomplished, formal interface specification and drawing control was implemented and any changes required configuration management processing and specification control.

Interface verification and integrity testing was accomplished as an integral part of vehicle and acceptance testing in VTF. Test data as applicable to the interface was reviewed extensively prior to vehicle acceptance. In addition to actual test data, interface simulators and test tooling configurations were reviewed to assure specification compliance. Since interface functions cross numerous subsystem areas, close coordination was maintained with other cognizant personnel to ensure compliance with specification requirements.

After vehicle acceptance and delivery to the Cape, further interface verifications were performed during subsystem and system test prior to spacecraft mate.

Early comprehensive technical definition and effective coordination precluded major interface problems impacting the program. Minor problems did occur but these were identified early during interface verification, negotiated and resolved. The Gemini interface specifications were well-organized and comprehensive. The program demonstrated the necessity of having three major items affecting interface compatibility complete and well defined. Those three areas are 1) controlled tooling for mechanical interface verification, 2) interface electrical schematics delineating all circuit elements described in the specifications, and 3) negotiated and contractually controlled test procedures for verifying both sides of the interface and the interrelated functional compatibility. Limiting any of these areas reduces interface effectiveness.

14. AEROSPACE GROUND EQUIPMENT

a. Martin Baltimore and ETR

The general requirements for checkout of the GLV at the contractors Vertical Test Facility (VTF) and at ETR were predicated on the following:

1) The launch vehicle shall be a modified version with essentially the same performance specifications as the present Titan II ICBM, retaining the general aerodynamic shape and basic systems and propulsion concepts (AFSSD Exhibit No. 62-12, Statement of Work). Changes to the launch vehicle shall include:
   - (a) Addition of new systems or modification to present systems as required to assure pilot safety during the countdown, launch, and operation through powered flight.
   - (b) Changes which will improve the probability of mission success.
2) Presently-developed Titan II launch vehicle auxiliary equipment shall be used to the maximum extent possible.
3) AGE for prelaunch, launch control and checkout shall use presently designed or developed hardware to the extent of availability and adequacy. However, this does not preclude the development of new equipment where cost and/or time advantages may be realized without degradation of the over-all program reliability of schedules. New AGE shall be developed to fulfill unique requirements of the Gemini Launch Vehicle. Special attention shall be devoted to new AGE required to reduce launch checkout and countdown times.
4) All equipment, including the AGE checkout and launch equipment, shall be built to approved specifications as described in the documentation paragraph (AFSSD Exhibit 62-12, Paragraph 5.1.2).
5) Integrate the Government-furnished equipment (GFE) into the auxiliary equipment (SSD Exhibit 62-12, Paragraph 5.1.9).

The major test objectives to be accomplished were:

1) VTF (Figure II.C-33)
   - (a) Verify the functional operation and integrity of the GLV subsystems (Subsystem Functional Verification Tests).
   - (b) Verify the functional operation and compatibility of the GLV subsystem during a simulated countdown (Combined Systems Acceptance Test-trial runs).
   - (c) Demonstrate to the customer the functional operation and compatibility of the GLV subsystems during a simulated countdown, launch and flight (Combined Systems Acceptance Test).
2) ETR
   - (a) The GLV shall be checked out and launched from Complex 19, ETR. This complex was modified, as required, to serve both the launch vehicle and the spacecraft (Reference AFSSD Exhibit 62-12, Paragraph 4.5, 27 December 1961).

Details of the AGE utilized on the Gemini Program can be found in Martin Documents LV-33, "AGE Plan-Checkout and Launch Control (ETR)"; LV-34, "AGE Plan-Checkout and Launch Control (VTF)"; LV-38, "AGE Plan-Erector and Umbilical Tower Operating Ground Equipment"; and LV-30, "AGE Plan-Ground Instrumentation Equipment (VTF and ETR)."

Basically, the Gemini checkout philosophy called for a decentralized approach; i.e., an equivalent equipment was provided to check the corresponding airborne system. This relationship of the various airborne systems and the checkout equipment is illustrated in Figure II.C-34.

Each checkout set could operate on its equivalent airborne system virtually independent of other equipment. The only time that the checkout equipment used automatic operation was during critical events or time periods; consequently, the system was essentially a manually operated one. However, during the countdown phase, all operations performed by the checkout equipment were coordinated by the launch control equipment.

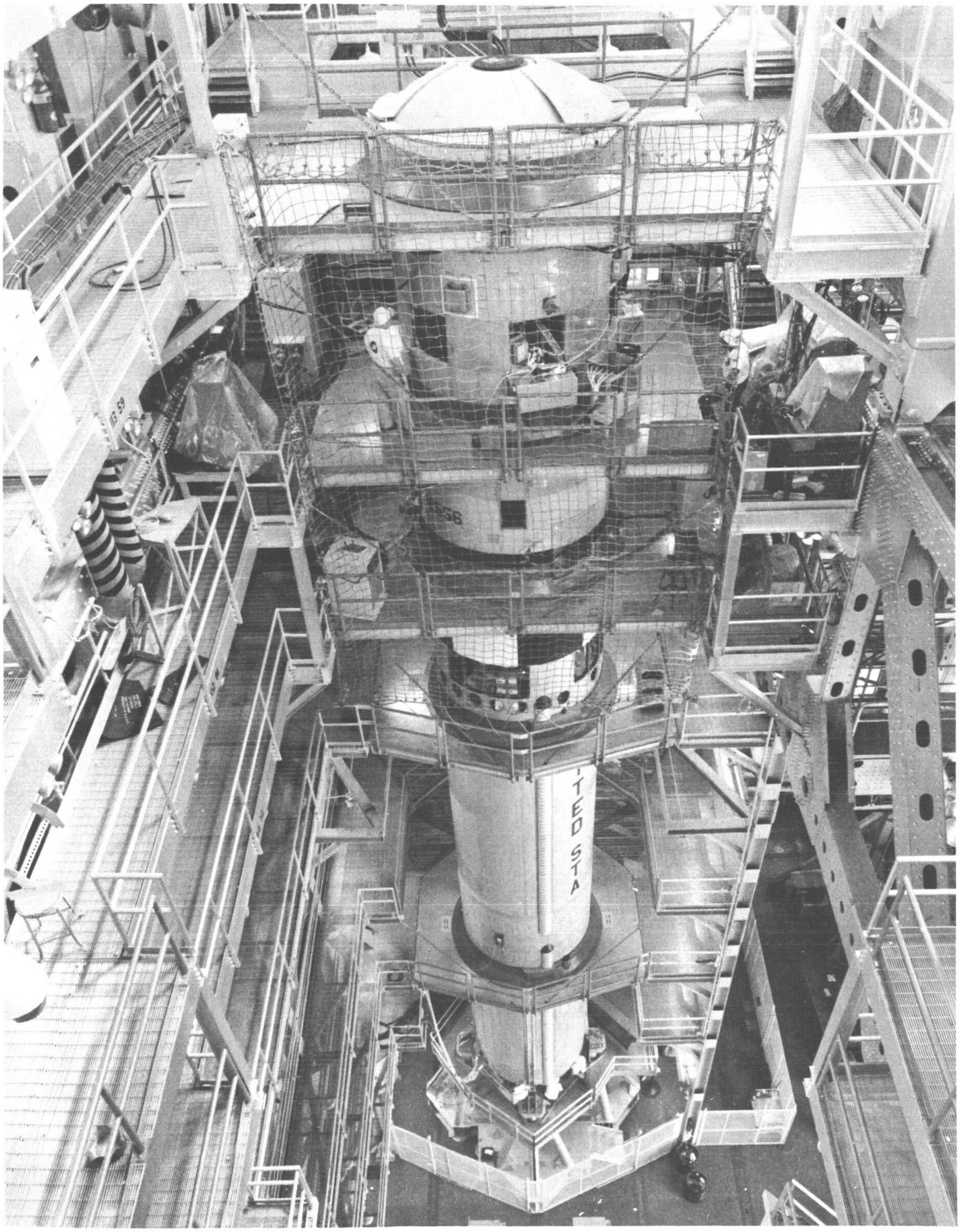


Figure II. C-33. VTF

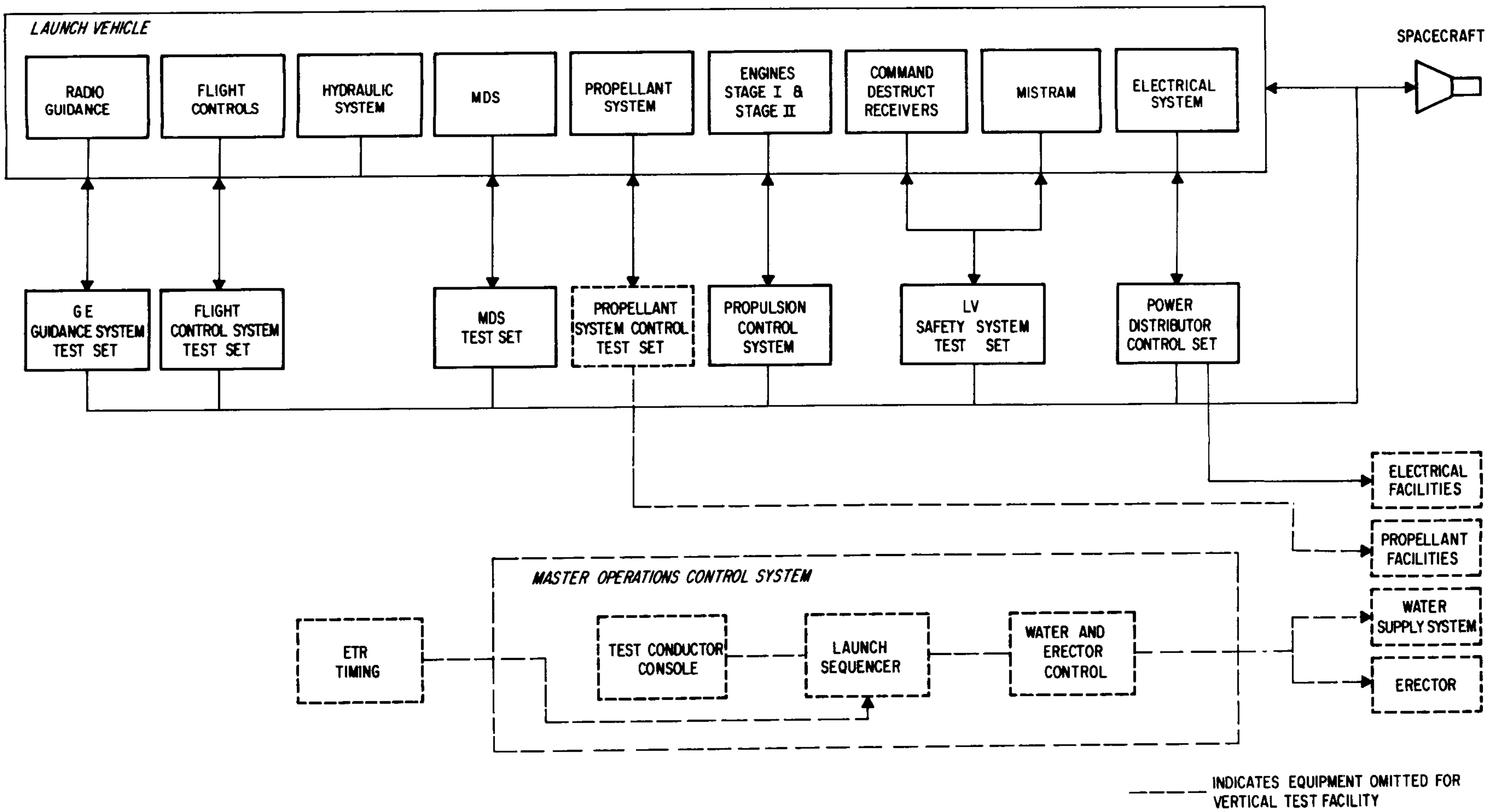


Figure II. C-34. Checkout and Launch Control Systems - Simplified

Launch control was obtained through the use of the Master Operations Control System (MOCS) and other related equipment, including closed circuit television and a community time display board. The MOCS coordinated the time sequence for checkout of the launch vehicle, provided remote control of facilities such as the water supply system and erector, and displayed the state of readiness of the entire complex activity as the various time check points were reached. Through the use of hold-fire and shutdown circuits, the system was used to permit or inhibit launch. Since actual launch functions were not required at the contractor's facility, the MOCS was peculiar to ETR.

b. Support Areas

(1) Airborne Systems Functional Test Stand (ASFTS)

This facility was located at Baltimore in the VTF and is described in Section II. C-16.

(2) Gyro Test Facilities

Acceptance testing of gyro components, i.e., TARS and rate gyros, was accomplished in Baltimore in what was designated the Gyro Laboratory. Test equipment was provided for static and dynamic testing under ambient conditions. Commercially developed test equipment (e.g., rate tables, measurement devices, power supplies) was used in conjunction with Martin developed control panels and readouts which were rack mounted.

A duplicate facility was provided at ETR for the checkout of flight hardware after shipment from Baltimore. Identical test procedures and test set verification procedures were also used at ETR.

(3) General

In addition to the above, certain basic support areas were provided at both the contractors facility and at ETR. These included a battery laboratory, clean room, receiving inspection test areas, controlled storages, etc.

c. Facilities Modifications

Complex 19, (Figure II. C-35) prior to assignment as the Gemini Launch Complex, was a Titan I site. Considerable modification was therefore required to convert the facility. The major changes were associated with the different propellants used by the GLV, and changes associated with the spacecraft and its operational safety. The following paragraphs summarize in more detail the modification program.

(1) Erector

Considerable modification was required to the Complete Vehicle Erector to accommodate the new vehicle. First, a protective enclosure for the spacecraft was required, and an additional elevator for access to it. Changes in work platforms were necessary, and modifications to the structure itself and the actuation system due to the added weight on the erector.

(2) Propellant Handling

New holding facilities, transfer, and distribution facilities were required for the Gemini vehicle. The Titan I fuel facility was removed, and the existing oxygen facility converted for spacecraft liquid and gaseous oxygen requirements.

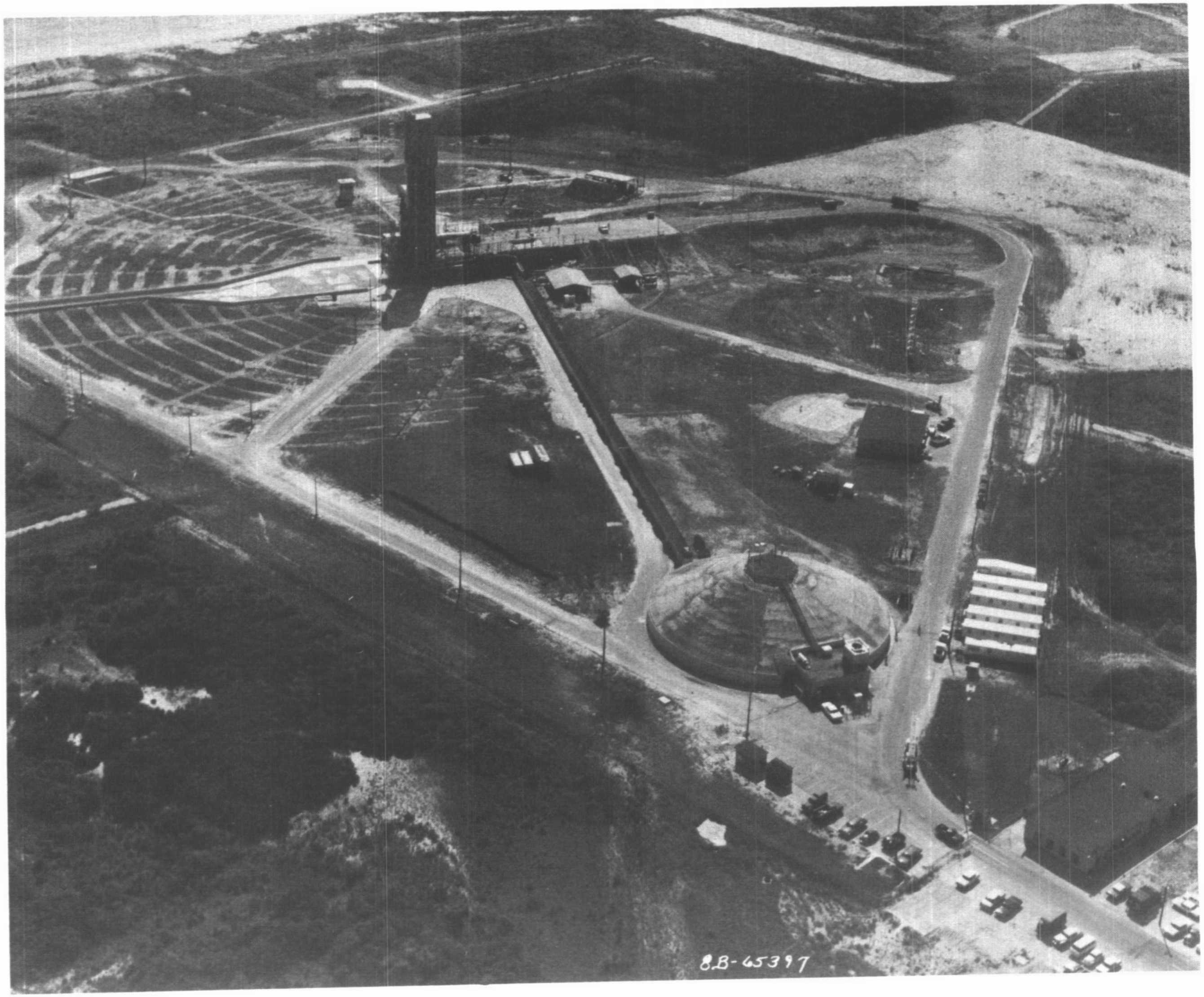


Figure II. C-35. Complex 19

A new holding area was added for spacecraft liquid hydrogen requirements. The gaseous nitrogen supply was increased by converting existing helium facilities. Additional nitrogen was required for propellant handling, tank pressurization, water control, and spacecraft needs.

(3) Water Systems

Because of the highly toxic propellants, water flow requirements were increased. This required enlarging and leak-proofing the flume and skimming basin. Additional spray and washdown facilities were required to service the spacecraft, the propellant holding and distribution areas, and washdown of any area where propellants might be spilled. In addition, a man-rated water spray system was added which encompassed the immediate area around the launch stand. This system was to provide the astronauts protection from fire and propellants in case of an on-the-pad abort.

(4) Decontamination

A facility was required for personnel decontamination when working with propellants. In addition, emergency showers were provided throughout work areas in case of accidental exposure to propellants without adequate protective clothing.

(5) Air Conditioning

An additional air conditioning facility was required to provide cooling to the spacecraft enclosure. Modifications were also required to the existing launch vehicle air conditioner.

(6) Other Changes

Many other changes were required to service the different configuration of the vehicle, but which provided essentially the same function as for the Titan I. This included such items as electrical facilities, umbilical configuration, control cables, and personnel accommodations.

15. RELIABILITY TEST PROGRAM

A Reliability Test Program was contractually required of the Martin Co. on selected items in order to provide adequate assurance that the reliability goals would be achieved. This test program consisted of Environmental Life Testing (ELT) and/or Failure Mode Testing (FMT) on ten (10) Gemini peculiar airborne components.

ELT involved testing the selected components at simulated flight environment, vibration and/or temperature, equivalent to 100% of the qualification levels or Production Monitoring Test (PMT) levels as defined by the individual test specifications until a predetermined number of failures or hours of operation had occurred. In the event that no failures had occurred at the end of the test period, the environment stress was, in some cases, increased until failure occurred. A failure analysis was conducted after each failure.

FMT was conducted by increasing the selected failure producing stress, vibration and/or temperature, (and in two cases an Electro-Interference Test) until an out-of-tolerance condition was reached. The stress level was then backed off to determine if the component returned to within specification requirements. Failure to return was considered a catastrophic failure. Once a failure mode of a component had been established, any additional testing on that component was directed toward other modes of failure. No attempt was made to determine any statistical significance of a failure mode on either a time base or a margin of safety base.

The reliability test program gave some additional confidence that the selected components were mature enough to be used on a man-rated program, although each component had previously passed a very extensive qualification test program, and most of the failure modes were found then. One significant failure occurred that affected the entire program. The static inverter failed in vibration in the second axis tested, and this failure occurred at qualification level of test (a test that qualification units had passed successfully). Failure analysis showed that the vendor had made an unauthorized change in the internal wiring on the production line (the wires were not supported as they had been), and they failed under vibration. Because of this failure, the vendor and all others were warned that if any changes were to be made to the hardware after it had passed qualification testing, requalifying would be necessary. This policy was maintained throughout the program with few known exceptions. In two different cases, changes were made to components without notifying the contractor. In one case, two subvendors made "cost saving" improvement changes to piece parts used in a time delay relay without notifying the vendor, and both of the changed parts failed during later testing. In the other case, a vendor made changes to a spring in a connector without notifying the contractor, and the connector failed to disconnect properly.

It is believed that a more efficient means of determining reliability by test would be to extend the qualification test program on selected items (and selected environments), in order to actually determine the margin of safety on these items. The qualification requirement that no failures are accepted means that after completing testing, the basic integrity of the component is still unknown (qualification testing is a test to <u>pass</u> not a test to <u>failure</u>). All testing is expensive, therefore, the maximum amount of information should be obtained from every piece of hardware designated for component testing.

16. AIRBORNE SYSTEMS FUNCTIONAL TEST SET-UP (ASFTS)[28]

a. Background

The Gemini ASFTS program was initiated in January 1962 at Martin Baltimore to functionally test the airborne system of the Gemini Launch Vehicle. While particular emphasis was given to the greatly modified flight control and the all new malfunction detection systems, ASFTS was a functional bench layout of the total vehicle. The defined system configuration at the initiation of the program was non-redundant with the malfunction detection system being fully automatic. During the period January through May of 1962, engineering, procurement, documentation and manufacturing installation were completed in preparation for the test program scheduled to start June 1, 1962. At this time, direction was received from the procuring agency to redesign the airborne system to a redundant configuration with the malfunction detection system to become basically a manual system. During the several months required for system redesign and procurement on the redundant configuration, a limited test program was conducted on the completed non-redundant configuration.

After redesign and modification, the major test effort took place between November 1962 and August 1963. ASFTS continued to serve the Gemini program for modification (ECP) design proofing, failure analyses and other investigations as listed in paragraph (c) below.

b. Description

ASFTS can be described as a functional block diagram of the Gemini Launch Vehicle and associated AGE. The various subsystems were bench, rack, platform and board-mounted. This allowed complete accessibility of test points to facilitate engineering development and failure mode analysis. Its subsystem could be either functionally isolated or integrated with other subsystems.

The ASFTS facility was located on the ground floor of the Baltimore Vertical Test Facility in a laboratory environment complete with AGE, test equipment, and simulation equipment.

c. Use

ASFTS was used for the following purposes:

1) Progressive component, subsystem, system and integrated systems development and design clean-up.
2) AGE/Airborne system development and design clean-up.
3) Engineering change evaluation.
4) Spacecraft interface confirmation.
5) Systems logic - redundancy and MDS statusing.
6) Parametric variation effects.
7) Procedures development.
8) Flight control stability - utilizing an analog computer for airframe in-flight simulation.
9) Failure mode investigations in support of test and flight anomalies and failures.

d. Advantages of ASFTS

When compared to a complete vehicle functional mockup as used in many programs, the ASFTS approach represented two major advantages, (1) cost and (2) test accessibility.

A cost comparison study performed during the program showed conservatively that a full mockup would have cost over twice as much as the cost of ASFTS.

The laboratory-type environment with bench and board layout provided a far handier and accessible situation for engineering design testing and problem solving than a full mockup.

In addition to its use throughout the program for evaluating design changes, it was also invaluable in trouble-shooting vehicle problems encountered at ETR. In several cases including GLV-12, hardware was sent to ASFTS and successful duplication of malfunctions there pin-pointed the cause of the problem.

The flexibility and quick reaction capability associated with the use of ASFTS in support of the flight phase was of untold value to the success of the program. In any list of items to be recommended for consideration by other program efforts, an ASFTS-type of approach would certainly be at the top of the list.

17. QUALIFICATION TEST PROGRAM

a. Component Control and Definition of Qualification Testing

The evolution and control of a component (black box, actuator, etc.) is quite complex, as shown in Figure II.C-36. The qualification test portion of this evolution is equally complex and should encompass more than environmental testing. In order to reduce the chance of failure during qualification testing, it is advantageous to perform engineering development tests early in the program. In addition, a production monitoring test is recommended to uncover latent fabrication defects and preclude such failures during qualification. The definition of these tests is as follows:

1) Development tests are environmental tests to certify the design concept and to determine optimum compatibility with design criteria. Independent of test specification requirements are the specified tests conducted during the design phase to develop hardware, evaluate operating limits, establish safety margins over and above design levels and aid in selecting the best of several candidate components. <u>If the development tests are properly selected and thorough and the production hardware does not change significantly, the qualification test program becomes a mere formality.</u>
2) Production Monitoring Testing (PMT) is discussed in a separate section (II.C-18). It is defined as acceptance testing of production hardware in environments of sufficient severity to uncover latent defects, but not high enough to cause any significant damage to the component. Latent defects are those which are not visible or apparent with normal inspection techniques. A reduced level of vibration relative to qualification level was used on each selected component in the Gemini Program. Production components normally went through PMT prior to operational usage. This philosophy was employed for the qualification test program (components undergoing qualification testing had prior usage comparable to the flight article).
3) Qualification tests are a series of functional tests performed under simulated environments, which have a factor of safety above mission requirements. The test article used is identical with the production equipment. The purpose of these tests was to demonstrate, prior to operational usage, that the equipment would function satisfactorily in the anticipated environment.

Using the Titan I and Titan II environmental criteria as a base, qualification tests were conducted on components which were new, which underwent a change in design or usage, or which experienced a more severe GLV environment.

The test program required rigid controls and precise documentation. All failures were recorded, and Quality Control assured compliance to approved specifications, so that qualification testing became a formal program.

Failures played a major role in qualification testing. They were classified as out-of-tolerance measurements, changes in configuration, ruptures, internal rearrangement and corrosion. Each failure was examined and corrected individually. Here, important decisions were made relative to retest direction. If a major failure occurred and the fix required new or additional parts, the component was restored to new hardware status and the entire test cycle rerun. This has a serious effect on any schedule. It normally takes six weeks to conduct the required environmental qualification tests; failures may prolong this to six months. In this event, the vendor's evaluation test program would seem to be weak or nonexistent. Minor failures <u>may</u> only require rerunning the environment, which may effect the fix (i.e., usually vibration). In general, no failure was permitted to pass without some retesting. The test hardware used was selected from the initial production units. Vendor "hand-built" qualification units may pass the required tests, but the production hardware may not be able to pass the acceptance tests, much less qualification test levels.

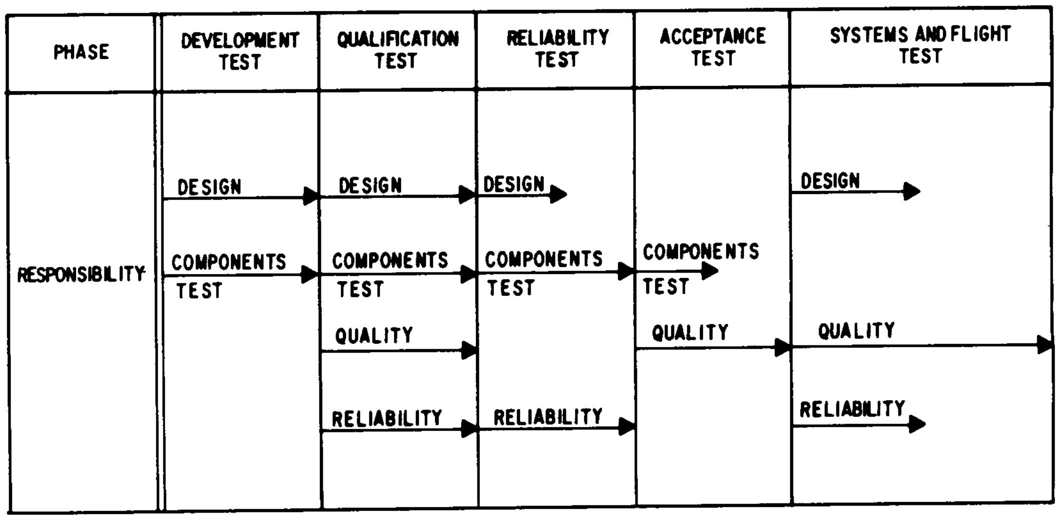


Figure II. C-36. Component Evolution and Control

The environmental test requirements for each component was specified in the contract. All test results in the form of formal reports were submitted to SSD/Aerospace for review and approval.

The test cycle presented many unique problems and unknowns. It was necessary to review all these problems with the proper supervisory people, in order to keep them up to date with the program. The vendors were made to feel that technical assistance was available from the contractor. Personnel at the directors' level visited vendors' plants to assist them during the peak of the qualification test program. The inhouse qualification testing was easier to control; however, the testing was conducted to the same rigid standards as imposed on the vendors. The Martin Company qualified fifty four (54) Gemini peculiar components (plus forty-eight (48) components previously qualified on Titan II to the same or more stringent requirements), Aerojet General qualified twelve (12) components, General Electric qualified eight (8) components, and Advanced Communications Inc. qualified one (1) component. It should be noted that three (3) (in most instances) units of each component were required to pass all of the tests without failure, and that all components were qualified prior to the first manned launch.

b. Conclusions

The lessons learned from the Gemini Qualification Test Program were many and should be invaluable to a new program. The significant areas are described below.

It is important to separate the evaluation/development testing from formal qualification testing. The evaluation tests should start with prototype hardware, as soon as the level of the critical environments are established.

Failures are an integral part of the test program but no qualification test schedule has ever been permitted a "failure pad", because this would be admitting that an inadequate evaluation test program has been conducted, or the design group had serious doubts as to the integrity of the components being tested. When a failure occurs during qualification testing, perserverance and technical judgment are the only tools available. The vendor is in a position to lose money, because his production line does not move and the hardware usually undergoes changes. In some cases, the vendor's technical capability becomes stretched to the point where he does nothing. He may refuse assistance, citing proprietary rights. A failure usually results in some retesting and may require reverting to the start of the test program. In some cases, it is desirable to continue testing when an out-of-tolerance condition occurs, in order to subject that part of the hardware to other tests that may have an effect on the part. It is then possible that one fix could correct several defects at the same time and gain valuable test time. Vibration was the cause of 30% of the qualification test failures, and this was probably to be expected, but electro-interference, seal, temperature, and salt fog accounted for 8-10% each (see Figure II.C-37) and this appears to be high for nondynamic environmental type failures. The designers must be made aware of the environments and levels early. Failures due to most environments are inexcusable and are rare, if the designer is given all of the facts regarding the component and its intended usage at the start of his design effort. No flight failures were recorded on GLV hardware throughout the program, but one significant failure occurred after engine ignition that caused a shutdown on the pad. This failure (a tandem actuator) probably could have been prevented by a better design review or a better, more extensive, qualification program. No dynamic impulse test, probably the only test that could have found

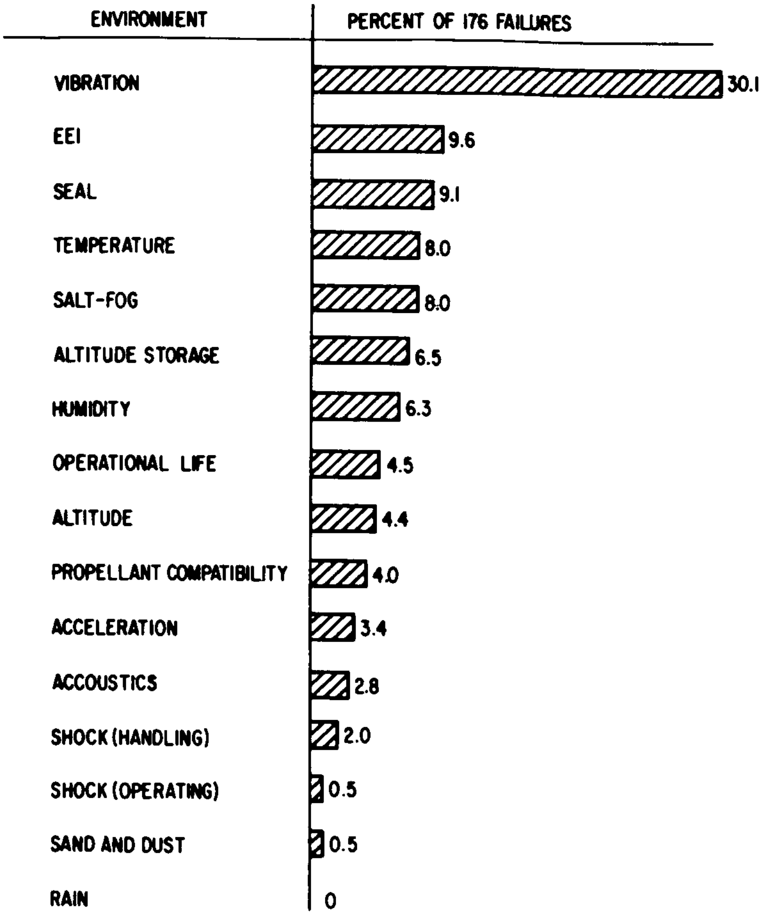


Figure II. C-37. Qualification Test Failures by Environment the problem, was included nor even contemplated until after the failure. It is impossible to put too much time into the study of all of the environmental effects on a component during its design phase.

It is extremely important that any contract with a vendor that includes qualification testing should specify that the units used for qualification must be identical to production hardware. The vendor should select approved materials and parts at random and avoid hand selection and fitting. It is possible for a unit to "pass" qualification and yet "similar" units fail to pass acceptance testing, because the vendor has made changes or not exercised the same control with the production units. A well defined contract can save many future headaches on both sides. It is just as important to contractually control the vendor's production hardware to assure no changes are made without customer knowledge and concurrence.

The environmental criteria evolved from Titan I and Titan II data. The levels were deliberately conservative for each environment, but only the levels specified for vibration were of such magnitude as to be controversial, and this was because the levels were specified for the various compartments, rather than for the individual components. During captive firings and flights of the Titan I vehicles, numerous accelerometer measurements were taken and reduced for the response spectrums. A review of several hundred spectrums showed a great variation in shape and amplitude. Even when sorted by compartments, the spectrums still showed variations between locations and between firings for the same location. General practicality and technical judgment then dictated an overall specification for each compartment, rather than an individual spectrum for each piece of equipment. Titan II levels of vibration were predicted before first flight by extrapolating the Titan I levels and using techniques that provided for the inclusion of structural effects, as well as the influential vibration sources.

The amount of conservatism obtained by this method of developing the criteria depends on what company or agency made the comparison between flight data and test criteria. A BB & N report [29] states that the "---- existing Titan II specifications are quite representative of the missile vibration measurements ---." Martin and G. E. have stated that they believe the environments for the individual components are conservative, varying from "slightly" in some areas to "several hundred percent" in other areas.[30]

The deciding factor in using the identical vibration criteria for GLV that was used on Titan II was that it appeared inconceivable to test a GLV component to a level different (lower) than a T-II component was tested and yet have it located adjacent to the T-II qualified unit in the vehicle.

18. PRODUCTION MONITORING/ACCEPTANCE TESTING

Production Monitoring Testing (PMT)[31] is the application of an environment (usually vibration) to each component of a production run as part of the component acceptance technique. The environmental level chosen was sufficiently severe to uncover latent defects, but not high enough to accumulate significant fatigue or other damage to the parts. Latent defects are those hidden from conventional inspection techniques and may be divided into two general types:

1) Gross defects - cold soldered joints are a classic example of this type of defect.
2) Defects which are a function of environment - this type is more difficult to define and find, but an example might be the mechanical interference of electrical parts resulting in a short.

Corollary benefits of PMT during the initial production runs included the identification of defects undetected by immature quality control processes, and identification of design problems not uncovered during the development and qualification processes. Later in the program, PMT aided in the discovery of unauthorized changes to components by vendors.

Acceptance testing that has included vibration has been looked at with jaundice eye by some members of the industry. It is felt that it is possible to stress components to the point that they may fail the next time they are vibrated - during flight. A great deal of study went into the levels of vibration used on the Gemini components during PMT and evolved from the Titan I and Titan II programs.[31] It is worth noting that no component failures were recorded during flight during the entire Gemini Program, and over 700 components received PMT.

Figure II.C-38 shows the PMT failures by type of malfunction.

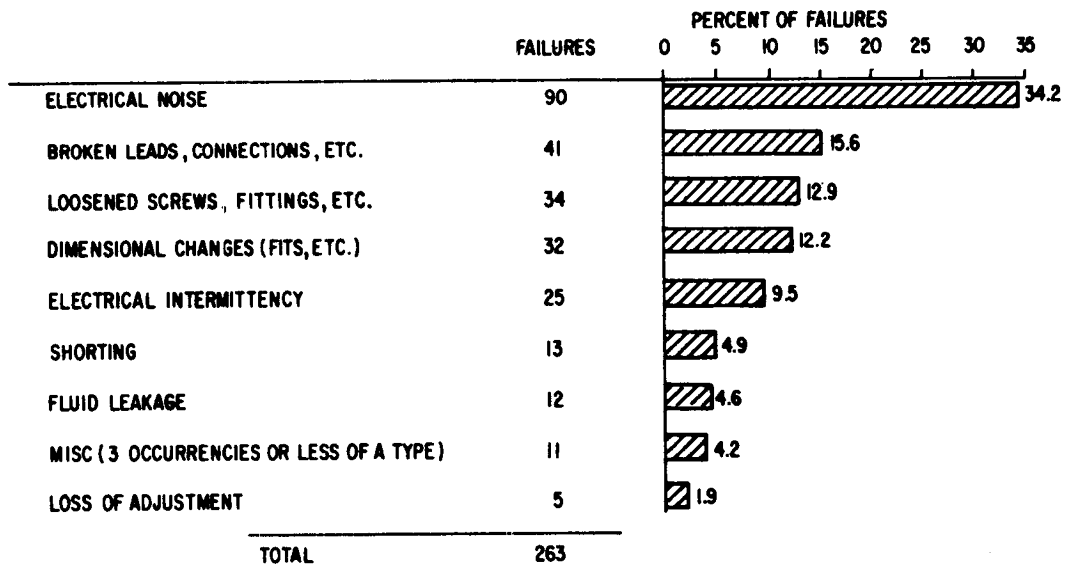


Figure II. C-38. PMT Failures by Type of Malfunction

19. PROPELLANT TRANSFER AND PRESSURIZATION SYSTEM (PTPS)

a. Background

Selection of the method and equipment for performing propellant loading to meet the requirements of the Gemini Launch Vehicle test and launch operations was made after detailed evaluation of four possible methods. These were:

1) Baldwin-Lima-Hamilton Digital Weighting System
2) Launch Vehicle Tank High Level Point Sensors
3) Calibrated Storage Vessels
4) Positive Displacement Flowmeters

Method 4 was selected as the primary means of measuring loads. This system was capable of indicating the quantity of propellant loaded in each GLV tank to an accuracy of $\pm$ 0.35% by weight. The high level sensors and BLH Digital Weighing System were used as secondary checking systems. The high level point sensors were installed and calibrated in each GLV tank at VTF.

The propellants utilized were

Fuel	Blend of liquid chemicals, hydrazine (N_2H_4) and unsymmetrical dimethylhydrazine, $\{N_2H_2(CH_3)_2\}$
Oxidizer	Nitrogen tetroxide (N_2O_4).

b. Description

Schematics of the fuel and oxidizer PTPS's are shown on Figure II. C-39 and Figure II. C-40. These show the systems in the final configuration beginning with GLV-6 and used thereafter.

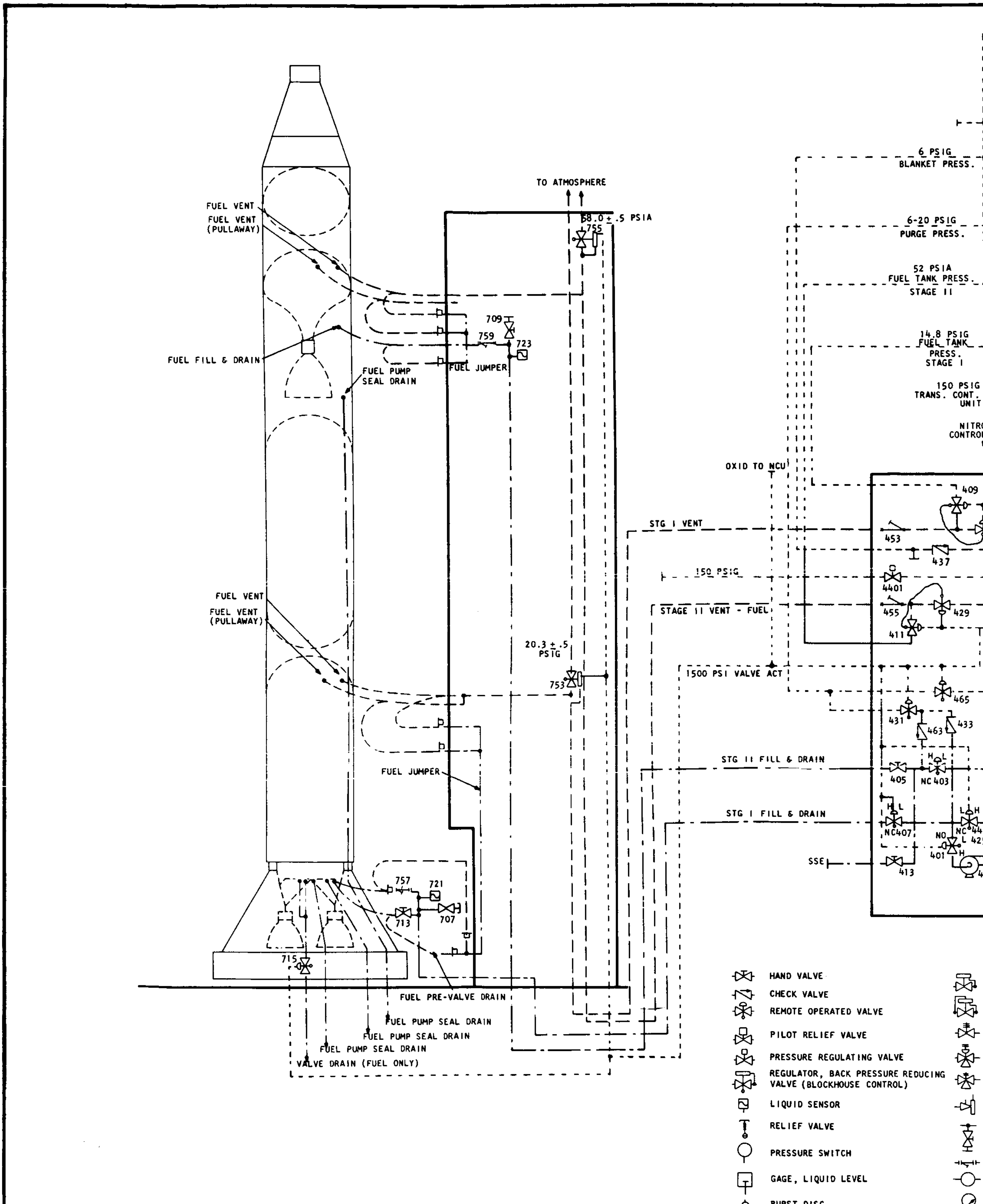
c. Significant Modifications

(1) Tandem Flowmeters (ECP 472)

Redundant propellant system tandem flowmeters were incorporated prior to the GT-6 mission to provide increased confidence in loading accuracy and accommodate "in line" spares. With this configuration, a single loading was used to verify each tank high level sensor location. This change placed an additional flowmeter in tandem with each of the existing flowmeters and piping modifications were made to permit flowing all four flowmeters (Stage I and Stage II) in the forward direction during prechill, to allow a check of the calibration of all four meters.

(2) Redundant Propellant Holding Area (ECP 440)

Redundant capability of the PTPS was provided to improve reliability and recycle time to support rendezvous launch windows and to prevent a launch postponement due to failure of a single system. This effort required the modification of the original propellant transfer and pressurization system to accept redundant Ready Storage Vessel (RSV), Temperature Control Unit (TCU), and heat exchange circulating units. Components and equipment were provided from Complex 16 and were modified and refurbished to suit the Complex 19 requirements. Interconnecting piping was incorporated to provide redundant crossover from either RSV to either TCV by any of four (4) pumps. This was accomplished prior to the launch of GT-6.



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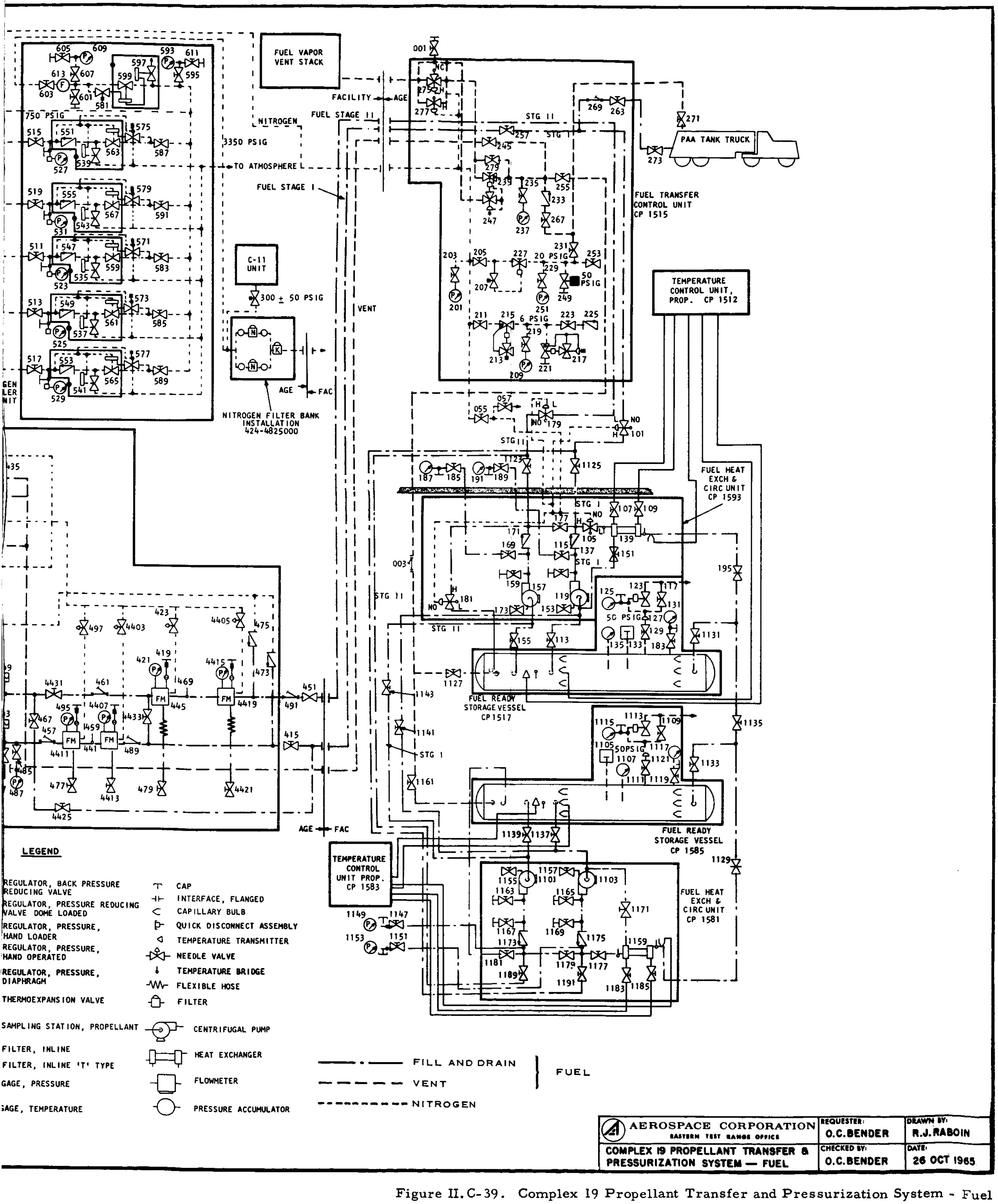
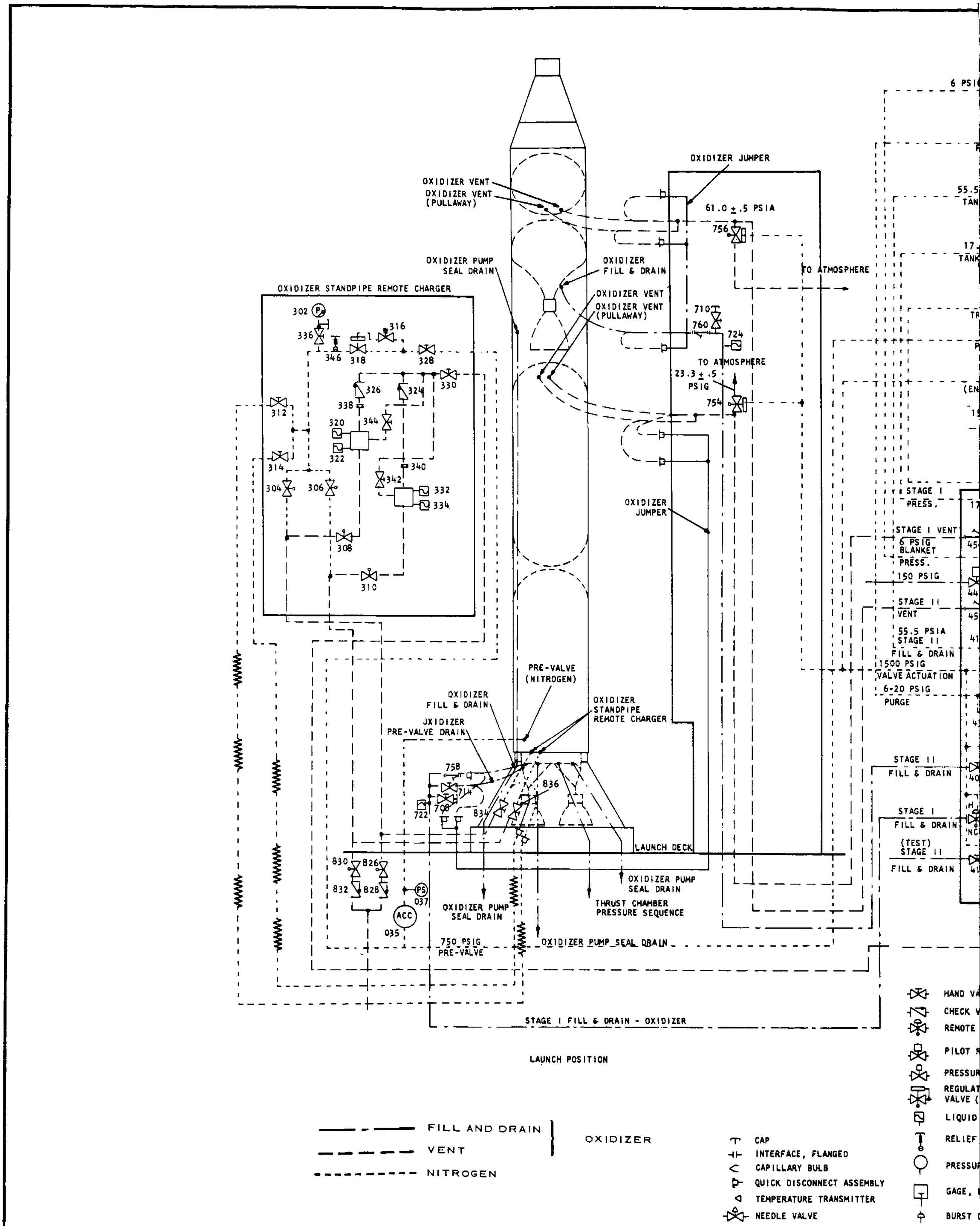


Figure II.C-39. Complex 19 Propellant Transfer and Pressurization System - Fuel



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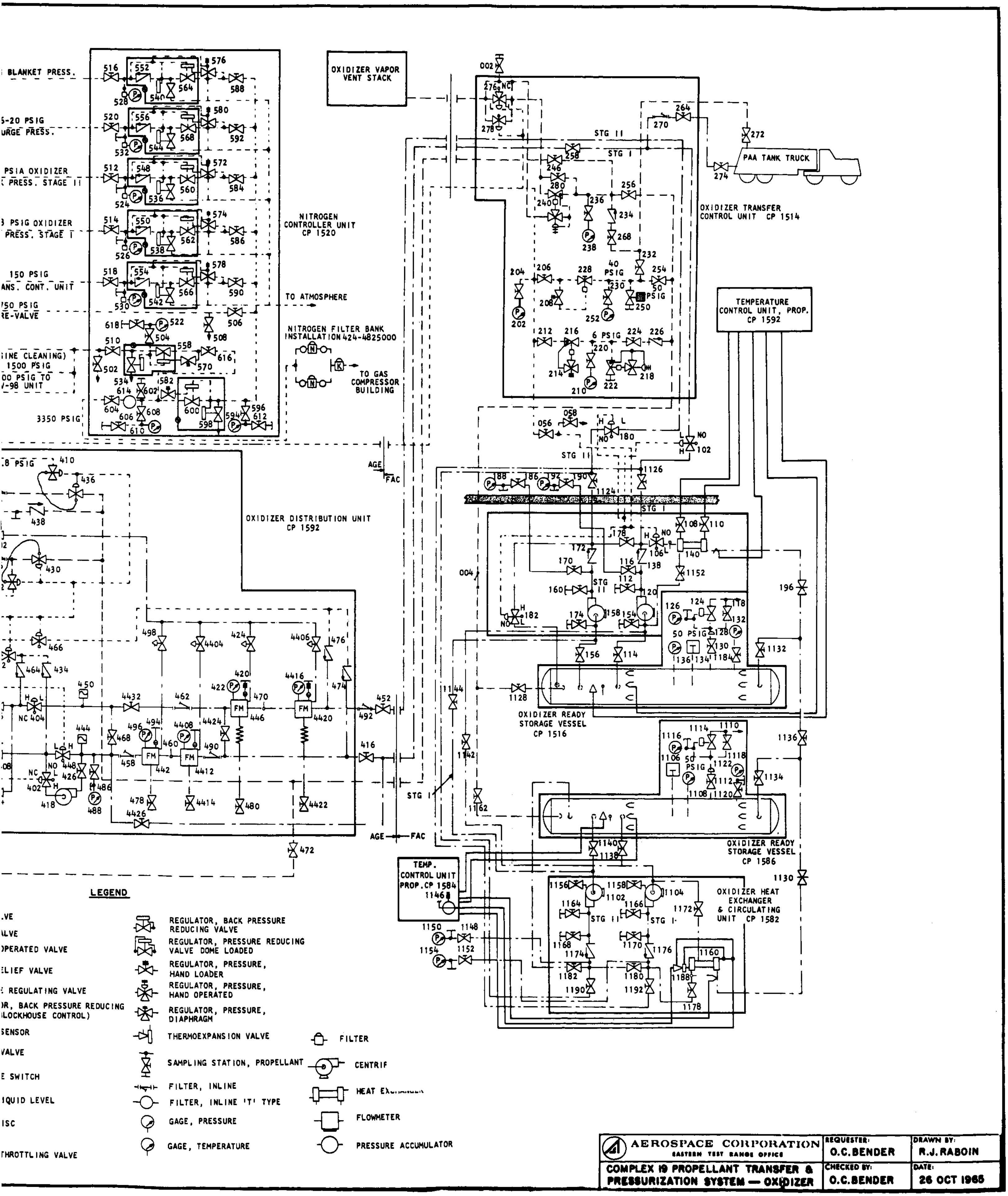

Figure II.C-40. Complex 19 Propellant Transfer and Pressurization System - Oxidizer

(3) Flowmeter Modifications

During the early phase of the program two significant problems were experienced with the flowmeters. They were Automatic Temperature Compensation (ATC) and leakage at the counterbase plate seal assembly.

ECP 543 was implemented to improve the reliability of the flowmeters and loading accuracy. This effort involved the removing of the ATC and gear changer, adding a stack extension and redesigning the counterbase plate seal assembly.

The stack extension was designed to support the shroud assembly and to permit purging over the counterbase assembly. A new dynamic type seal was incorporated for the counterbase plate shaft takeoff. This modification also changed the read-out of the flowmeter from pounds to gallons of propellant. Loading procedures were revised.

To support this significant change, extensive evaluation and life tests were conducted in laboratory and flow facilities. Modified flowmeters for Pad 19 were incorporated to support the GT-8 and subsequent operations.

SECTION II-C

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SECTION II-C

DEFINITIONS

AFBSD	Air Force Ballistic Systems Division
AGC	Aerojet General Corporation
AGE	Aerospace Ground Equipment
APS	Accessory Power System
ASCO	Auxiliary Sustainer Cut-off
ASFTS	Airborne Systems Test Stand
ATC	Automatic Temperature Compensator
AVCO	Advanced Communications Corporation
BECO	Booster Engine Cutoff
Cap Com	Capsule Communicator
cg	Center of Gravity
CSAT	Combined Systems Acceptance Test
d-c	Direct Current
ECP	Engineering Change Proposal
ELT	Environmental Life Testing
ETR	Eastern Test Range
FM	Frequency Modulation
FMT	Failure Mode Testing
FOD	Flight Operation Division
fps	Feet per Second
G & C	Guidance and Control
GE	General Electric
GEMSIP	Gemini Stability Improvement Program
GLV	Gemini Launch Vehicle
GT	Gemini Titan
ICBM	Intercontinental Ballistic Missile
IGS	Inertial Guidance System
IPS	Instrumentation Power System
LO	Liftoff
M/B	Martin Baltimore
MCC	Mission Control Center (NASA) Houston
MDFJPS	Malfunction Detection Fuel Pressure Switch
MDP	Malfunction Detection Package
MDS	Malfunction Detection System
MDTCPS	Malfunction Detection Thrust Chamber Pressure Switch
MISTRAM	Missile Tracking and Measurement System
MOCS	Master Operations Control System
NASA	National Aeronautics and Space Administration
NPSH	Net Positive Suction Head
OAMS	Orbital Attitude and Maneuvering System

OPPS	Oxidizer Pressurant Pressure Switch
PCM	Pulse Code Modulation
PMDS	Prelaunch Malfunction Detection System
PMT	Production Monitoring Test
POGO	Vehicle Longitudinal Instability Problem
POPOI	Pressure Oxidizer Pressurant Orifice Inlet
PSTP	Propulsion System Test Program
PSV	Pressure Sequencing Valve
PTPS	Propellant Transfer and Pressurization System
R & D	Research and Development
RESS	Redundant Engine Shutdown System
r-f	Radio Frequency
RGS	Radio Guidance System
RSO	Range Safety Officer
RSP	Rate Switch Package
RSV	Ready Storage Vessel
S/A 1, S/A 2	Subassembly 1, 2
SECO	Sustainer Engine Cutoff
SMRD	Spin Motor Rate Detection
SSD	Space Systems Division, United States Air Force
STO	Systems Test Objectives
TARS	Three Axis Reference System
TCA	Temperature Control Amplifier
TCPS	Thrust Chamber Pressure Switches
TCU	Temperature Control Unit
TCV	Temperature Control Valve, Thrust Chamber Valve
TCVPSV	Thrust Chamber Valve Pressure Sequencing Valve
TPS	Telemetry Power Switch
UDMH	Unsymmetrical Dimethylhydrazine
$(V - \Gamma)$	V - Velocity, Γ - inertial flight path angle
VTF	Vertical Test Facility
W_{PN}	RGS Yaw Steering Command

D. GUIDANCE EQUATIONS AND PERFORMANCE ANALYSIS

1. GUIDANCE EQUATIONS AND GUIDANCE PROGRAM

a. Design and Implementation

Early in the Gemini Program it was decided by the Air Force and NASA that the guidance equations would be formulated by the Aerospace Corporation. Members of the Electronics Division of Aerospace designed the equations from which the Burroughs Corporation coded the guidance program to be wired into the trays of the Burroughs A-1 computer at the Mod III radar site. The computer program, before wiring, was formally validated by Aerospace. Before each mission, the Gemini Launch Systems Directorate certified to the Air Force that the guidance equations and the guidance program were ready for manned flight. For all launches that involved equation changes requiring computer rewiring, the equations verification and program validation were repeated by Aerospace.

The Gemini guidance equations in essence evolved from the Mercury equations, which were originally developed for the early Atlas Weapon System. A set of explicit radar guidance equations was designed to insert the payload into a free flight orbit at a specified altitude, velocity, and flight path angle. For the fixed launch azimuth or for the non-rendezvous case, the orbit plane was such that the capsule would pass over a pre-selected point on the surface of the earth. The Gemini Program was the first to require a variable launch azimuth capability that would maneuver the GLV parallel to the target plane.

Many Gemini-unique requirements were placed upon the GLV guidance equations, but before these are considered, the improvements encompassed in the portions of the equations kindred to its predecessors must be mentioned. The primary functions of earlier guidance equations were to compute pitch and yaw steering commands to achieve the proper altitude and flight path angle at insertion, and to send the cutoff signal (SECO) when the required velocity was reached. A block diagram displaying the radio guidance system with the radar signal inputs into the computer and the steering commands and cutoff signal issuing from the computer is shown in Figure II. D-1. The Gemini equations, performing these functions, incorporated more extensive and elaborate filtering than had been used previously. In particular, the computation of the altitude turning rate, the predicted attitude at cutoff, and the initial pitch and yaw velocities were all filtered. In other words, the components which formed the attitude errors as well as the errors themselves were smoothed. Unlike its predecessor, the time-to-go equation for Gemini employed distinct velocity and rate of velocity filters. The filter time constants were designed as a compromise among response to random radar noise, systematic radar biases, and vehicle perturbations. The radar model used was based on information provided by GE in Memo 57, dated 1 January 1963, and the vehicle characteristics were supplied by Martin.

For guidance initiate, which occurred at a predetermined time after liftoff, a velocity gate was designed to detect large discrepancies from nominal due to perturbed flights. This "glitch" test discriminated between large bias errors and random noise. Using radar data preedited by the equations and identified by quality flags, this test could provide for data rejection and/or data replacement.

An innovation of the Gemini equations was their ability to target in real-time. The achieved orbit of the target vehicle was defined to the computer by ephemeris data at twenty minutes before GLV liftoff. Upon receipt of the T-3 minutes signal from the blockhouse, the equations computed a roll program based on an analytical scheme that biased the launch azimuth to reduce the error between present position and the target at insertion. During the first stage, the velocity vector determined by this variable launch azimuth was directed into the target plane. During second stage the steering technique maneuvered the yaw velocity vector to be parallel to the target plane at SECO. Yaw position steering was not

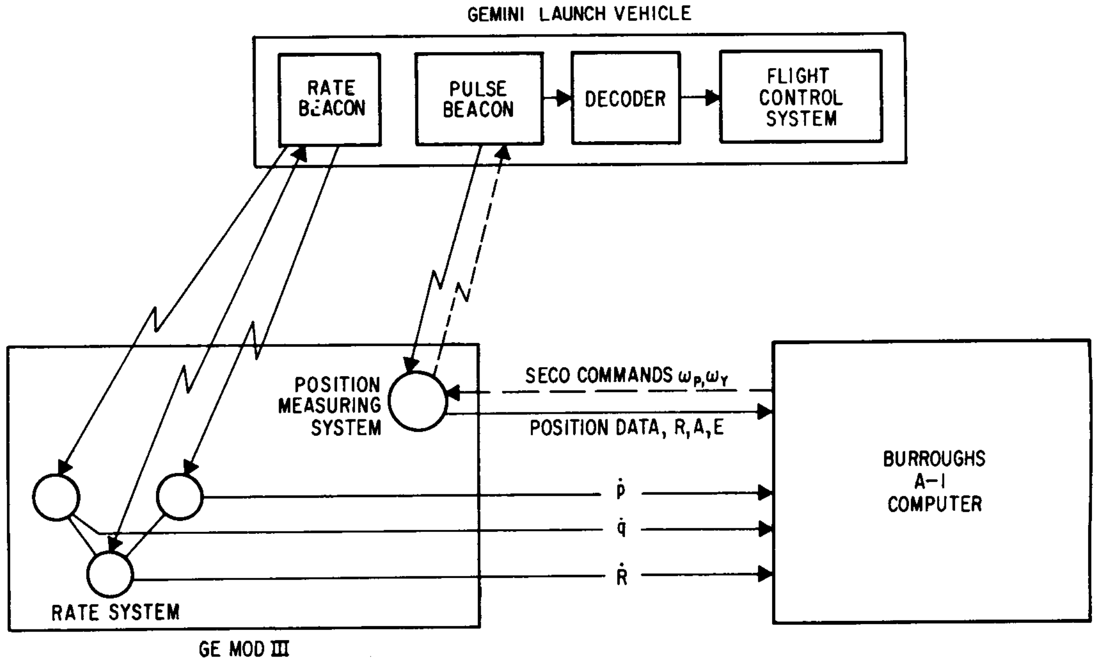


Figure II. D-1. GLV Guidance System

used on Gemini. The objective was to null the yaw velocity error at insertion. Additional targeting options were introduced into the equations at later dates in the Gemini program; these are described in Section II. D-1-b. Range Safety constraints limited the span of allowable launch azimuth, and a manual constant in the computer was designed to limit the maximum amount of wedge angle removal for payload considerations.

Another unique feature in the Gemini equations was its computation of parameters to be used by the backup inertial guidance system (IGS) in the spacecraft. Four targeting quantities were transmitted to the IGS at T-3 minutes. At 100 and 140 seconds, an updating value of $\dot{Z}$ [velocity in the inertial IGS coordinate system analogous to Vy in the radio guidance system (RGS)] was transmitted to the IGS. This updating was used to correct platform misalignment. The RGS value transmitted at 100 sec had a 3-sigma accuracy of 2.19 fps and the 140 sec correction was accurate to 3.03 fps.

The importance of precision timing in a rendezvous launch is obvious. The nominal time origin for the equations was the GLV liftoff pulse. However, if the computer failed to receive the liftoff pulse, a scrub could be avoided by a provision in the equations to use an elapsed time commencing from the platform release signal, and, in the event of that failure, a prediction could be made from the T-3 minutes signal.

As sections or "figures" of the guidance equations were completed, they were sent to the Burroughs Corporation for programming. Each figure contained a function of the equations, such as Targeting, Steering Commands, Cutoff, etc. Upon receipt of the equations from Aerospace, the Burroughs Gemini Project Engineer assigned a staff of programmers to begin coding for the Mod III guidance computer. The program was broken into subassemblies corresponding to the figures of the equations. A direct language program of the guidance equations to be used on a Burroughs general purpose computer was also written for the purpose of comparison with the interpretive guidance program. When all errors were traced and removed by this comparative method, the total assembled program was transmitted to Aerospace for validation. Figure II. D-2 illustrates the order of Burroughs programming procedures.

In addition to the implementation of the guidance program, an answer package was generated consisting of input simulation tape, master intermediate tape, master output tape, and plots. These, too, were validated by Aerospace.

Upon program acceptance by Aerospace, computer trays were wired in Paoli under strict quality control supervision. Two sets of trays, for redundancy, were then shipped to ETR where they were checked out using the answer package and countdown routines.

For launch-to-launch program revisions, rewiring was accomplished at ETR from wiring list instructions sent from Paoli, based on Aerospace equations changes. Each revision required a new answer package.

The special features of the Gemini guidance equations mentioned above imply many interfaces. The Burroughs computer utilized seven paths for receipt or transmission of data. A block diagram of computer interfaces and information flow is shown in Figure II. D-3. The ancillary functions provided by the equations and computer were the following:

1. Automatic receipt and verification of target ephemeris data from MCC-Houston.
2. Synchronization to Greenwich mean time at T-6 minutes.
3. Synchronization to countdown time at T-3 minutes.
4. Transmission of the roll angle to the blockhouse and receipt of verification.
5. Transmission of the targeting data to the IGS buffer and receipt of verification.
6. Receipt of platform release and liftoff signals from the blockhouse.
7. Remoting to MCC-Houston of real-time position and velocity data for plotboard displays.

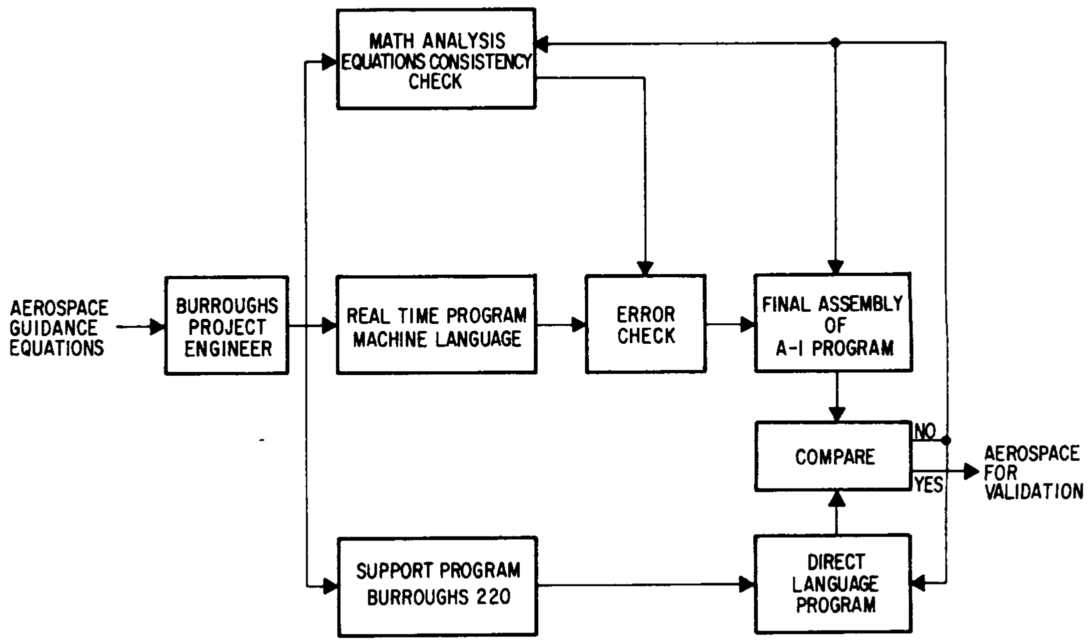


Figure II. D-2. Implementation of Guidance Program

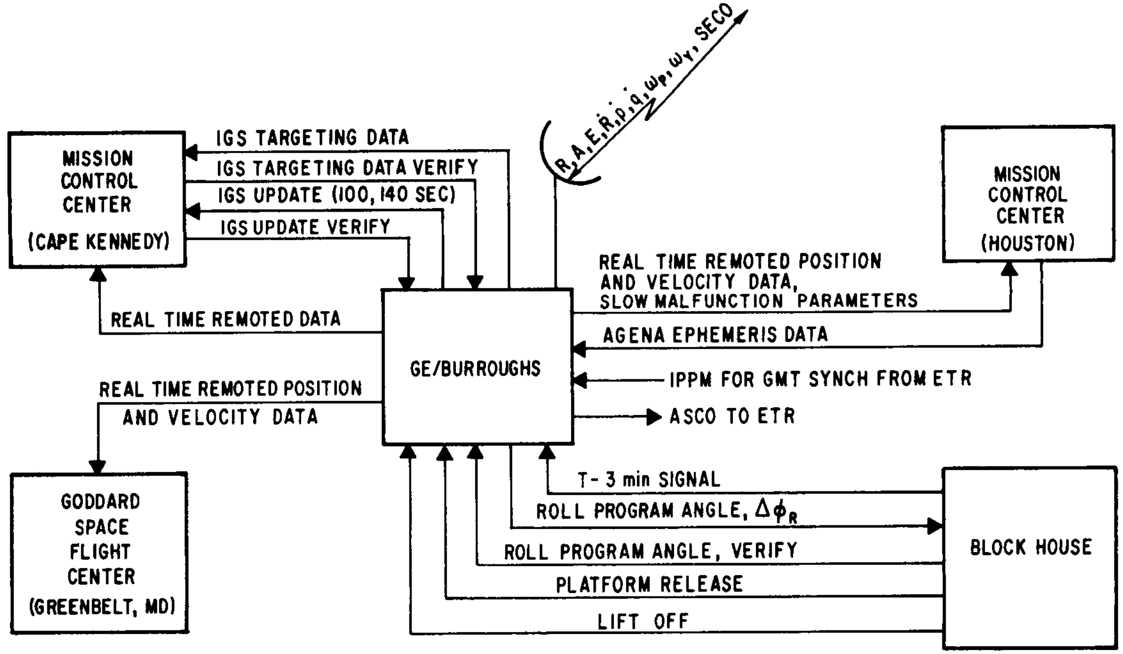


Figure II. D-3. RGS Computer Interfaces

8. Remoting guidance parameters to MCC-Houston.
9. Remoting guidance parameters to Goddard Space Flight Center (GSFC), Maryland.
10. Transmission of the IGS updates to the IGS buffer at 100 and 140 seconds and receipt verification.

The Burroughs A-1 computer had been designed for Atlas Weapons Systems guidance and, by use of the Pilot Safety Program, was man-rated for use on Mercury. Some of the ancillary functions listed above required the design of special equipment to augment the capabilities of the existing computer. By the time the Gemini-unique equipment was required, the state of the art had advanced to a point where the Data Exchange Unit (DEU) and Test Conductors' Console had been manufactured with components of a reliability exceeding that of the computer main frame. Design experience had progressed allowing the specifications and acceptance tests to be more stringent in the Gemini-unique equipment than on the A-1 main frame.

Design ground rules for the new equipment required that interface lines between the existing computer and the DEU be isolated to inhibit malfunction feedback from the DEU. No programming of any existing guidance programs or interpretive routines and no rewriting of existing field procedures or countdowns were required.

At the time the original guidance program was completed by Burroughs, the phase of program validation by Aerospace began. The card decks were assembled and a listing of the program was made. A step-by-step check ascertained that the program was equivalent to the statements of the guidance equations. An interpretive routine run made on the 7094 computer at Aerospace caused the IBM computer to perform exactly like the Burroughs guidance computer at the launch site. Each program step for each computer cycle was then calculated on a manual desk computer and the results compared with the interpretive run print-out. All decision logic paths were checked in the same manner. It was mandatory for validation that the coding produced results within the accuracy quoted in the equations specification and that the program had correctly coded the equations intent.

The second half of the program validation was then initiated. This effort was integral with the equations verification, in that the program was proven operative in the complete guidance system. It was demonstrated that the equations and program correctly performed their task of fulfilling mission requirements. Many interpretive runs with various radar and vehicle perturbations were reviewed. The quantization, scaling, countdown errors, and inherent delays in the guidance computer were all scrutinized to prove that computer time restrictions were not violated and that the equations were not degraded.

In addition to the varied interpretive runs, the equations verification included examination of trajectories made with a scientific simulation. These nominal and anomalous cases were used to ascertain the range and extent to which the equations would produce their intended results under different conditions.

The entire validation procedure followed documentation in a formal test plan which specified a series of trajectories in which the RGS equations and the Burroughs program were tested to establish a level of confidence in their adequacy. The series of simulations with normal and marginal performance in the vehicle subsystems and radar demonstrated that the guidance equations/program would operate satisfactorily with normal vehicle operation and that they would not in themselves cause a mission failure. A report was prepared by the Electronics Division of Aerospace summarizing the results of the tests, listing the values of pertinent quantities for each case, and including computer listings and plots.

After completion of the validation, the equations and program were certified to be flight ready.

As a time and cost saving measure, many of the computer scientific (non-interpretive) simulations that were run for the guidance equations validation were utilized in the guidance accuracy study. The purpose of the study was to determine an estimate of the rms errors of relevant parameters at the defined insertion point (SECO +20 sec).

The effects of vehicle perturbations, radar random errors, and radar systematic errors were computed separately and then root-sum-squared (rss) to obtain the overall effect. A typical summary of results is shown in Table II. D-1

To determine the effects of radar random errors only, a sample of 40 runs subjected to a random noise generator was taken, and the results root-mean-squared to provide the one sigma error on velocity, flight path angle, etc. In order to evaluate the error attributable to systematic radar biases, the one sigma bias error was added to each of the radar input quantities separately, R, A, E, $\dot{R}$, $\dot{p}$, $\dot{q}$, then root-sum-squared for the effect on each insertion parameter. The error caused by various vehicle perturbations on each insertion variable was evaluated by individual simulations. The final rss combination, then, was as described above.

Table II. D-2 shows the predicted insertion accuracies for flights GT-1 through GT-12. The predictions were altered three times in the Gemini Program, first because a drift compensator incorporated into the equations for GT-2 greatly improved yaw velocity accuracies, and later because new knowledge was acquired about the Mod III radar through flight test results. The changes are discussed in more detail in Section II. D-1-b. Table II. D-2 shows the close correlation between the actual insertion errors throughout the Gemini Program and those predicted.

b. <u>Guidance Equations Flight History</u>

The guidance system performed satisfactorily on all Gemini launches with insertion results always within 3-sigma predictions. The four insertion parameters achieved on all flights are compared with their estimates in Table II. D-2. Modifications of the equations for flights GT-2 through GT-12 are summarized on Table II. D-3 and described in more detail in the text that follows.

Vehicle center of gravity shifts and thrust misalignment that had not been simulated in the dynamics model were apparent in the GT-1 flight results. The relatively noiseless flight showed that the equations responded sluggishly to the effective drift. The insertion errors in Table II. D-2, although less than the predicted 3 sigma values, indicated a need to eliminate the drift problem in order to improve insertion accuracies. Since time constraints precluded hardware redesign, the vehicle simulation was made more realistic and compatible changes were incorporated into the guidance equations. They were as follows:

1. A drift compensator was added to sense the attitude steady state error in pitch and yaw.
2. A manual constant was added to bias the insertion yaw velocity. This effectively eliminated the residual drift error not removed by the drift compensator.
3. The time-to-go bias for SECO was adjusted for more realistic consideration of thrust variations.
4. Filter gains and time constants were modified to achieve a compromise between corrective response to drift and radar noise smoothing.
5. Wave guide and transit delay biases in the $\dot{p}$ and $\dot{q}$ lateral rates were added to make the radar correct at SECO.

These equation changes had significant effects upon the 3 sigma accuracy estimates for GT-2, as shown in Table II. D-2. Also shown are the accuracies achieved on GT-2 using the revised equations. From a guidance point of view, the flight was very successful.

The equations, which were originally designed to switch to the track-only mode at 240 seconds after liftoff, were modified for GT-3 to eliminate this mode. This was based on the fact that after the GT-2 flight, GE revised the model of biases on lateral rates, $\dot{p}$ and $\dot{q}$. An analysis of the equations using this new information showed that a combination of rate and track input signals at low elevation angles produced better insertion results than the use of track data alone.

Table II-D-1. Typical Insertion Accuracy Estimates

Variable	Perturbations	Random Errors	Systematic Errors	R.S.S. Total
V	5.34	7.75	2.10	9.7
Γ	.021	.034	.016	.043
V_y'	7.77	3.10	1.14	8.44
r	521	286	372	701

where

V - inertial velocity (ft/sec)

Γ - inertial pitch flight flight path angle (deg)

V_y' - yaw velocity with respect to the target reference plane (ft/sec)

r - geocentric radius (ft)

Table II-D-2. GLV Insertion Accuracy

GT Mission	Velocity		Flight Path Angle		Geocentric Radius		Yaw Velocity	
	Predicted 3 Sigma	Actual Minus Planned	Predicted 3 Sigma	Actual Minus Planned	Predicted 3 Sigma	Actual Minus Planned	Predicted 3 Sigma	Actual Minus Planned
1	24	+18.5	0.200	-.125	5060	-2424	77	-79.5
2	23.7	+ 7.5	0.105	-.01	2103	-1104	24.6	- 4.5
3	29.1	-16.9	0.129	+.04	2103	+ 376	25.3	- 4.5
4	29.1	-13.0	0.129	+.065	2103	+1252	25.3	0
5	29.1	- 2.1	0.129	-.01	2103	- 583	25.3	+ 3.4
6	29.1	-11.6	0.129	+.05	2103	+ 476	25.3	- 6.7
7	29.1	-11.0	0.129	+.05	2103	+ 758	25.3	-12.9
8	29.1	+ 9.0	0.129	-.04	2103	- 264	25.3	-11.4
9	29.1	-16.0	0.129	-.11	2103	-2127	25.3	+ 0.5
10	38.1	- 7.0	0.28	+.008	3942	+ 61	25.3	- 6.0
11	38.7	- 8.3	0.26	+.008	3510	- 202	25.3	+ 5.1
12	38.7	-15.0	0.26	+.0002	3510	- 894	25.3	+ 4.1
Mean Error (II thru XII)	-	- 7.7	-	+.0046	-	- 205	-	- 3.0
Flt Test 3 Sigma Dispersion (II through XII) 75% confidence N = 11		34.2	-	0.186	-	3620	-	23.3
Theoretical 3 Sigma Dispersion	38.7	-	0.26	-	3510	-	25.3	-

Table II-D-3. GLV Guidance Equations Modifications

Vehicle Effectivity	Major Equation Changes	Reasons for Change
GT-2	Drift compensator added.	C G shifts and thrust misalignment not simulated in model caused large errors on GT-1.
	Manual constant to remove yaw bias.	Compensator left residual error, removed by constant.
	Revised filter gains and time constants.	To make equations more responsive to drift.
GT-3	Switchover to track only mode at 240 seconds eliminated.	Receipt of updated GE information on lateral rates, $\dot{p}$ and $\dot{q}$.
GT-6	Maximum payload rendezvous mode included.	Payload considerations.
	Automatic switching to non-rendezvous mode provided.	Range Safety and payload constraints.
	Provision to accept the required final velocity (V_f) transmitted from MCCH by equations in non-rendezvous mode.	To be used if logic switched automatically to non-rendezvous at 105° launch azimuth.
GT-8	A biased mode rendezvous launch was provided for partial wedge angle removal.	If the launch azimuth reached 105° and payload permitted, the GLV could steer out some of the wedge angle.
	Correction of lateral rate $\dot{p}$ and $\dot{q}$ biases to eliminate data error at SECO.	On GT-3, Mod III data showed an increasingly negative flight path angle accompanied by increasing altitude.
GT-10	Revision of predicted insertion accuracies.	Low frequency oscillation in $\dot{p}$ lateral rate observed in GT-8 and GT-9.
GT-11	Revision of predicted insertion accuracies.	For same reason as GT-10, but values arrived at in a more rigorous manner.

As a result of the slow malfunction monitors' experience on GT-2, scaling of the remoted data displayed on their plotboards was changed. Another correction incorporated into the GT-3 equations was the removal of the transit time bias and wave guide delay on $\dot{p}$ and $\dot{q}$ after SECO. Because these acceleration-dependent biases were made to be correct at SECO in the GT-2 equations, an error had been introduced into the data during free flight. For GT-3, the data displayed on the plotboards after SECO was correct. Modifications improving the targeting and the IGS update logic were also implemented in the GT-3 equations.

On GT-3, the GE/Burroughs flight data between SECO and SECO +20 sec showed an increasingly negative flight path angle accompanied by an increasing altitude. This inconsistency did not appear in NASA data, which correctly indicated a positive flight path angle. GE initiated a study of Mod III radar data to determine the reason for this discrepancy. A solution of the problem, however, was not available until GT-8, so that no corrections were included in the GT-4 equations. The equation revisions for GT-4 incorporated differential nodal regression effects in the targeting logic and new initialization constants.

The GT-4 flight resulted in accuracies well within 3 sigma limits, and the flight path angle/altitude discrepancy seen on GT-3 was not apparent. The random noise level late in flight was approximately one-third the value estimated from the GE noise model. Since the noise had been consistently low through all Gemini flights, GE was requested at this time to update the noise model for possible re-optimization of guidance filters and a re-evaluation of accuracy estimates. For GT-5, the equations remained essentially the same as for GT-4, except for minor revisions.

Flight data from GT-5, however, showed the largest magnitude of random noise during the latter part of flight ever experienced using the GE radar as the tracking medium. Although the steering commands reflected the increase in noise level, the achieved guidance accuracies at insertion were the best to date.

Because GT-5 represented an increase in noise level of approximately ten times the magnitude of noise seen on flights GT-1 through GT-4, Aerospace decided not to alter the equations' filters on the basis of the revised GE model delivered after GT-5. Accuracy estimates for the remaining Gemini launches continued to be based on the original conservative noise model.

GT-6 was to be the first rendezvous launch of the Gemini program, with the Agena orbit data and the time of GLV liftoff determining the actual flight trajectory. For this and all subsequent rendezvous missions, however, a nominal set of Agena ephemeris data and specific insertion conditions was initially specified in order to provide a nominal setting of the computer manual constants and a value for the coefficients at guidance initiate. (After each launch, actual ephemeris data was used with a nominal vehicle and noiseless radar to obtain a post-flight reference trajectory.) Trajectory studies were made for the five possible days of launch, indicating the yaw velocity bias (in the form of a computer manual switch setting) for each day. A curve was generated from a series of trajectories for varying times of launch which enabled an adjustment of this bias 20 minutes before liftoff in accordance with real time conditions. This is shown in Figure II. D-4. This method of adjusting the yaw velocity bias was used for all rendezvous launches.

During the months preceding GT-6, NASA mission plans and payload considerations made it evident that the equations would have to incorporate features in addition to the real-time calculation of launch azimuth. To avoid the possibility of human error, a fairly extensive revision of the targeting logic provided the automatic capabilities described below and indicated on the launch azimuth curve of Figure II. D-5.

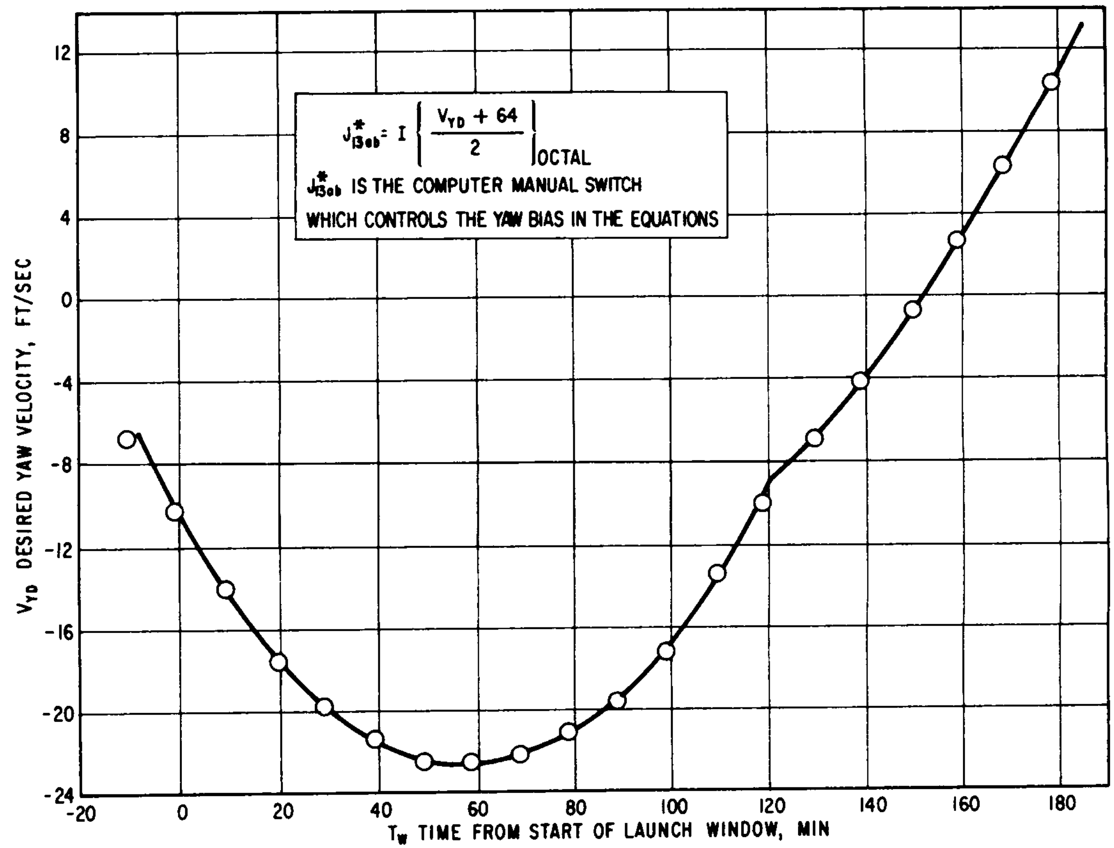
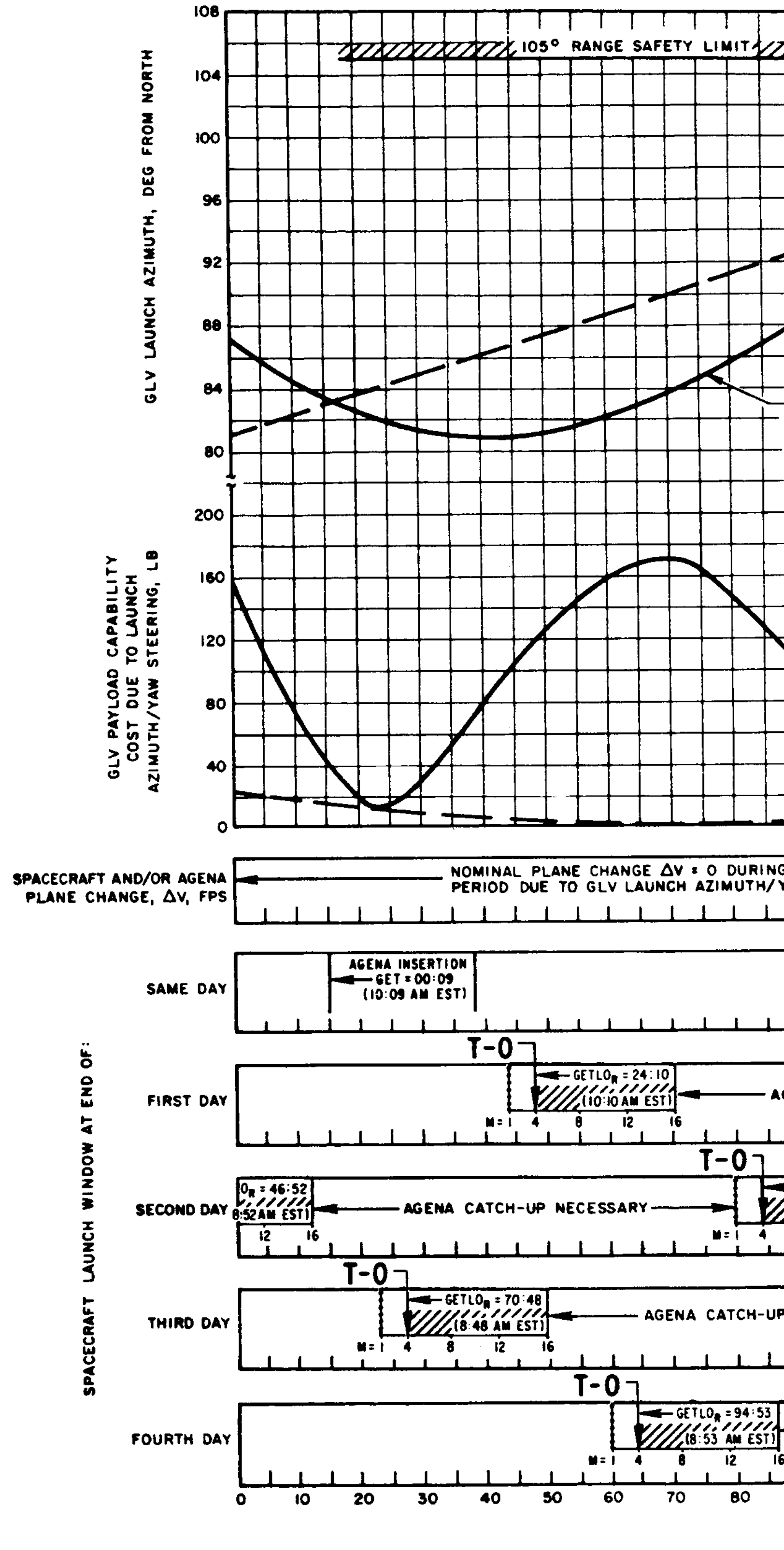


Figure II. D-4. Yaw Velocity Bias vs. Time of Window Opening


GLV LAUNCH AZIMUTH, DEG FROM NORTH
108
104
100
96
92
88
84
80
105° RANGE SAFETY LIMIT
GLV PAYLOAD CAPABILITY COST DUE TO LAUNCH AZIMUTH/YAW STEERING, LB
200
160
120
80
40
0
SPACECRAFT AND/OR AGENA PLANE CHANGE, ΔV, FPS
NOMINAL PLANE CHANGE ΔV = 0 DURING
PERIOD DUE TO GLV LAUNCH AZIMUTH/Y
SPACECRAFT LAUNCH WINDOW AT END OF:
SAME DAY
AGENA INSERTION
GET = 00:09
(10:09 AM EST)
FIRST DAY
T-0
GETLO$_R$ = 24:10
(10:10 AM EST)
M = 1 4 8 12 16
SECOND DAY
O$_R$ = 46:52
8:52AM EST)
12 16
AGENA CATCH-UP NECESSARY
T-0
M = 1 4
THIRD DAY
T-0
GETLO$_R$ = 70:48
(8:48 AM EST)
M = 1 4 8 12 16
AGENA CATCH-UP
FOURTH DAY
T-0
GETLO$_R$ = 94:53
(8:53 AM EST)
M = 1 4 8 12 16
0 10 20 30 40 50 60 70 80

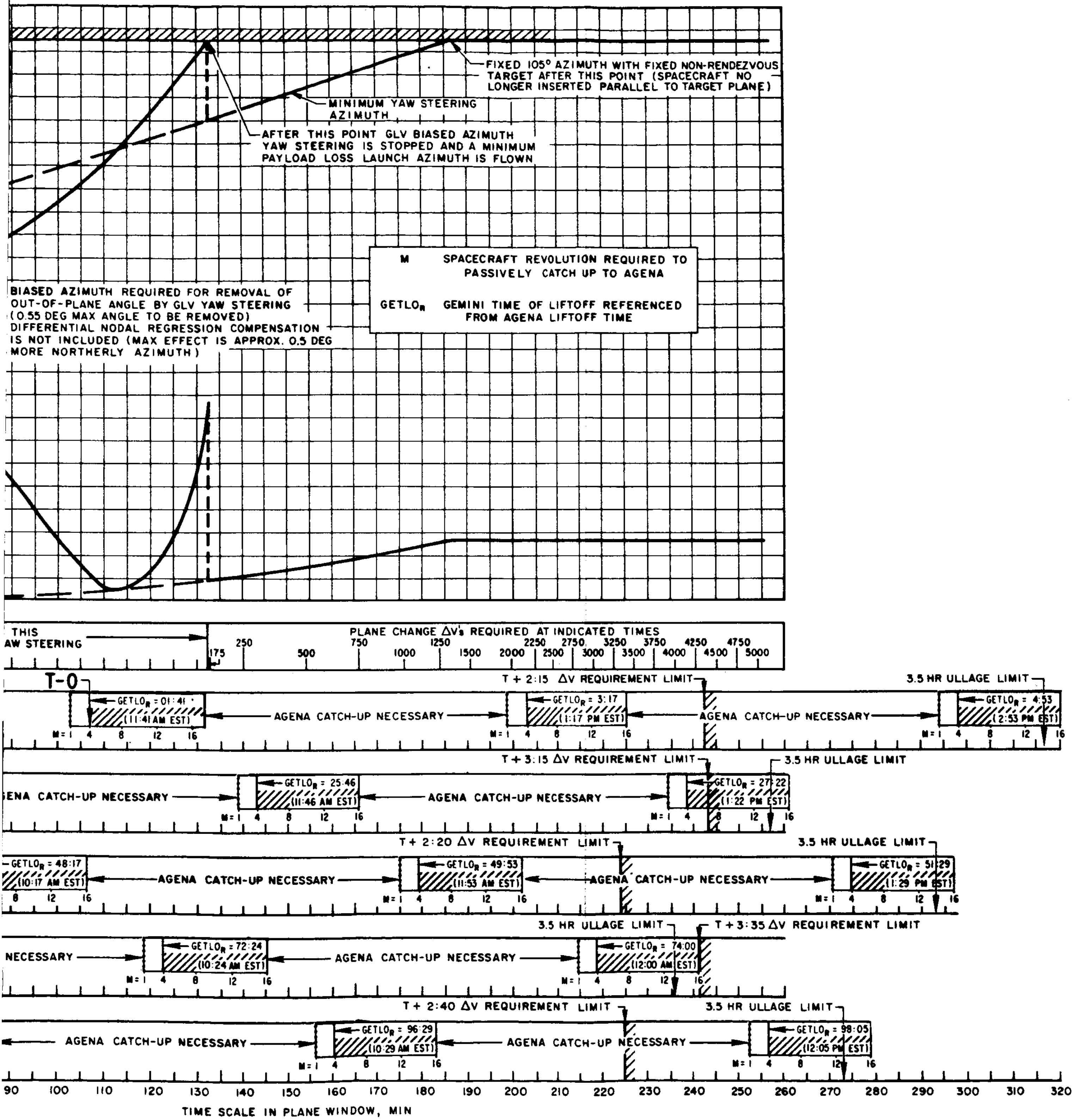


Figure II. D-5. Launch Windows for Gemini VI

1. Decision logic was added so that a manual switch setting would cause the equations to specify a maximum payload launch azimuth when the calculated biased-launch azimuth of 105 degrees was reached. At this point, GLV biased-azimuth yaw steering was stopped and a minimum payload-loss launch azimuth was computed.
2. With the targeting computations in this maximum payload mode, logic was added to provide an automatic switch to the non-rendezvous mode when a launch azimuth of 105 degrees was reached the second time.
3. The equations were changed so that the final velocity transmitted from Mission Control Center/ Houston (MCCH) at L-18 minutes would be used as the required velocity in the non-rendezvous mode as well as for rendezvous. This alteration was compatible with the automatic switch to non-rendezvous described in (2) above.

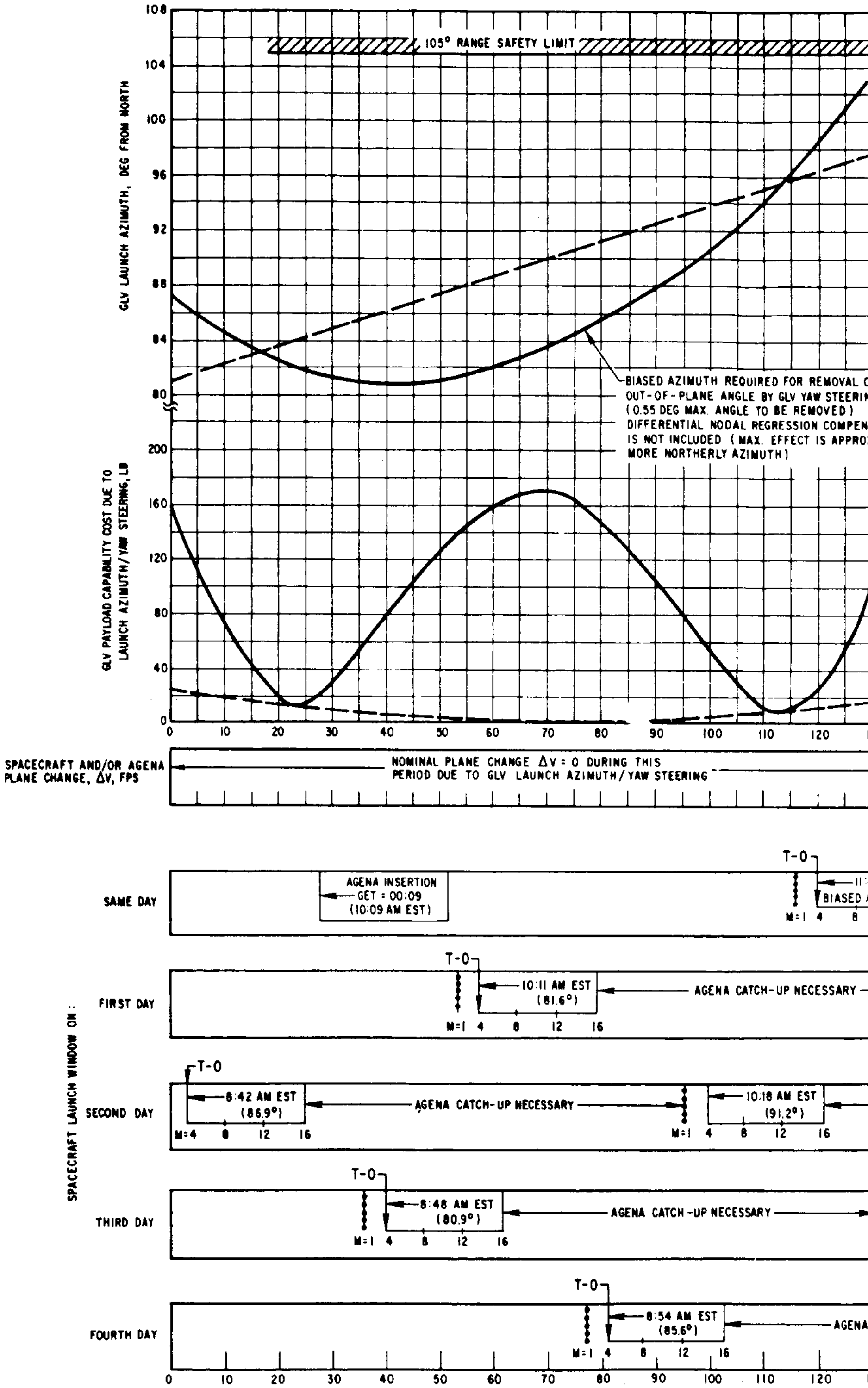
These GT-6 equations, after verification and validation by Aerospace, were wired into the computer trays by Burroughs. Because of the failure of the GT-6 Agena target vehicle on 25 October 1965, the GT-7 (fourteen-day mission) was launched six weeks later and guided into orbit by these same GT-6 equations, switched to the non-rendezvous mode. Eleven days later the same trays, with different manual switch settings, were used to calculate the 81.4^{o} launch azimuth, and to bias steer the GLV-6A to rendezvous with the Gemini 7 capsule. The guidance system performed excellently on both flights.

While the GT-6 equations were being implemented for use in the first rendezvous launch, Aerospace recognized that the 13 unused words remaining in the computer would be insufficient for future equations changes. A major program cleanup was undertaken by Burroughs which obtained 56 additional free cells. This permitted the inclusion of post-SECO plotting on the displays in the Mod III computer room to be used for an immediate indication of insertion parameters. Further targeting modifications requested by MSC were incorporated into the GT-8 equations, providing a new capability--a choice between a new biased mode launch and the parallel mode, controlled by a manual decision switch. The new mode, providing partial wedge angle removal, is shown in Figure II. D-6 at 105^{o} launch azimuth between approximately 130 min and 185 min from opening of the plane window. The launch mode would have been selected on the basis of real-time payload calculations, with the choice indicated to the Burroughs test conductor from Houston over the Missile Operation System (MOPS)

At this time in the Gemini Program a great deal of effort was spent on determining and coordinating procedures between the guidance officer at MCCH and the Burroughs test conductor. Eight pieces of information pertinent to the final setting of manual switches in the Burroughs computer were to be transmitted verbally over the MOPS system at T-60 min. They were as follows:

1. Predicted GLV Greenwich Mean Time of Liftoff
2. Agena inclination angle.
3. Data source of (2).
4. Wedge angle limit (determined on the basis of real-time performance data).
5. J_{6d*} - The decision logic for parallel or biased steering at 105^{o} launch azimuth.
6. GAATV Greenwich Mean Time of Liftoff.
7. Time of Window Opening - (used with Figure II. D-4 to determine the yaw velocity bias set-in.)
8. Time the second 105^{o} launch azimuth would be reached.

This procedure was used successfully on GT-8 and all subsequent launches. It was during the preparation of the GT-8 equations that the results of a study by Aerospace and GE on the flight path angle/ altitude discrepancy observed on GT-3 became available. Lateral rate refraction biases perturbed the Mod III radar, causing an indication of velocity underspeed and positive flight path angle and altitude errors. Corrected values of lateral rate biases were incorporated into the GT-8 equations in order to remove insertion parameter errors at SECO.


105° RANGE SAFETY LIMIT
GLV LAUNCH AZIMUTH, DEG FROM NORTH
108
104
100
96
92
88
84
80
BIASED AZIMUTH REQUIRED FOR REMOVAL O
OUT-OF-PLANE ANGLE BY GLV YAW STEERIN
(0.55 DEG MAX. ANGLE TO BE REMOVED)
DIFFERENTIAL NODAL REGRESSION COMPEN
IS NOT INCLUDED (MAX. EFFECT IS APPRO
MORE NORTHERLY AZIMUTH)
GLV PAYLOAD CAPABILITY COST DUE TO LAUNCH AZIMUTH/YAW STEERING, LB
200
160
120
80
40
0
0 10 20 30 40 50 60 70 80 90 100 110 120
SPACECRAFT AND/OR AGENA PLANE CHANGE, ΔV, FPS
NOMINAL PLANE CHANGE ΔV = 0 DURING THIS PERIOD DUE TO GLV LAUNCH AZIMUTH/YAW STEERING
SPACECRAFT LAUNCH WINDOW ON :
SAME DAY
AGENA INSERTION
GET = 00:09
(10:09 AM EST)
T-0
BIASED
M=1 4 8
FIRST DAY
T-0
10:11 AM EST
(81.6°)
AGENA CATCH-UP NECESSARY
M=1 4 8 12 16
SECOND DAY
T-0
8:42 AM EST
(86.9°)
AGENA CATCH-UP NECESSARY
M=4 8 12 16
10:18 AM EST
(91.2°)
M=1 4 8 12 16
THIRD DAY
T-0
8:48 AM EST
(80.9°)
AGENA CATCH-UP NECESSARY
M=1 4 8 12 16
FOURTH DAY
T-0
8:54 AM EST
(85.6°)
AGENA
M=1 4 8 12 16
0 10 20 30 40 50 60 70 80 90 100 110 120

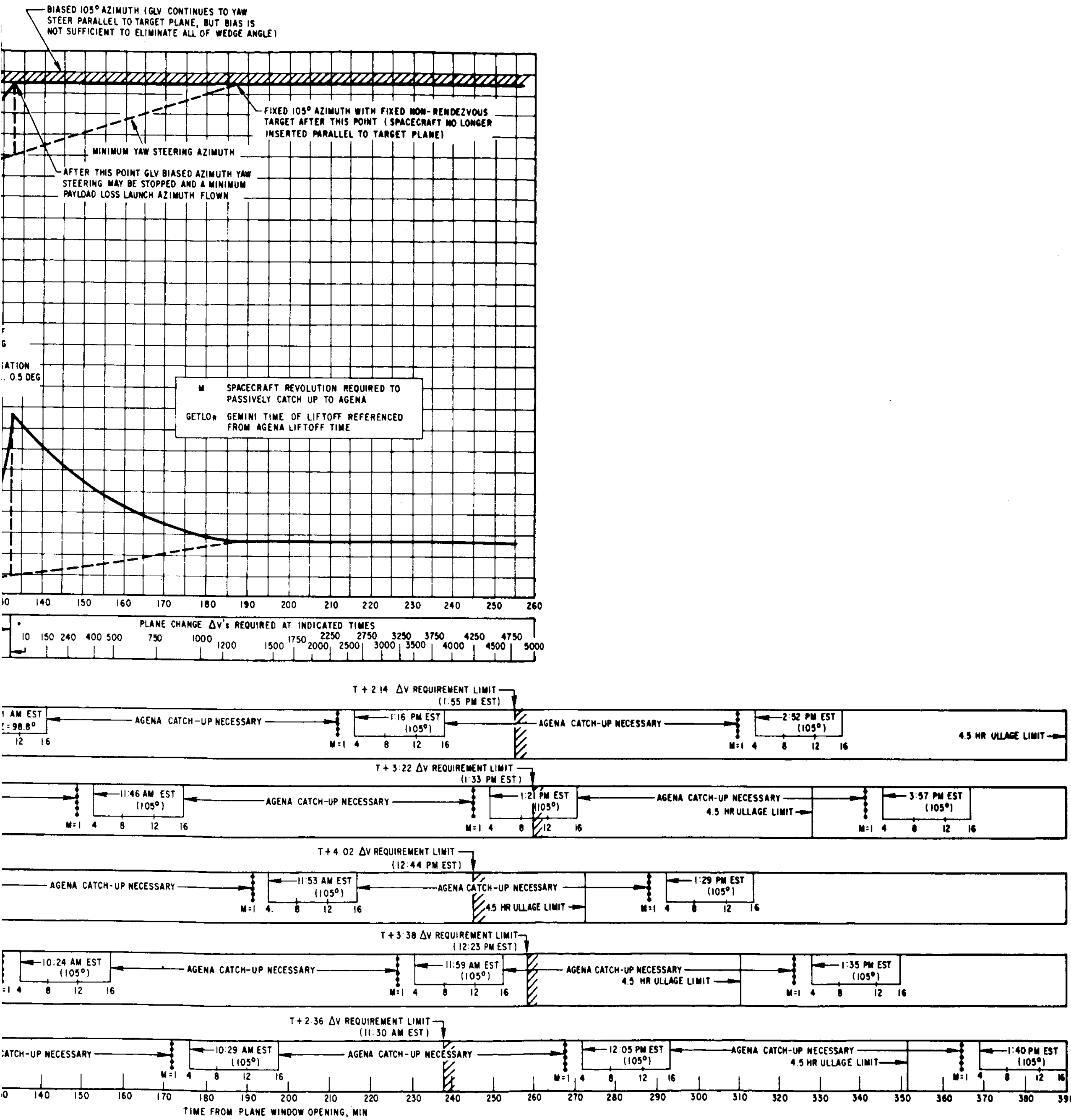


Figure II. D-6. Launch Windows for Gemini VIII

When all modifications were made to the guidance equations for GT-8, only three unused program steps remained in the computer. This condition was unchanged through the final launch.

An anomaly occurred on GT-8 at 7.5 sec before SECO, when the RGS called for an abnormal pitchdown maneuver. An abrupt negative increase in the slope of the $\dot{p}$ data curve was seen at this time, causing a 69-fps pitch velocity error 3 sec later. This $\dot{p}$ anomaly caused pitchdown commands to reach 18 percent of full scale as shown on Figure II. D-7. Study by Aerospace and extensive effort by GE in which comparisons were made between Mod III and Mistram data indicated a low frequency noise on $\dot{p}$, not only in GT-8 data but, to a lesser degree, on past flights as well. The effect, appearing in the $\dot{p}$ channel predominantly and varying inversely as the baseline of the rate leg, was attributed to tropospheric conditions. A simulation made by Aerospace to determine the contribution of the $\dot{p}$ error on insertion accuracy produced the results shown below. The table also shows the contribution of equivalent drifts evaluated by the Martin Company after the flight of GT-8. These can be compared with the insertion conditions in Table II. D-2.

Insertion Parameter	$\dot{p}$ Lateral Rate Error (Mod III-100K Mistram)	Revised Pitch and Yaw Drift Rates	Totals
ΔVelocity, ΔV, fps	8.0	0.1	8.1
ΔFlight Path Angle, $\Delta \gamma$, deg	-0.0045	-0.0081	-0.0126
ΔYaw Velocity, ΔV_y, fps	0	-13.4	-13.4
ΔGeocentric Radius, Δr, ft	55	-188	-133

For GT-9, revised pitch and yaw drift rate prediction techniques were developed by Martin for use in determining yaw velocity bias manual constants. Since the $\dot{p}$ phenomenon was considered by GE to be a random effect not predictable in advance of flight and since it resulted in only one sigma velocity and flight path angle errors on GT-8, no changes were made in the guidance equations.

On GT-9, the Mod III radar again showed a significant low frequency in the $\dot{p}$ lateral rate channel. This drift began at an elevation angle of approximately 10 degrees and caused disturbances in steering before SECO. Aerospace simulations of a $\dot{p}$ low frequency superimposed in the nominal post-flight reference trajectory produced results at insertion closely matching those of the NASA tracking network. Since the anomalous radar performance was caused by tropospheric disturbances, no fix could be incorporated into the GT-10 equations. However, since the simulation showed that the effect could degrade insertion conditions in pitch to approximately a 3-sigma level, it was necessary to reflect this degradation in future insertion accuracy predictions.

For GT-10, the insertion estimates changed considerably as shown on Table II. D-2. Since a GE noise model including these $\dot{p}$ phenomena was not available, flight data with gross assumptions were used for the analysis. Only slight changes were made to the Gemini guidance equations to include a high inclination launch capability.

The guidance equations performed satisfactorily on GT-10. The level of the $\dot{p}$ low frequency oscillation was low and did not noticeably degrade insertion. High frequency random noise was at about the same 3 sigma level as was seen on GT-5.

No changes were made to the guidance equations after GT-10. The predicted insertion parameters, however, were again modified for GT-11. These, shown on Table II. D-2 differed only slightly from the GT-10 estimates, but were obtained by using a more rigorous technique for calculating the effects of the $\dot{p}$ lateral rate anomaly. The $\dot{p}$ errors, as determined from Gemini Mod III/Mistram/IGS comparisons, were combined statistically with other radar and vehicle dispersions.

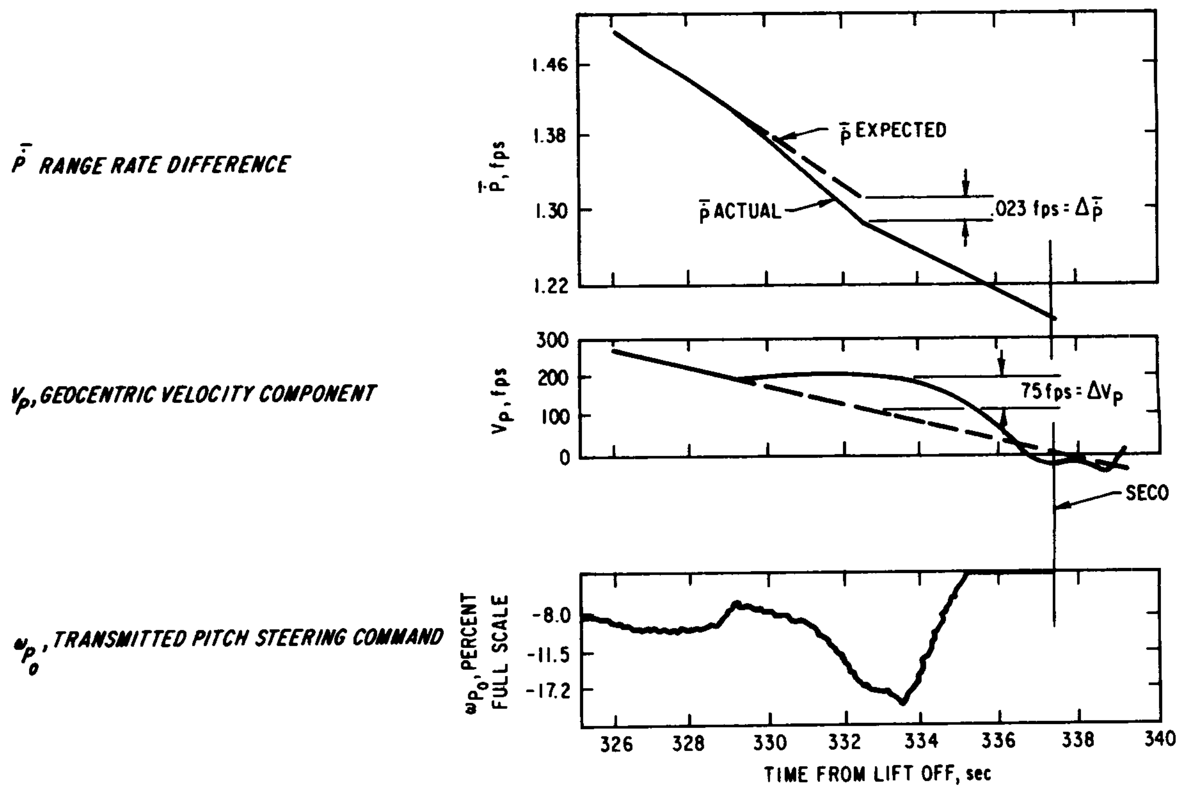


Figure II. D-7. GLV-8 Guidance Anomaly

No guidance anomalies were present on either GT-11 or GT-12, both flown with the GT-10 guidance equations.

It should be mentioned that when a set of equations was used without wired change for another flight, guidance initialization constants were not updated to reflect the new nominal. The velocity was therefore not centered in the ±220 fps velocity gate. This offset, combined with non-nominal vehicle thrust, caused the gate or "glitch test" to be failed on GT-10 and -11. The logic then shifted to the data extrapolation mode for a few seconds, after which normal steering was begun. No degradation of insertion was ever caused by this discrepancy.

In reviewing the overall insertion accuracies of Table II. D-2 which were achieved by the GLV, it should be observed that the flight test dispersions in all cases were close to but somewhat less than the theoretical dispersions. All mean errors were quite small, except for velocity. When the contribution due to RESS tailoff is removed, however, and the insertion velocities of the first seven flights are adjusted to be compatible with the $\dot{p}$ and $\dot{q}$ bias adjustments made in the equations prior to GT-8, the true guidance velocity mean error is reduced to -2.5 fps and the flight test 3-sigma dispersion is reduced to 26.4 fps (compared to 34.2 fps unnormalized).

2. PERFORMANCE ANALYSIS

a. Development of Payload Capability and Trajectory Prediction Techniques

(1) Simulation Models and Inputs

(a) Simulation Model. Throughout the Gemini Program, Aerospace Corporation trajectories were prepared with the Aerospace "N-Stage Simulation". This was a high speed, six-degree-of-freedom digital computer simulation. In the simulation, the ellipsoidal earth was simulated by means of the Fischer Ellipsoid. The Patrick Reference Atmosphere (Annual) was utilized up to 150,000 ft, while the nominal 1959 ARDC Model Atmosphere was utilized above 150,000 ft. The gravitational potential function utilized powers through 4 of geocentric radius, as well as powers through 4 of the sine of geocentric latitude. Launch and radar site locations were furnished in Fischer coordinates by NASA and the General Electric Company

The launch vehicle simulation itself was highly detailed, and included a complex, time-varying engine model. The capability was provided to calculate and print, at any desired frequency, over 400 trajectory, engine performance, and guidance parameters. A complete simulation of Gemini Launch Vehicle guidance equations was employed to simulate Stage II guidance.

During the early phases of the Gemini Program, use was made of Titan II weapon system data wherever possible. These included vehicle weight, engine performance, and drag characteristics. Later, specific launch vehicle data supplanted the Titan II data. Wind tunnel tests were run to obtain improved estimates of drag, lift, and moment coefficients.

Approximately three thousand N-Stage Simulation trajectory runs were made by the Aerospace Corporation throughout the Gemini Program. Many of these were based upon nominal parameter values which applied to all twelve Gemini Launch Vehicles. However, prior to each flight, specific parameters relating to particular launch vehicles were incorporated to prepare best preflight estimates of the launch vehicle trajectory. These best preflight estimates of the nominal launch trajectories were published in the Systems Test Objectives documents issued by Aerospace Corporation prior to each flight.

(b) Incorporation of Specific Vehicle Parameters. At the beginning of the Gemini Program, all trajectory and payload performance predictions were based upon nominal values for all parameters. Therefore, all launch vehicles had the same payload capability except for variations due to mission differences. As vehicle parameters became available they were incorporated into predictions. Frequently, incorporation of the specific values created substantial changes in predicted payload capability.

For example, when the actual engines planned for GLV-10 were incorporated in payload capability predictions, payload capability changes due to performance differences from nominal were -47 lb for Stage I specific impulse and -75 lb for Stage II specific impulse. When actual propellant tank volumes were utilized in predictions for GLV-4, the payload capability changes were +2 lb due to Stage I volume differences and +12 lb due to Stage II volume differences from nominal. When the actual pitch programmer and power supply to be used on GLV-8 were measured, the payload capability change from the previously assumed nominal value was -48 lb. When actual propellant tank weights had been measured for vehicles 1 through 6, it was determined that, on the average, the tank weights were lighter than the previously used predictions. Consequently, the weight estimates were reduced, with a corresponding +11 lb change in payload capability due to Stage I tank weights and +50 lb due to Stage II tank weights. These changes were applied to vehicles 7 through 12. Then, when the actual tank weights for these latter vehicles were measured, the payload capability effects of the measurements were incorporated in predictions.

Other examples of significant payload capability changes due to incorporation of measured vehicle parameter values are available. Each change was incorporated as soon as it became known in order to have available the most up-to-date prediction possible. This was desired to keep NASA continually informed regarding the payload capability margin for each of the vehicles, and provide them with sufficient information so that mission changes could be made to improve payload capability or take advantage of excess capability. It was also desired to show the desirability (or necessity) or making performance improvement changes to the Gemini Launch Vehicle. A number of performance improvements were considered for the Gemini Launch Vehicle during the early and mid-phases of the program. These are discussed in other sections of this report.

Figure II. D-8 illustrates the changes in predicted GLV minimum payload capabilities versus time and changes in Gemini Spacecraft weights, without experiments, versus time. Since the experiments averaged about 160 lb for S/C 3-12, the actual margins between predicted capabilities and spacecraft weights were less than those depicted in Figure II. D-8.

(2) Constraints

As with any launch vehicle, the Gemini Launch Vehicle was constrained to remain within specified limits throughout its flight envelope. In particular, the GLV was constrained by aerodynamic heating,

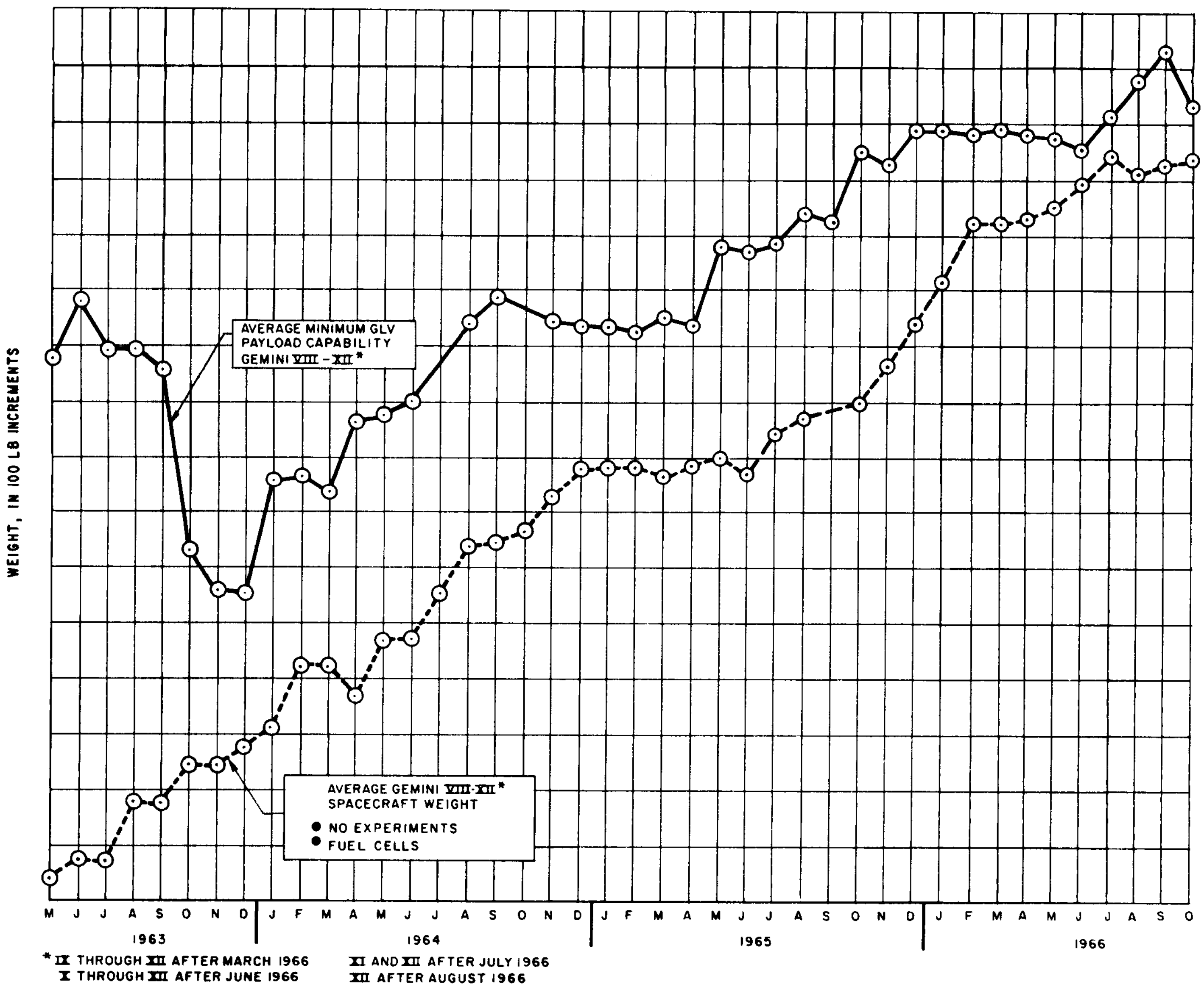


Figure II. D-8. History of Spacecraft Weight (Without Experiments) and Predicted GLV Minimum Payload Capability for Missions VIII through XII aerodynamic loads, axial acceleration, guidance radar look angles, guidance radar elevation angle, dynamic pressure and angle of attack at staging, Stage I hydraulic actuator hinge moment, and spacecraft abort criteria.

Studies were concluded early in the Gemini Program which quantitatively established limits in these constraint areas. Maximum or limiting values of some parameters were selected for nominal trajectories such that, if the nominal trajectory remained within these bounds, dispersed trajectories would remain within the true launch vehicle and guidance system capabilities. A brief description of each of the constraints utilized throughout the Gemini Program is provided below.

1. Aerodynamic Heating - The maximum allowable heating for the GLV was based upon the difference between allowable and actual stresses, and included heating effects due to angle of attack and effects of combined heating and loads. All analyses indicated that, even with maximum dispersions, the GLV remained within its allowable stress limits.

2. Aerodynamic Loads - The maximum dynamic pressure acceptable for the nominal trajectory was 780 psf. The maximum longitudinal acceleration for the normal trajectory was 5.75 g for Stage I and 7.54 g for Stage II. If the nominal trajectory remained within these bounds, dispersed trajectories would remain within the launch vehicle aerodynamic heating and loads capabilities.

3. Flight Controls and Guidance -

   (a) Stage I - TARS Pitch Program - Trajectory shaping during Stage I flight was chosen to optimize payload, subject to the constraints of heating and loads discussed above. Checks were made on the altitude and flight path angle at BECO to ensure that look angle constraints in subsequent Stage II flight were not violated.

   (b) Stage II - Mod III Guidance - Pitch look angle constraints for the Gemini Launch Vehicle were defined in an Aerospace report[1] as functions of yaw look angle limits. Analysis indicated that Gemini flights limited to $\pm 0.55^\circ$ of wedge angle removal by launch azimuth biasing and Stage II yaw steering would remain within the $\pm 20^\circ$ yaw look angle boundary used throughout the Gemini Program.

   A possible constraint, occurring at the end of guidance, was the tracking elevation angle. An exact limit on this was not determined, but it was believed that angles lower than 6.5° could have caused excessive radar noise, primarily in the rate system, which could have degraded guidance accuracy.

4. Staging - Maximum q at staging was constrained to less than or equal to 50 psf, while angle of attack was maintained less than or equal to 3°.

5. Hinge Moment - The hydraulic actuators used to position the Stage I engine thrust chambers had to have the capability of withstanding not only inertial forces but also the external torques produced by aerodynamic forces on the thrust chambers. The GLV Stage I actuator torque capability was 16,500 ft-lb.

   The load torques (hinge moments) on the actuators of the two Stage I thrust chambers were calculated for the most severe combination of loads following a malfunction and switchover in the vicinity of maximum dynamic pressure. The total load torque did not exceed 14,500 ft-lb, which was well within the Stage I actuator torque capability.

6. Abort Criteria - The velocity and flight path angle for the dispersed launch vehicle trajectory could not exceed the boundaries required for safe spacecraft abort. Abort boundaries were defined by NASA for each mission.

(3) Dispersion and Probability Analyses

While the nominal payload capability for each Gemini Launch Vehicle was of considerable importance, of even greater importance was the predicted minimum payload capability. The minimum payload capability was that weight of spacecraft that could be put into the desired orbit even under the "most disadvantageous" launch vehicle performance. "Most disadvantageous" was defined for the Gemini Launch Vehicle as being the minus 3-sigma payload capability, otherwise described as being that payload capability which would be equalled or exceeded 99.87% of the time. This percentage was shifted to 99.4% in the latter part of the Gemini Program, as will be discussed below.

Gemini Launch Vehicle dispersion analyses were initially carried out by determining the payload capability effects of dispersions in a large number of key vehicle parameters. The parameter dispersions that were used were the 3-sigma dispersions, which were based upon test data and theoretical analyses. The Martin Company provided dispersion analyses (LV-274 report series) for all 12 vehicles. Besides showing payload capability variations due to parameter dispersions, this series of reports also showed root-sum-square dispersions of trajectory parameters versus time from liftoff. Among these dispersions were Mach number, dynamic pressure, and heating parameter. From time to time, Aerospace Corporation ran a series of trajectories to check the Martin Company results in particularly significant areas.

Throughout the Gemini Program, attention was given to refining estimates of 3-sigma parameter dispersions. Particular attention was given to those parameters which had the most significant effects upon trajectory and payload capability performance. Thus, Aerospace, the Martin Company, and Aerojet-General Corporation were continually monitoring engine thrust, specific impulse, and mixture ratio performance to determine 3-sigma dispersion estimates based upon flight test results. This monitoring of engine performance resulted in dispersion changes during the Gemini Program. Likewise, pitch programmer performance was studied by the Martin Company and Aerospace to better define the nominal and dispersed performances. The individual pitch programmer bias from actual preflight tests on the hardware was used in performance predictions for GLV 5 through 12.

Each of the individual parameter dispersions was assumed to be independent of all other dispersions in dispersion analyses performed early in the Gemini Program. Each dispersion, too, was assumed to be normally distributed about the mean. These assumptions permitted root-sum-squaring of the individual payload capability effects to produce an overall 3-sigma payload capability dispersion. It was clear from the beginning of the Gemini Program that a very good estimate of the overall 3-sigma dispersion could be determined by considering the variations of a limited number of key parameters. These parameters were those which most affected the shape of the vehicle trajectory in the pitch plane. The parameters which were selected by Aerospace early in the program and used throughout for simplicity and continuity are as follows:

Stage I	Stage II
Thrust	Thrust
Specific Impulse	Specific Impulse
Outage	Outage
Dry Weight	Dry Weight
Usable Propellant Weight	Usable Propellant Weight
Pitch Programmer Error	
Pitch Gyro Drift	
Winds	
Atmospheric Density	
Engine Thrust Misalignment in Pitch	

Since the distribution of outage is non-Gaussian, the 99% probability level of outage was used in the root-sum-square process. Since outage was such a dominant factor in determining the predicted payload capability dispersion, the resulting overall root-sum-square was actually some probability level between 99.87% and 99%.

To provide an example of the relative sizes of the individual parameter dispersions and corresponding payload capability dispersions, the predicted GLV-12 dispersions are provided in Table II. D-4. It will be observed in this table that two different outage dispersions are provided for each launch vehicle stage. It was determined after the launch of GT-1 and prior to the launch of GT-2 that use of outage dispersions with the assumption of nominal predicted propellant temperatures was not conservative. Experience with GT-1 had shown that the propellant temperature conditioning equipment at ETR and the uncertainties in launch weather made the likelihood of achieving predicted propellant temperatures in all tanks at liftoff exceedingly small. Since other propellant temperature conditions would result in different outage dispersions, it was necessary to reevaluate outage dispersions based upon estimated temperature dispersions. It was eventually concluded that the effects of the estimated temperature dispersions could be approximated by employing the effects due to dispersions in oxidizer alone of ±2 deg Fahrenheit. This explains the inclusion of the outage dispersions in Table II. D-4 for the ±2 deg F propellant temperature tolerance. It also explains the change in nominal and minimum payload capability that occurs in going from the nominal temperature case to the -2°F temperature case.

For GLV's 9 through 12, a more sophisticated approach was taken to determine the probabilities associated with the predicted payload capability dispersions. In particular, the outage dispersions were treated as non-Gaussian. The root-sum-square of the other payload capability dispersions was combined through a double convolution process with the non-normal Stage I and Stage II outage dispersions to result in a cumulative probability function. This function expressed the probability that achieved payload capability would be equal to or greater than a given level. As a result of this improved probability analysis, the probability level associated with the overall payload capability dispersion predicted for these GLV's ranged from 99.4 to 99.3%.

These cumulative probability functions were computed for both the nominal propellant temperature case and the -2 deg F propellant temperature tolerance case. The -2 deg was chosen in preference to the +2 deg case because it represented the more pessimistic payload capability at the planned launch time. An example of the predicted probabilities is shown in Figure II. D-9. This is the figure that was used to predict probabilities of nominal insertion for GLV-11 and GLV-12. It is assumed in the figure that the probability of desired insertion is identical to the probability of achieving the indicated payload capability level. Desired insertion is defined, in this case, as one involving a nominal GLV Stage II shutdown and tailoff, followed by a nominal spacecraft injection into orbit.

Also shown in Figure II. D-9 are the achieved payload margin differences from nominal. The normalized differences from nominal are compatible with the assumptions used in drawing the curves. The normalizing factors are also listed in Section II. D-2-C(5). It will be seen that the normalized achieved payload margin differences from nominal agree well with the prediction curves, thereby attesting to the validity of the predictions.

(4) Performance Reporting and Updating

Gemini Launch Vehicle payload capabilities were computed by both Martin-Baltimore and The Aerospace Corporation, and reported in a variety of ways. The Martin Company issued a series of monthly reports (LV-22-) in which loadings, weights, tank volumes, and outages were specified for each launch vehicle. Dispersions and payload capabilities were computed and presented as functions of launch time and launch day. In addition, Martin issued a specification change notice prior to each flight and updated the payload capability predictions for that flight. This was further refined at F-2 Days to include effects of predicted weather at ETR. Real-time prediction techniques were used during the loading and countdown phases of each launch.

Table II. D-4. GLV-12 Dispersions

Stage	Parameter	Maximum Dispersion (3σ)	Effect on Payload, lb
I	Thrust, %	2.4	106
I	Specific Impulse, sec	1.7	118
I	Outage, lb — 1 Predicted Propellant Temperatures	1,703	279
I	Outage, lb — 2 ±2°F Propellant Temperature Tolerance	1,672	274
I	Dry Weight, lb	21	3
I	Usable Propellant Weight, lb	876	36
I	Pitch Programmer Error, %	1.87	105
I	Pitch Gyro Drift, deg/hr	39.2	108
I	Winds	Directional Modified Synthetic 1% Avidyne	108
I	Density	Extreme PAFB Profile	24
I	Engine Thrust Misalignment, Pitch, deg	0.314	89
II	Thrust, %	3.2	19
II	Specific Impulse, sec	2.3	192
II	Outage, lb — 1 Predicted Propellant Temperatures	413	449
II	Outage, lb — 2 ±2°F Propellant Temperature Tolerance	406	441
II	Dry Weight, lb	13	13
II	Usable Propellant Weight, lb	197	17
	Root-Sum-Square	1 Predicted Propellant Temperatures	622
		2 ±2°F Propellant Temperature Tolerance	614

Reduction in both nominal and minimum payload capability for -2^{o} propellant temperature case		-56
OVERALL ALLOWANCE FOR NON-NOMINAL PERFORMANCE (including related probability)	1 Predicted Propellant Temperatures	622 (99.4%)
	2 -2^{o} Propellant Temperature Case	670 (99.3%)

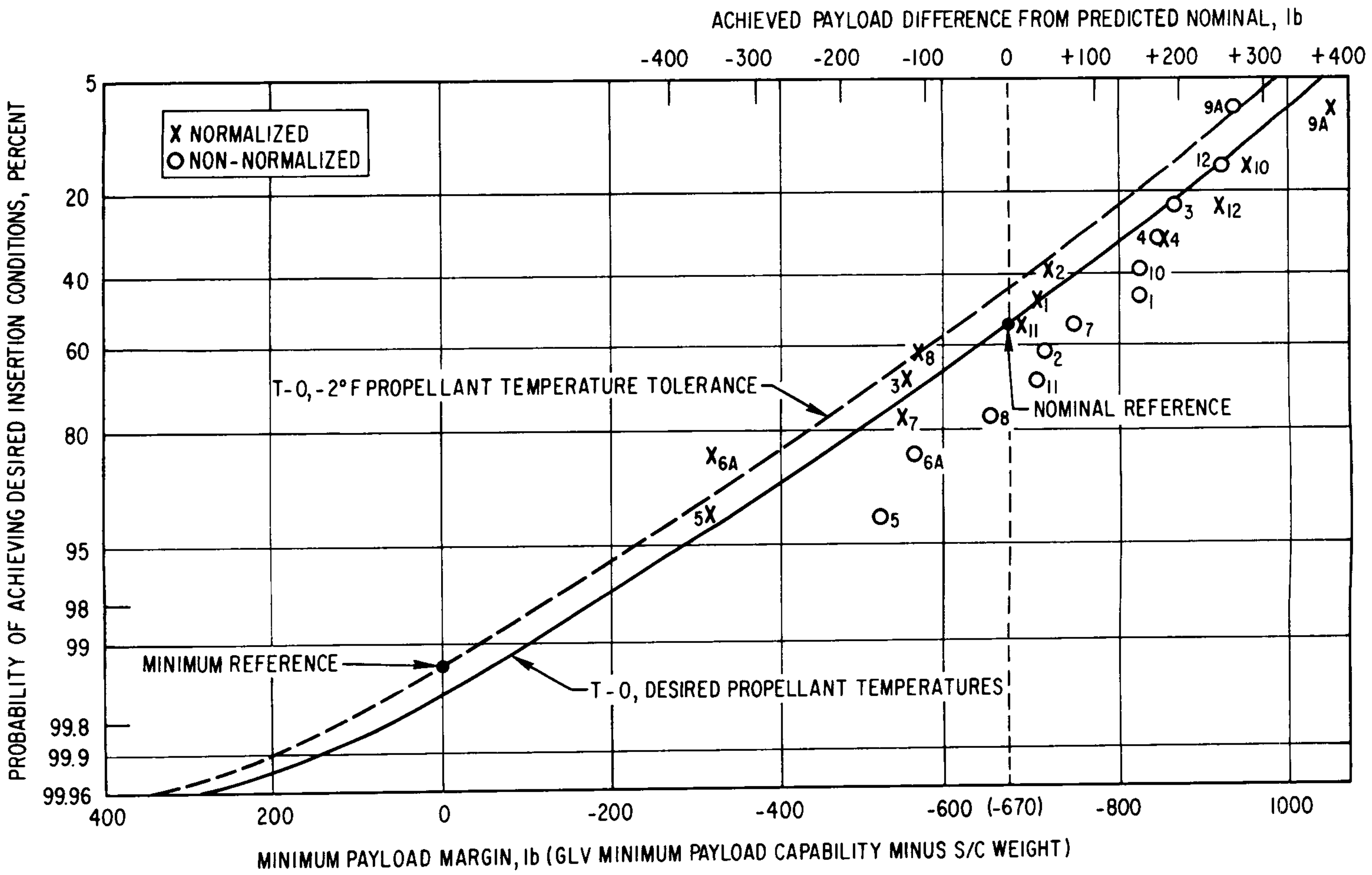


Figure II. D-9. GLV Insertion Probability

The Aerospace Corporation presented payload capability estimates to NASA/MSC via a series of monthly performance reports. The GPO panel meetings at Houston were also used to present payload capabilities and discuss potential improvements or degradations. Detailed performance versus probability predictions were given to NASA at the Technical Review which preceded each flight and these were updated at the Flight Safety Review Board briefings at ETR to reflect last minute changes. During the countdown phase, Aerospace reviewed the Martin real-time computerized payload calculations and performed manual backup calculations using measured propellant temperature data.

The Aerospace monthly performance reports contained a thorough review of the performance capability for each mission and discussed performance changes in great detail. The summary charts included both nominal and minimum payload capabilities as well as data on spacecraft current and predicted weights, experiment weights, and predicted launch margins. Some of the typical reported changes which affected overall payload capability included variations in thrust and specific impulse, propellant loads, pitch program biases, burnout weights, target ephemeris data, and trajectory simulation. A running account of spacecraft weight growth and GLV payload capability was presented as shown in Figure II. D-8. Bar charts such as that of Figure II. D-10 were included to show achieved versus predicted capabilities. Probability curves were also included to show the effect of weight or capability changes on mission success. A table of exchange ratios and curves of payload versus launch azimuth and launch window were developed to indicate how these factors affected payload capability. Finally, all current studies which might have had a potential effect on payload capability were reviewed and discussed in detail.

(5) Propellant Temperature Prediction, Control, and Monitoring

Propellant temperatures had to remain within certain limits because of engine operating requirements, maintenance of minimum ullage volumes, and achievement of a propellant consumption mixture ratio close to the value upon which the propellant loading and payload capability were based. The importance of this area was emphasized at T-5 hours during prelaunch operations on GT-1 when the predicted "oxidizer equivalent temperatures"* for T-0 were too high (by over 5° for Stage I and over 10° for Stage II), resulting in a calculated payload capability of approximately 700 lb lower than that expected. This would have resulted in a minimum payload capability of over 400 lb less than the spacecraft weight. Actual temperature rise rates lower than predicted by the computer program, along with manipulation of erector curtains and polyethylene wrap, resulted in barely acceptable temperatures by launch time.

After the GT-1 flight, considerable effort was expended to obtain desired propellant temperatures. Improvements were made in propellant temperature instrumentation in the AGE, loading procedures, erector curtain control, weather predictions, propellant heating prediction computer program, real-time calculation of payload capability and ullage limits, and dissemination of information to launch operations personnel. Updating of some items such as the Martin propellant heating prediction program continued after each launch.

Tight tolerance goals of $\pm 2^{\circ}$ oxidizer equivalent temperatures for both stages were set. By maintaining these tolerances, the predicted minimum payload capability for a 3.5 hour launch window, with mixture ratio optimized for the middle of the window, could be held within a range of 120 lb.

*Oxidizer equivalent temperature is defined as the difference between the oxidizer temperature predicted and the oxidizer temperature desired for the fuel temperature predicted.

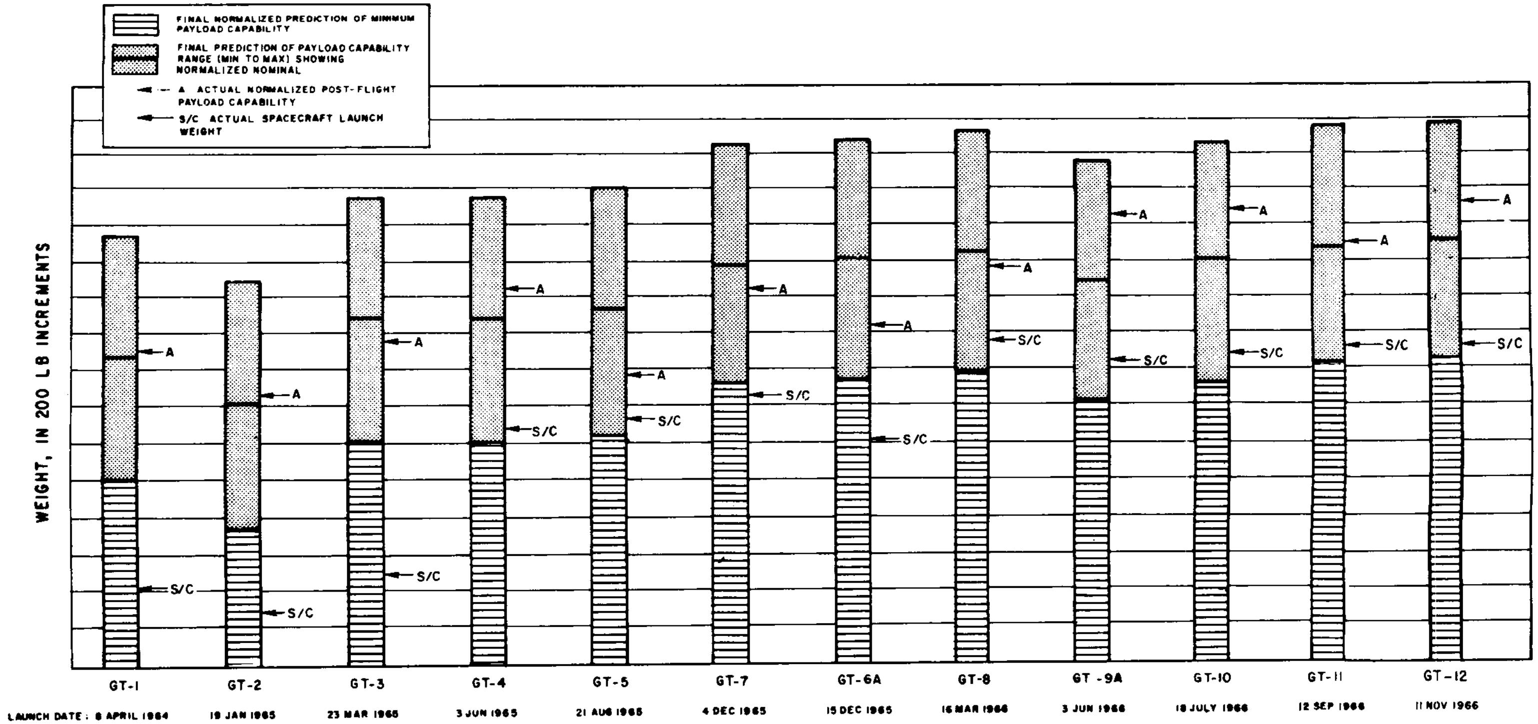


Figure II. D-10. Comparison of Normalized Predicted and Achieved Payload Capabilities

Satisfactory propellant temperatures were obtained on GT-3 through GT-12, since actual flight propellant temperatures were generally within the desired constraints.

b. Performance Improvement Program

(1) Spacecraft Weight Growth and Mission Requirements

Since the inception of the Gemini Program, a vigorous program of payload capability improvement was pursued to meet the ever-increasing requirements in this area. Initially, it was estimated that the total weight of the spacecraft, including experiments, would run about 7000 lb for the long duration missions and 7250 lb for the rendezvous missions. It quickly became apparent that these would be exceeded. The early spacecraft weight growth rate was approximately 35 to 40 lb per month, and it was not until deletion of the paraglider configuration that some relief was obtained. Increase in the size of the spacecraft OAMS tanks provided another spur in the search for higher GLV payload capability. Ultimately the spacecraft weights grew to the point where predicted GLV performance margins relative to the minimum (99.4% probability) payload capability were consistently negative. Comparison between actual spacecraft weights and achieved payload capabilities is shown in Figure II. D-10.

In addition to spacecraft weight increases, changes in mission requirements had a significant effect on GLV payload capability. On early flights a five-hour launch window requirement was imposed, necessitating large ullage volumes in the propellant tanks to allow for propellant temperature increases. This in turn meant less propellants loaded and a reduced GLV payload capability. Optimizing the mixture ratio for the worst case in the window under dispersed propellant temperature conditions also resulted in performance decreases. For certain missions the requirements for high initial apogees and for launch azimuths considerably less or greater than 90 deg degraded the GLV payload capability. Finally, the requirement to have the launch vehicle steer out as much as 0.55 deg of wedge angle to increase availability of OAMS propellant in the spacecraft, reduced the GLV probability of achieving the desired insertion conditions.

(2) Preflight Improvements

(a) Colder Propellants - Propellant temperature conditioning equipment was included in the AGE so that propellants could be chilled to 20°F for oxidizer and 26°F for fuel before loading. This would allow greater propellant masses to be loaded in the fixed GLV tank volumes and thus increase payload capability.

The payload capability advantage in going to chilled propellants was reported in 1964 as +190 lb due to reducing ullage limit temperatures from 65°F to 50°F (+225 lb) and re-orificing engines to obtain target mixture ratios at 40°F rather than 65°F (-35 lb).

A new factor of temperature effect on propellant enthalpy has been introduced, which was not in the original engine model or analyses. If the enthalpy effects are correct* (there is a question since the Stage II value used is about twice as large as the theoretical), the 190 lb gain would drop to 100 lb.

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*Enthalpy effects on I_{sp} presently being used are:

$$\left(\frac{+0.0255 \text{ sec } I_{sp}}{+1^{\circ}\text{F propellant temp}}\right)_{\text{Stage I}} \quad \text{and} \quad \left(\frac{+0.05 \text{ sec } I_{sp}}{+1^{\circ}\text{F propellant temp}}\right)_{\text{Stage II}}$$

Table II. D-5 contains updated payload capability increases based on reviewing the problem under the recent GLV mission requirements, actual GLV and Titan III propellant temperature experience, and present temperature influences on payload capability. As indicated, actual payload capability gained was from approximately 250 lb to 40 lb, depending upon the validity of the enthalpy effects.

Also shown is the payload capability advantage that would have accrued in going to constant temperature propellants under recent mission requirements. This information may be of value to the Titan III program since consideration is being given to conditioning propellants on some of the missions without reorificing the engines. It appears that if the enthalpy effects for Stage II are correct, there is a slight loss in performance capability for Stage II in going to colder propellants (determined from Case 3: item 4 plus item 6 for Stage II).

(b) Weight Reduction. Another of the steps taken to increase GLV payload capability was the initiation in 1962 of a formal weight reduction program. The Martin Company reviewed each subsystem and listed those areas in which some weight reduction could be effected. This list was reviewed by SSD/Aerospace in terms of costs, effectivity, and technical feasibility. Some items were deleted, and the remainder was presented as a proposal for formal evaluation. Recommendations were then made to NASA/MSC and direction was received to implement some of these items. Table II. D-6 lists the improvements proposed, gain in terms of payload improvement, and whether or not these were implemented. A total of 122 lb of payload capability was gained from this effort.

A fortuitous gain in performance resulted from the weighing of Stage II at the factory. It was observed that this portion of the vehicle was approximately 65 lb lighter than the design weights. Since this was consistently observed in the first six vehicles, the nominal weights for the remaining vehicles were adjusted accordingly, resulting in a net payload capability increase of 65 lb.

(c) Reduced Minimum Ullages. In 1962, attention was given to the performance gain available by reducing the minimum ullages* in the propellant tanks from the values used on the Titan II weapon system. Structural studies were conducted by Martin and engine start tests at reduced ullages were incorporated in the AGC Gemini Propulsion System Test Program. Minimum ullages were reduced, effective for GLV-1, as indicated below.

Tank	Minimum Ullage, ft^3		GLV Payload Capability Gain
	Titan II	GLV	
Stage I:			
Oxidizer	97	48 (-50%)	+270 lb
Fuel	79	39 (-50%)	
Stage II:			
Oxidizer	83	63 (-24%)	+ 60 lb
Fuel	20	16 (-20%)	

This change yielded the largest single GLV performance gain in the Gemini Program.

*Minimum ullage is defined as the minimum gas space required in a propellant tank at engine ignition to ensure adequate tank pressurization and engine inlet pressures.

Table II. D-5. Payload Capability Effect of Cold Propellants

Item	Case 1 Unchilled TITAN III Experience Applied to GLV	Case 2 GLV Experience With Chilled Propellants	Case 3 Constant Temperature Propellants
1. Engine Acceptance Target Mixture Ratio	Stage I: 1.93 Stage II: 1.80 with 75° propellants	Stage I: 1.93 Stage II: 1.80 with 40° propellants	Stage I: 1.93 Stage II: 1.80 with 40° propellants
2. Ullage Limit Temperatures	≈ 85° for both stages	≈ 50° for both stages	39° for both stages
3. Flight Temperatures for Mixture Ratio Optimization	≈ 80° for all propellants	≈ 45° for all propellants	37° for all propellants
4. Payload Capability Gain Due to Increased Propellant Load, Including M/R Effect on Isp	- 0 -	Gain over Case 1: Stage I: +173 lb Stage II: + 74 lb +247 lb	Gain over Case 2: Stage I: + 95 lb Stage II: + 38 lb +133 lb
5. Payload Gain Due to Decreased Outage Variation Through A 1.5 hr. Launch Window	- 0 -	- 0 -	Gain over cases 1 and 2: ≈ +80 lb
6. Payload Loss Due to Lower Propellant Enthalpy	- 0 -	Loss over Case 1: Stage I: - 62 lb Stage II: -146 lb -208 lb	Loss over Case 2: Stage I: -18 lb Stage II: -40 lb -58 lb
7. Net Payload Effect (If Item 6 is correct)	- 0 -	Gain over Case 1: ≈ +40 lb	Gain over Case 2: ≈ +155 lb

Table II-D-6. GLV Weight Reduction Program

Proposed Weight Reduction Item	Payload Gain (lb)		
	Effected	Rejected	Reason for Rejection
Deletion of T-II Vernier Provisions	4.5		
Bonded Wire Bundle Clips to Replace Welding		11.2	High Cost
Delete Provisions for 2 Telemetry Antennas	7.0		
Delete Unused Dome Lands	16.0	1.0	High Cost for Stage I
Reduce Chem-Milling Tolerances	16.0	1.8	
Delete Unused T-II Parts	1.8		
Eliminate Compartment 3 Air Conditioning Provisions	3.9		
Delete Spare Wires	6.0		
Delete Unused Telemetry Wire		33.0	Retained for Contingency
Redesign Stage II Fill and Drain System		23.3	High Cost
Lightweight Command Receiver	20.0		
Lightweight Telemetry Transmitter		25.5	High Cost
Removal of Telemetry Program Board and 2 Telemetry Antennas	47.0		
TOTALS	122.2	95.8	

(d) Shutdown Level Sensor Removal. Early in 1963 Martin proposed that the feasibility of removing the low level propellant shutdown sensors from the shutdown circuits on both stages of the GLV be studied. This would eliminate the large possibility of premature shutdowns due to faulty level sensor operation and would also increase payload capability by reducing the amount of trapped propellants. Data from exhaustion shutdowns on the test stand and on Titan II R and D flights indicated that such shutdowns did not noticeably jeopardize mission success. The shutdown function of the sensors was eliminated although they were retained for instrumentation purposes and for closed loop operation if later found desirable. Payload capability gains due to this change were 27 lb for Stage I and 136 lb for Stage II.

(e) Stage I Mixture Ratio Reorificing. Changing the Titan II engine target mixture ratios on acceptance test from 1.93 for Stage I and 1.80 for Stage II to approximately 1.95 and 1.84 would have allowed complete filling of both oxidizer and fuel tanks to ullage limits when the engines were operated in the anticipated GLV flight environment. However, as mixture ratio increased, specific impulse decreased for both stages. The net effect on payload was

$$\frac{\approx +32 \text{ lb payload}}{+0.01 \; \Delta \, MR_{I}} \quad \text{and} \quad \frac{\approx 0 \text{ lb payload}}{+0.01 \; \Delta \, MR_{II}}$$

Some of the other areas investigated were engine effects such as heat transfer and combustion stability, possible mission changes, and the impact of other potential performance improvement items such as further reduced minimum ullages and constant temperature propellants. As a result of these studies, the Stage II engine mixture ratio change was eliminated because there was no payload advantage, and the Stage I engine target mixture ratio was changed to 1.945, effective for GLV 4. This resulted in a payload improvement of 50 lb.

(3) Flight Improvements

(a) Flight Test Propulsion Performance Biases. Titan II and GLV engine performance data were monitored throughout the Gemini Program. By May of 1965, sufficient data had been accumulated to indicate that significant changes in the form of biases were likely to occur between acceptance test and flight. This analysis included the results of 10 Stage I flights and 16 Stage II flights. For GT-4 through GT-10, the biases indicated by the analysis were included in preflight trajectory and performance predictions. The biases were +8,200 lb Stage I thrust, +1.7 sec Stage I specific impulse, and +900 lb Stage II thrust. A Stage II specific impulse bias was not indicated by the statistical analysis. The payload capability gain for these biases was approximately +93 lb.

(b) Reoptimization of Pitch Program. When the Stage I thrust bias of 8,200 lb and specific impulse bias of 1.7 sec discussed above were incorporated into GLV-4 preflight predictions, the added efficiency of Stage I resulted in over-lofting of the Stage I trajectory. This was disadvantageous for two reasons. First, high-dispersed trajectories could result in pitch look-angles which exceeded the existing allowable limits. Second, over-lofting was inefficient from the standpoint of causing excessive gravity losses and Stage II pitch maneuvering. Because of these considerations, a new pitch program was developed for GLV-4. Use of the new pitch program, which eliminated the over-lofting resulted in an improvement in GLV-4 payload capability amounting to +55 lb.

The new pitch program developed for GLV-4 was later used for all succeeding launch vehicles. This was because the same or similar engine performance biases were expected for the succeeding vehicles. Therefore, the payload capability of GLV-5 through 12 was also increased by approximately 55 lb through adoption of the new pitch program. Variations in the actual commanded vehicle pitching rates were caused only by variations due to launch vehicle hardware; that is, to TARS package and power supply variations.

(4) Non-Implemented Improvement Studies

Although considerable performance improvements were adopted for the GLV, a number of studies were made during the course of the program which initially appeared attractive, but were later eliminated for technical or high-cost reasons. The most promising are summarized in Table II. D-7 and are discussed in the paragraphs below.

The use of a lower insertion altitude was studied in some detail, since this offered a tradeoff of 25 pounds of payload capability for every nautical mile decrease in insertion altitude. Studies by both the Martin Company and Aerospace Corporation indicated that the technique was feasible, but would require a depressed Stage I trajectory which approached the aerodynamic heating constraints of the GLV. At the lower insertion altitudes, there was a degradation in spacecraft insertion accuracy (due to more atmospheric noise at the lower elevation angles at guidance cutoff), as well as a requirement for the spacecraft to use its OAMS to make up the difference in perigee altitude in achieving rendezvous. The net effect was a payload improvement of 185 pounds for a reduction in insertion altitude from 87 n mi to 80 n mi. The technique was disapproved by NASA in October 1965, since at that time the payload margins relative to the predicted spacecraft weights were all positive and the decrease in insertion accuracies was deemed undesirable.

Another technique which the Martin Company investigated was the use of air-conditioned blankets around the Stage I and Stage II propellant tanks to maintain the propellants at a constant cold temperature (37^{o}). Since a 5-hour launch window was considered in the loading criteria, this would gain approximately 200 pounds of payload over the existing propellant conditioning and loading techniques. This proposal was rejected on the basis of feasibility, since a detailed design layout and release mechanism for the blankets had not been established, and the development time was such that only the last few vehicles in the program could benefit from such a scheme. There was also some question as to the adequacy of the thermal heat transfer in the blanket/air-conditioning system. As it turned out, NASA's redirection of the launch window to 1.5 hours reduced the relative payload gain for the constant temperature technique, and the inclusion of reduced enthalpy in the specific impulse calculations of engine performance at low temperatures made the constant temperature propellant scheme appear even less attractive.

A third technique for achieving additional payload margin was to reduce holddown time prior to GLV liftoff. The analysis indicated that a reduction of 0.7 second would result in a payload gain of 45 pounds. Although this gave sufficient time to detect engine start transient failures, analysis of the GLV structure indicated that vibrations from the engine start were possibly not sufficiently damped, and the proposal was rejected on this basis.

Reduced minimum ullages were adopted early in the program, and studies were later funded to consider still further ullage reductions to the following values: Stage I oxidizer 17 ft^3; Stage I fuel 15 ft^3; Stage II oxidizer 32 ft^3; and Stage II fuel 13 ft^3. These could have yielded an additional 180 pounds of

Table II-D-7. Non-Implemented Payload Improvements

Proposed Payload Improvement	Payload Gain (pounds)	Reason for Rejection
Lower insertion altitudes	185	Reduced insertion accuracy
Constant temperature propellants	200	Feasibility not established
Reduced hold-down time	45	Ignition transient not damped in GLV structure
Further reduced minimum ullages	180	Flight safety
Stage II Tank stretch, Stage I engine skirt extension	400	Major hardware development/ High cost/Late schedule

payload capability (140 lb for Stage I, 40 lb for Stage II). Limited engine start tests would have been required. Structural analysis showed that there were no problems as far as the vehicle itself was concerned under normal operation, but a major redesign of MDS systems would be required to prevent tank failure on the launch stand in the event of autogeneous pressurization failures on Stage I before liftoff. These technical problems were sufficient to cause rejection of this improvement item.

A number of other performance improvement techniques were considered, such as increasing the length of the Stage II tanks and increasing the expansion ratio of the Stage I engine nozzle. Although these could yield payload improvements on the order of 400 pounds, they entailed considerable redesign, test, and costs, and would only be available near the end of the program. Thus the cost per pound of payload improvement was extremely high, and these proposals were rejected on that basis.

(5) Mission-Dependent Performance Changes

In addition to keeping very close track of all launch vehicle parameters to correctly predict trajectory and payload capability, it was also necessary to take into account performance changes due to the Gemini mission differences and mission changes. For example, if the apogee were changed for a specific Gemini mission, it was necessary to adjust the predicted GLV payload capability in accordance with the apogee change. Similarly, if the launch azimuth and/or yaw steering were changed, the payload capability effects were computed and incorporated in the predicted launch vehicle capability.

Table II. D-8 summarizes several of the parameters which varied throughout the Gemini Program and which had to be taken into account in predicting trajectory and payload capability performance. The length of the launch window helped to determine the amount of propellants which could be loaded. The amount of these propellants, in turn, had a direct bearing upon launch vehicle payload capability. As the ullage limit window was decreased, the payload capability increased at rates as high as 40 lb/hr. Payload capability variation with launch azimuth was considerably more complex. Illustrated in Figure II. D-11 is this variation as it was predicted for GLV-12. Also included in the figure is the payload capability variation due to GLV yaw steering. Yaw steering was required to permit insertion of the Gemini spacecraft into the target plane for rendezvous missions.

For all orbital Gemini missions (except GT-2), the required insertion velocity resulted in an apogee that ranged between 130 and 190 n mi. The GLV payload capability varied at the rate of +2.7 lb/-1 n mi; thus, payload capability could vary over a wide range, depending upon the apogee selected. Likewise the amount of spacecraft separation velocity increment also had a direct effect upon launch vehicle payload capability. If the separation velocity increment imparted by the spacecraft were increased, the GLV payload capability would correspondingly increase at the rate of 1.5 lb for every 1 fps less required of the GLV, assuming a constant orbit apogee.

In addition to these changes for a given Gemini mission, it was also necessary for each of the rendezvous missions to determine payload capabilities for alternate missions defined by NASA. The alternate missions were those which would be attempted if the primary mission could not be completed. Aerospace Corporation and associate contractors were provided information regarding the planned alternate mission or missions. Based upon this information, payload capability estimates were developed for the nominal launch time and for succeeding launch times which would occur in the event of a launch delay. As it turned out, there were two alternate missions flown in the Gemini Program. They were GT-6A and GT-9A. These are discussed elsewhere in this report.

T

Gemini Mission	Description	Launch Requirements* Launch Window hr	Launch Azimuth deg	Orbit Inclination deg
GT-1	Unmanned Orbital, No Recovery, No Spacecraft Separation	3.5	72	32.54
GT-2	Unmanned Ballistic, Maximum Heating Rate	3.5	105	- -
GT-3	Manned, 3-Orbit, Battery Power	3.5	72	32.54
GT-4	Manned, 4-Day, Battery Power	2.5	72	32.54
GT-5	Manned, 8-Day, Active REP Exercise, Fuel Cells	3.5	72	32.54
VII	Manned, 14-Day, Fuel Cells	2.0 payload, 2.5 ullage	83.6	28.87
VI-A	Manned, 1-Day, Non-Docking Rendezvous with GT-7 S/C, Battery Power	2.0 payload, 2.5 ullage	Variable, 80 to 105, 81.4 actual	28.87 nominal, 28.89 actual
VIII	Manned, 3-Day, ATDA Rendezvous, Fuel Cells	2.5 payload, 4.5 ullage	Variable, 80 to 105, 99.9 actual	28.87 nominal, 28.87 actual
IX-A	Manned, 3-Day, Agena Rendezvous, Fuel Cells	2.5 payload, 4.5 ullage	Variable, 80 to 105, 87.4 actual	28.87 nominal, 28.87 actual
X	Manned, 3-Day, Agena Rendezvous, Fuel Cells	0.0 payload, 1.5 ullage	Variable, 80 to 105, 98.8 actual	28.87 nominal, 28.86 actual
XI	Manned, 3-Day, Agena Rendezvous, Fuel Cells	0.0 payload, 1.5 ullage	Variable, 80 to 105, 99.9 actual	28.87 nominal, 28.85 actual
XII	Manned, 4-Day, Agena Rendezvous, Fuel Cells	0.0 payload, 1.5 Ullage	Variable, 80 to 105	28.87 nominal

* The values identified as "nominal" are the valu
upon the achieved target orbit. Achieved GLV l
** After addition of spacecraft separation ΔV.

able II. D-8. Mission Requirements

Insertion Requirements*					
Perigee/Apogee** n mi	Geocentric Radius int ft	Inertial Velocity ft/sec	Inertial Flight Path Angle deg	GLV Objectives	GLV Results
87/161	21,438,500 ≡ 87 n mi	25,766	0.00	Structures, Exit Heating, Guidance Qualification	Successful
87 apogee at spacecraft release	Same	25,736	-2.28	Separation Subsystem Qualification	Successful
87/130	Same	25,699	0.00	Performance in Meeting Insertion Requirements	Successful
87/158	Same	25,756	Same	Performance in Meeting Insertion Requirements	Successful
87/190	Same	25,807	Same	Performance to Spacecraft Insertion, Joint GT/GAATV Countdown Compatibility	Successful
87/183	Same	25,804	Same	Meet Insertion Requirements	Successful
87/146 nominal	Same	25,730 nominal, 25,730 actual	Same	Meet Insertion and Rendezvous Launch Requirements	Successful
87/146 nominal	Same	25,730 nominal, 25,728 actual	Same	Meet Insertion Requirements, Joint GT/GAATV Countdown	Successful
87/146 nominal	Same	25,725 nominal, 25,723 actual	Same	Meet Insertion Requirements, Joint GT/GAATDA Countdown	Successful
87/146 nominal	Same	25,720 nominal, 25,719 actual	Same	Meet Insertion Requirements, Joint GT/GAATV Countdown	Successful
87/151 nominal	Same	25,720 nominal, 25,714 actual	Same	Meet Insertion Requirements, Joint GT/GAATV Countdown	Successful
87/151 nominal	Same	25,730 nominal	Same	Meet Insertion Requirements, Joint GT/GAATV Countdown	Successful

es used prior to target insertion. The "actual" values shown are the required values based
unch and insertion parameters are not shown in this table.

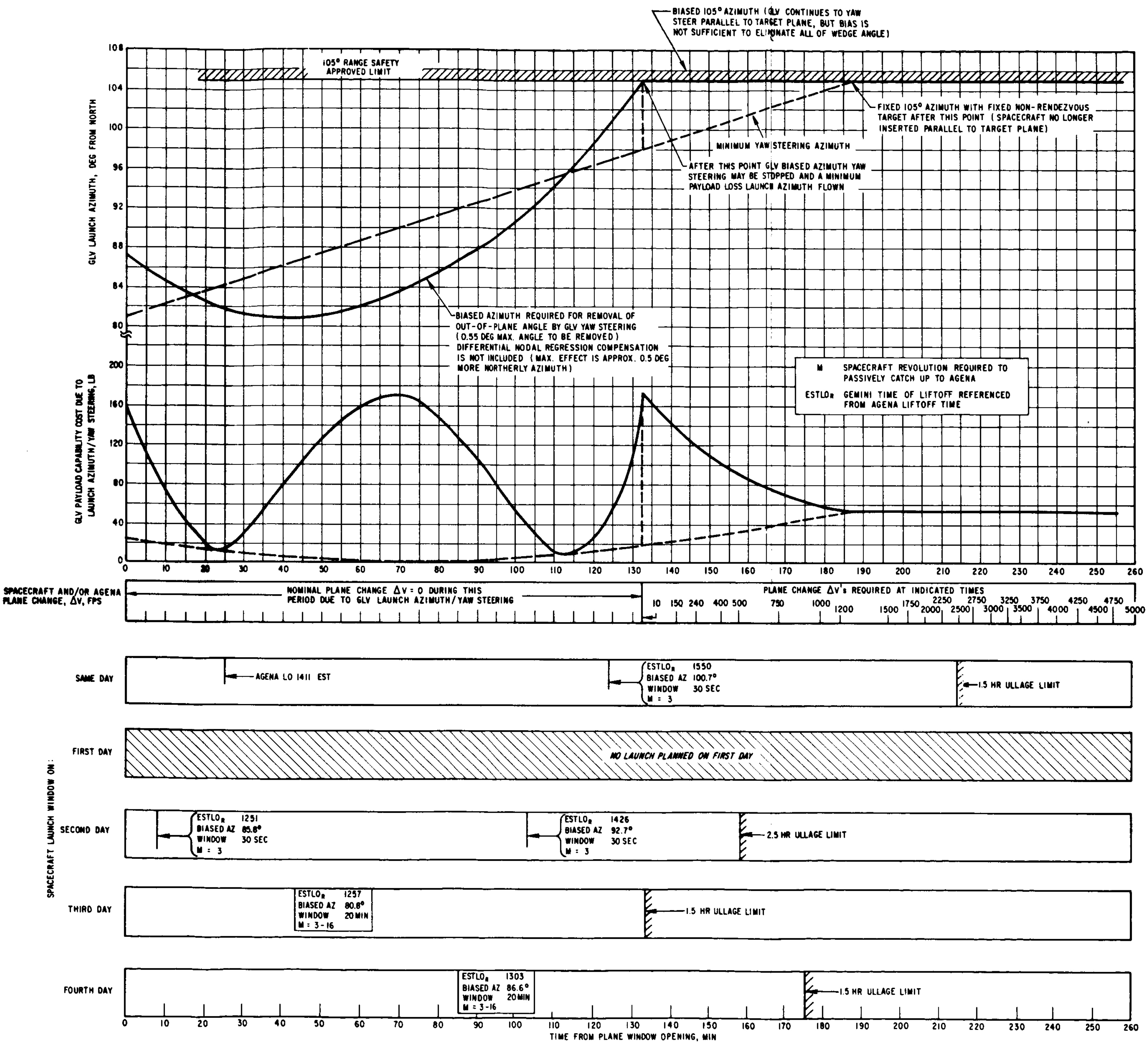

Figure II. D-11. Launch Windows for GT-12, Mission Gemini XII

c. GLV Flight Test Performance

(1) Propulsion Performance

Extreme attention was devoted throughout the Gemini Program by Martin, Aerospace, and AGC to obtaining accurate preflight predictions and post-flight analyses of vehicle propulsion performance. This area was of great importance because the GLV payload capability and trajectory performance were highly dependent on the propulsion parameters of mixture ratio (the major contributor to propellant outage), specific impulse and thrust for both stages.

Before the flight of GT-1, two propulsion performance computer programs were developed at Aerospace. Both were similar to programs used by TRW on the Titan II and Atlas programs. The first was a non-linear influence coefficient model that predicted propulsion parameters as functions of propellant pressures and temperatures, axial acceleration, and ambient pressure. This program operated closed loop in the N-Stage trajectory program. The second was a post-flight reconstruction program (used in conjunction with the N-Stage simulation) that calculated propulsion values that best matched flight data, such as acceleration (internal and external sources) and propellant consumption, on a "least-squares" basis. The program was identifed as the "Best Estimate of Engine Performance" (BEEP). Outputs were (1) flight integrated average propulsion parameters and (2) engine operation at "Standard Inlet Conditions". From this information differences in engine operation from predicted operation could be charged to changes in inlet conditions, to the engine operating differently than it did during acceptance ground tests, and/or to unknown sources.

To add confidence and continuity to the use of these programs, a contract was given to TRW Systems to perform propulsion analysis activities before the flight of GT-1 and preflight predictions and post-flight propulsion reconstructions on GT-1 and GT-2.

Before each flight, Martin, AGC, and Aerospace made independent propulsion performance predictions. These were compared and differences resolved. The major item analyzed was engine mixture ratio prediction since there was no closed loop propellant utilization system on board the GLV to correct for propellant consumption errors.

Anticipated engine performance dispersions were developed from Titan II engine acceptance test data. These were adjusted and biases incorporated as new analyses, statistical techniques, and flight results were applied.

Table II. D-9 contains the Aerospace BEEP summary of propulsion performance results for the 12 GLV's. The same set of biases was used in all the preflight predictions. Brief comments on the results are as follows:

Stage I thrust averaged approximately 0.5% below predicted, indicating that the bias should be reduced from +8200 lb to +6000 lb. Dispersions were tighter than theoretical.

Stage I specific impulse averaged slightly more than 0.1% above predicted. This was satisfactory and supported the +1.7 seconds bias that was added to the acceptance test value before making preflight predictions. Dispersions were a small amount larger than the theoretical allowance; however, no individual flight specific impulse fell outside the theoretical tolerance level.

There was an interesting relationship concerning specific impulse results on both stages that could be applied to improve predictions for future Titan II and Titan III family launches. It was noted by Aerospace (after GT-10 flew) that the specific impulse flight results on a stage tended to be close to a constant value, regardless of the acceptance test reported level. Thus, specific impulse flight biases and dispersions were smaller if the average of flight-derived specific impulses from previous flights were used for the next flight specific impulse prediction rather than the biased acceptance test level.

Table II. D-9. GLV Propulsion Performance Summary[1]

Vehicle		Flight Difference from Predicted, %[2]									
		Thrust		Specific Impulse		Mixture Ratio		Oxidizer Flowrate		Fuel Flowrate	
		S.I.C.[3]	Flt. Avg.	S.I.C.	Flt. Avg.	S.I.C.	Flt. Avg.	S.I.C.	Flt. Avg.	S.I.C.	Flt. Avg.
STAGE I	GT-1	-1.30	-1.49	-0.25	-0.25	0.66	0.39	-0.83	-1.10	-1.48	-1.49
	2	-0.23	-0.55	-0.09	0.09	0.29	-0.69	-0.04	-0.88	-0.32	-0.20
	3	-0.51	0.35	-0.01	0.21	-0.44	-0.38	-0.61	-0.08	-0.21	0.54
	4	-0.61	-0.25	0.24	0.35	-1.13	-1.20	-1.23	-1.00	-0.10	0.20
	5	0.02	-0.18	-0.11	-0.12	1.09	0.79	0.50	0.23	-0.59	-0.56
	6	-0.96	-0.68	0.02	0.13	1.12	0.69	-0.61	-0.60	-1.70	-1.28
	7	-0.12	-0.18	0.31	0.43	-0.58	-0.60	-0.62	-0.46	-0.04	0.15
	8	-1.01	-0.97	-0.24	-0.16	-0.58	-0.79	-0.96	-1.07	-0.39	-0.28
	9	-0.02	-0.26	0.63	0.59	-1.91	-1.37	-1.29	-1.30	0.63	0.08
	10	-0.55	-0.46	0.25	0.30	-1.92	-1.85	-1.46	-1.36	0.47	0.51
	11	-0.37	-0.72	0.27	0.25	-1.46	-1.56	-1.12	-1.49	0.34	0.07
	12	-0.36	-0.68	0.43	-0.11	-1.41	-0.49	-1.26	-0.73	0.15	-0.27
Avg.		-0.50	-0.48	+0.12	+0.14	-0.52	-0.59	-0.80	-0.82	-0.27	-0.21
3S		±1.24	±1.49	±0.83	±0.78	±3.29	±2.58	±1.71	±1.56	±2.15	±1.91
3σ, 75% confidence		±1.55	±1.86	±1.04	±0.97	±4.12	±3.22	±2.13	±1.95	±2.68	±2.38
				1→8:	Avg	+0.05	-0.22				
					3S	±2.56	±2.24				
Theoretical		--	0±2.4	--	0±0.61	0±1.38[4]	0±1.54[5]	--	--	--	--
STAGE II	GT-1	-1.42	-2.04	0.16	0.31	0.92	-0.19	-1.26	-2.60	-2.15	-2.42
	2	0.83	0.46	0.14	0.19	-1.71	-2.09	0.07	-0.43	1.81	1.64
	3	0.75	0.73	0.50	0.52	-0.49	-0.35	0.06	0.09	0.56	0.44
	4	0.92	0.23	0.39	0.35	-0.79	-0.39	0.24	0.60	1.04	0.95
	5	1.53	1.57	-0.07	-0.08	-0.31	0.04	1.49	1.66	1.80	1.63
	6	-0.42	-0.82	-0.35	-0.30	0.51	0.19	0.11	-0.45	-0.40	-0.64
	7	0.60	0.26	-0.38	-0.45	-0.26	0.28	0.89	0.81	1.15	0.52
	8	-0.15	-0.38	0.04	-0.04	-0.28	0.35	-0.29	-0.22	-0.01	-0.57
	9	-0.47	-0.17	0.14	0.09	-1.60	-1.33	-1.18	-0.74	0.43	0.60
	10	-0.08	-0.04	0.65	0.64	-1.54	-1.60	-1.27	-1.24	0.28	0.37
	11	-0.04	0.12	-0.23	-0.26	-1.32	-1.17	-0.29	-0.45	1.04	1.14
	12	-1.77	-1.35	0.09	0.40	-1.95	-1.20	-2.54	-2.18	-0.61	-0.99
Avg		+0.02	-0.12	+0.09	+0.11	-0.74	-0.62	-0.33	-0.40	+0.41	+0.22
3S		±2.91	±2.87	±0.96	±1.03	±2.73	±2.46	±3.26	±3.61	±3.35	±3.56
3σ, 75% confidence		±3.63	±3.58	±1.20	±1.29	±3.42	±3.07	±4.07	±4.51	±4.18	±4.44
				1 → 8:	Avg	-0.30	-0.27				
					3S	±2.38	±2.36				
Theoretical		--	0±3.2	--	0±0.73	0±2.28	0±2.52	--	--	--	--

NOTES:
(1) Data from Aerospace BEEP final reconstructions.
(2) Biases added to acceptance test thrust, specific impulse, and mixture ratio were +8200 lb, +1.7 sec, and 0 for Stage I; and +900 lb, 0 sec, and 0 for Stage II.
(3) Standard Inlet Conditions (S.I.C.)
(4) Value changed to ±2.43% for GT-11 and GT-12.
(5) Value changed to ±2.52% for GT-11 and GT-12.

Stage I mixture ratio on the first eight flights revealed less than a +0.1% bias at Standard Inlet Conditions. This was quite satisfactory. Dispersions at Standard Inlet Conditions were much larger than theoretical--approximately ±2.5% (3S) versus the acceptance test repeatability of ±1.38%. Titan II R & D flights had a repeatability of +0.1 ±2.2% so the GLV was compatible with previous experience. However, it was expected that the R & D results would be an outer bound. It must be pointed out that the BEEP flight mixture ratio repeatability encompasses not only engine hardware repeatability but also any unknown effects such as cavitation and simulation model inaccuracies. The negative mixture ratio shifts encountered on the last four flights averaged approximately -1.7%. After the occurrence of large negative shifts in mixture ratio on both stages of GLV's 9 and 10 with no cause identified, the Stage I mixture ratio (and unknown) repeatability value was increased from ±1.38% to ±2.43% in performance capability and propellant loading calculations for GLV-11. This gave propellant outage dispersions that were compatible with those encountered on the first 10 flights. After both stages on GLV-11 experienced the negative shifts, it was decided to also incorporate a -0.5% bias in flight mixture ratio in the preflight propellant loading calculations for GLV-12 to better optimize payload capability in the event of some negative shift. The shifts occurred again on GLV-12. See Section II. E. 3. h for details on the analysis of causes for the shifts.

Stage II mixture ratio had a -0.3% bias from predicted on GLV's 1 through 8, but dispersions at Standard Inlet Conditions were compatible with theoretical tolerances--approximately ±2.4% versus the acceptance test repeatability of ±2.28%. As on Stage I, the BEEP flight mixture ratio repeatability encompasses unknown effects in addition to hardware repeatability. For information, the Titan II R & D flights had a repeatability of -0.4 ±2.5%. Thus, there were no surprises and prediction techniques were considered adequate. No action was taken to remove the -0.3% bias since it contributed to smaller than nominal outages and, correspondingly, to above nominal payload capability. The negative mixture ratio shifts encountered on the last four flights averaged approximately -1.6%. After the shifts on GLV's 9 and 10, no action was taken to eliminate or partially compensate for an occurrence on GLV-11 since the outages that occurred were still smaller than nominal. After the shift on GLV-11, M/B completed a study of flight engine pressure data which indicated that the engine simulation model influence curves for the effect of oxidizer pump inlet pressure on mixture ratio should be changed. The change would eliminate approximately a -0.4% bias in mixture ratio on all flights. AGC concurred with the M/B analysis and a bias of -0.4% was incorporated in the propellant loading for GLV-12.

Stage II thrust averaged very close to predicted and verified the +900 lb bias added to acceptance test. Dispersions were within the theoretical allowance.

Stage II specific impulse averaged about 0.1% above predicted and thus was quite satisfactory. Dispersions were slightly larger than theoretical, but, as on Stage I, no individual flight specific impulse fell outside the theoretical tolerance band.

Stage I and Stage II propellant flowrates results are provided since they are the major parameters that are analyzed when searching for causes of shifts in thrust, specific impulse, and mixture ratio.

(2) Propellant Outage Analysis

Propellant outages* for Stage I and Stage II were the two largest factors in payload capability dispersion allowances (see Table II.D-13). Table II.D-10 provides a summary of the outage results on

*Outage in the amount of usable fuel (or oxidizer) that could not be properly consumed because all the usable oxidizer (or fuel) was exhausted.

Table II-D-10. GLV Normalized[1] Flight Outages, Lb. Fuel

Mission	Stage I[2]	Stage II[2]
GT-1	1799	49
2	1163	- 18 (32 ox)
3	644	274
4	3	184
5	1379	230
6	1545	320
7	725	302
8	189	328
9	- 88 (75 ox)	24
10	- 552 (901 ox)	24
11	- 612 (1007 ox)	117
12	- 319 (486 ox)	- 69 (122 ox)
1→12:		
AVERAGE	489 (MEAN 826)	147 (MEAN 167)
3S	±2521 ($3S_\lambda$ = ±2.87%)	±430 ($3S_\lambda$ = ±1.95%)
3σ, 75% Conf.	±3075 ($3\sigma_\lambda$ = ±3.50%)	±537 ($3\sigma_\lambda$ = ±2.44%)
1→8 WITHOUT MR BIAS:		
AVERAGE	488 (MEAN 729)	122 (MEAN 172)
3S	±1938 ($3S_\lambda$ = ±2.21%)	±389 ($3S_\lambda$ = ±1.77%)
THEORETICAL:		
AVERAGE	580[3] (MEAN 850, MAX (99%) 2600)	136[3] (MEAN 209, MAX (99%) 622)
3σ	±2285 ($3\sigma_\lambda$[4] = ±2.6%)	±570 ($3\sigma_\lambda$ = ±2.59%)

NOTES: (1) Actual outages were adjusted by normalizing for

a. Present engine model
b. Mixture ratio optimized for T-O
c. On-time liftoff
d. Present capability for controlling and predicting propellant temperatures
e. $3\sigma_\lambda$ = ±2.6% for Stage I and ±2.59% for Stage II
f. Incorporation in Stage I of a zero outage region along with increased ΔV during shutdown for fuel exhaustion
g. Mixture ratio biases to acceptance test of -0.5% for Stage I and -0.4% for Stage II

(2) The "trapped plus shutdown" propellants used in the outage calculations were as follows:

	Stage I		Stage II
	Ox Exhaustion	Simultaneous or Fuel Exhaustion	Command Shutdown with RESS
Ox, lb	128	128	96
F, lb	507	464	74

(3) Fuel bias

(4) $3\sigma_\lambda$ (three sigma lambda) is the measure of overall dispersion in outage. The theoretical values include the RSS of 1.38% for Stage I and 2.28% for Stage II for engine mixture ratio repeatability. Because of the mixture ratio shift problems on GLV's 9 and 10, an additional repeatability factor of 2.0% was RSS'd into Stage I to increase $3\sigma_\lambda$ to the flight-observed value at that time of approximately 2.6%.

the GLV flights compared to theoretical allowances. For a correct comparison of outages on the flights, these outages all have been normalized to the values that would have resulted if the preflight-predicted and post-flight-achieved ground rules and calculations had been the same. The configuration for GLV-12 was used. Comments on biases and dispersions for both stages are as follows:

Stage I: Outage dispersions are seen to be worse than theoretical when all flights are included in the sample. These results were strongly influenced by the negative mixture ratio shifts experienced on the last four flights. The mixture ratio bias of -0.5 percent that was incorporated in the GLV-12 preflight predictions reduced oxidizer outages by 847 lb or increased fuel outages by 443 lb. This bias was not necessary nor desirable for GLV's 1 through 8 and was too small for GLV's 9 through 12. This is shown by the results for 1 through 8 alone in Table II. D-10. Thus, if the mixture ratio shift problem could be eliminated, much better outage bias and dispersion performance would result on Stage I.

Stage II: Outage dispersions were better than theoretical, even though a negative mixture ratio shift problem was also experienced on this stage of GLV's 9 through 12. Dispersion allowances should not be reduced, however, since the major contributor--engine (and unknown) mixture ratio repeatability--was not smaller than the theoretical value of ±2.28 percent. From Table II. D-9, the Standard Inlet Condition mixture ratio results were -0.3% ±2.4% 3S, for GLV's 1 through 8; and -0.7% ±2.7% 3S, for GLV's 1 through 12. As for biases, if the mixture shift problem were eliminated, the need for a -0.3% bias (reduction of oxidizer outage by 115 lb and increase of fuel outage by 65 lb) is seen. This is justified from outage as well as mixture ratio results once a correction for being 1.2° too low in oxidizer equivalent temperature on the average on GLV's 1 through 8 is incorporated in the outages.

(3) Development of Post-Flight Reconstruction Program

Early in the Gemini Program it became clear that there was a need for a new computer program that would help in post-flight analysis. The need was to reconstruct the achieved trajectory by varying vehicle parameters to determine in the process the vehicle perturbations experienced in flight. These perturbations would be useful in explaining the trajectory and payload capability performance of the launch vehicle, and they would be useful in estimating performance dispersions for future launch vehicles. In June of 1964, Aerospace Corporation embarked upon an effort to develop a suitable trajectory and vehicle parameter reconstruction program. This was called the Post-Flight Reconstruction Program (PFRP).

The fundamental idea of the PFRP was to make a weighted least-squares-fit to the available trajectory and vehicle performance data. The former consisted of tracking data, such as Mod III and MISTRAM radar data, while the latter consisted of chamber pressure data, inertial guidance system data, etc. Since the mathematical models utilized in the reconstruction process were in many instances non-linear, it was necessary to iterate upon the desired vehicle parameters to obtain the weighted least-squares-fit. The weighting factors were based upon the a priori dispersions attributed to each of the parameters. A finite difference method was used for obtaining the approximate partial derivatives which were necessary in the iterative solution.

The Post-Flight Reconstruction Program was combined with a detailed vehicle simulation known as the Modular Vehicle Simulation (MVS), plus a data processor. The relationships among these three components are shown in Figure II. D-12.

Development of PFRP progressed to the point where it could be used for post-flight analysis in time for GT-5. PFRP was used for all subsequent vehicles, with improvements in models being incorporated as soon as they were made.

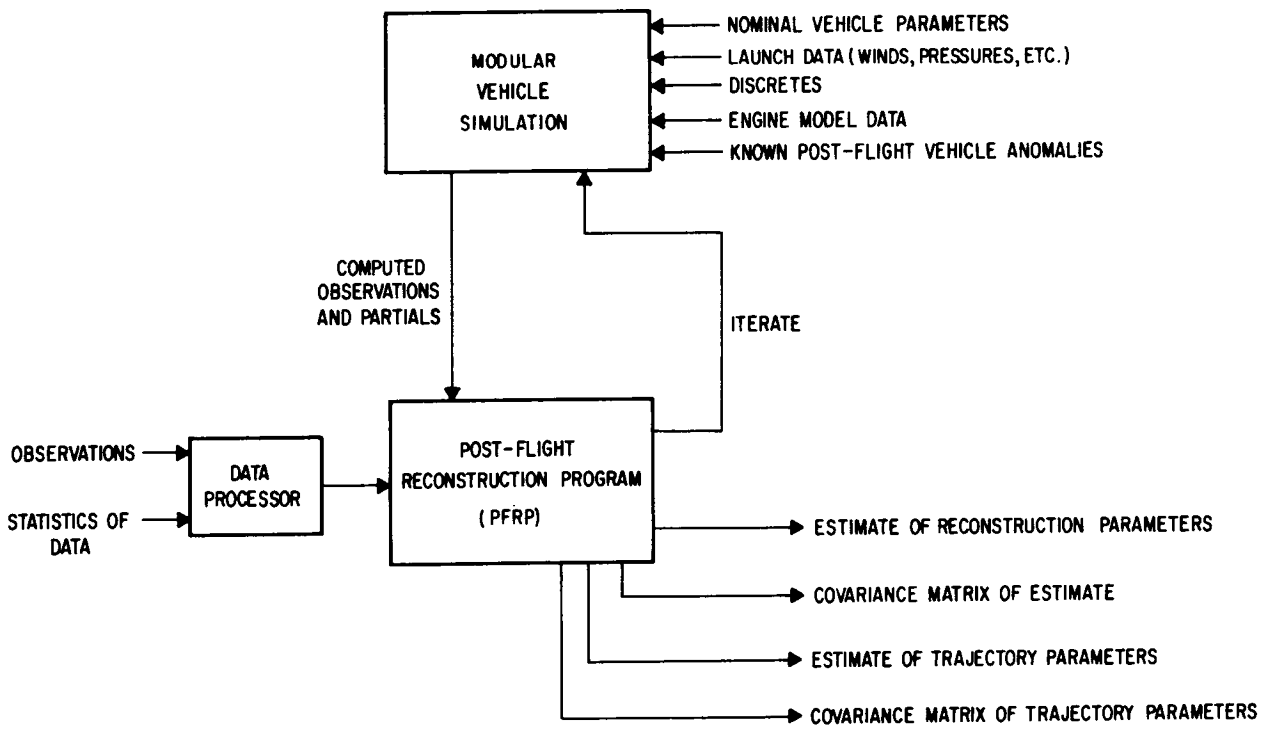

Figure II. D-12. Post-Flight Reconstruction Program (PFRP) Inputs and Outputs

PFRP was used extensively in the Gemini Program to determine vehicle parameters such as pitch programmer error, pitch gyro drift, engine pitch misalignment, pitch stand misalignment, and several other parameters useful in explaining the achieved trajectory and payload capability performance. An important area in which PFRP was useful was in the determination of engine performance parameters such as thrust, specific impulse, and mixture ratio. The engine performance parameters derived by PFRP were always compared with those derived by the Aerospace BEEP program. Comparison would occasionally result in improvements to either or both of the engine performance analyses.

PFRP results were also used extensively to define achieved trajectory parameters to make possible a quantitative comparison with the predicted nominal trajectory.

(4) Non-Normalized Performance Summary

Table II. D-11 contains the GLV nominal and minimum payload capability margins that were predicted by Aerospace on F-1 Day or in the Post-Flight Reference Trajectory and the margins between achieved payload capabilities and spacecraft weights. A summary of the achieved Stage II burning time margins is also provided in Table II. D-12. The achieved payload capability margins and burning time margins are of interest because they represent the actual margins that were available for each of the Gemini launches. The achieved payload capability margins are all adjusted to be compatible with the desired inertial velocity at Stage II engine cutoff, SECO. The adjustments for SECO velocity were made at the rate of +1.5 lb payload for each 1 fps less required of the GLV. The maximum adjustment used in the table is +26 lb for GT-1. The intent of this adjustment is to permit a direct comparison between the predicted and achieved payload margins.

It will be observed in Table II. D-11 that the achieved payload capability margins during the Gemini Program ranged from a minimum of 374 lb (GT-5) to a maximum of 1, 396 lb (GT-3). These margins correspond to burning time margins of 1.08 sec and 3.93 sec, respectively.

Although the statistical distribution of GLV payload capabilities is not normal, due to the non-normal nature of one of the dominant determining factors--outage, it is interesting to handle the achieved differences in Table II. D-11 as though they were normal. The mean payload capability difference from nominal is 88 lb. The standard deviation for the twelve Gemini flights is 135 lb. At a 75% confidence level, the estimate of the 3-sigma dispersion is 497 lb. At 90% and 95% confidence levels, the estimates of 3-sigma dispersion are 584 and 648 lb, respectively. These statistical parameters can be compared with those included in the paragraph below in which a normalized payload performance summary is discussed.

(5) Normalized Performance Summary

Table II. D-13 is a normalized performance summary for all of the Gemini Launch Vehicles. Both the predicted and achieved payload capability margins have been normalized to be compatible with payload capability prediction methods developed throughout the Gemini Program and utilized for performance predictions for GLV-12. The intent of the normalization is to show comparisons between predicted and achieved payload capability margins when both of these are derived using the most accurate methods known.

The predicted payload margins in Table II. D-13 are normalized so that all are based upon the following:

1. Stage I and Stage II engine thrust biases added to acceptance test thrusts to match least-squares-fits to GLV flight thrust histories (different from vehicle to vehicle)
2. Stage I and Stage II specific impulse biases added to match least-squares-fits to GLV flight specific impulse histories (different from vehicle to vehicle)

Table II-D-11. Actual (Non-Normalized) Predicted and Achieved GLV Payload Capability Margins

Mission	Predicted Payload Capability Margin, lb		Achieved Payload Capability Margin, lb	Difference, Achieved minus Predicted Nominal, lb
	Minimum	Nominal		
GT-1	+508	+1017	+1171	+154
GT-2	+366	+1025	+1066	+41
GT-3	+577	+1199	+1396	+197
GT-4	-62	+593	+767	+174
GT-5	-135	+526	+374	-152
GT-7	+69	+709	+786	+77
GT-6A	+265	+891	+778	-113
GT-8	-162	+493	+471	-22
GT-9A	-282	+372	+638	+266
GT-10	-217	+416	+571	+155
GT-11	-175	+497	+528	+31
GT-12	-51	+619	+869	+250
				Mean +85
				1S ±135

Table II-D-12. Predicted and Achieved GLV Burning Time Margins

Mission	Predicted Nominal Burning Time Margin, sec	Achieved Burning Time Margin, sec	Difference, Achieved minus Predicted, sec
GT-1	2.95	3.22	+0.27
GT-2	2.72	2.90	+0.18
GT-3	3.40	3.93	+0.53
GT-4	1.64	2.12	+0.48
GT-5	1.47	1.08	-0.39
GT-7	1.95	2.18	+0.23
GT-6A	2.44	2.34	-0.10
GT-8	1.34	1.28	-0.06
GT-9A	1.03	1.85	+0.82
GT-10	1.16	1.65	+0.49
GT-11	1.39	1.48	+0.09
GT-12	1.73	2.53	+0.80

Table II-D-13. Normalized Predicted and Achieved Payload Capability Margins

Mission	Normalized Predicted Payload Capability Margin, lb		Normalized Achieved Payload Capability Margin, lb	Difference, Achieved minus Predicted Nominal, lb
	Minimum	Nominal		
GT-1	+574	+1244	+1277	+33
GT-2	+463	+1133	+1178	+45
GT-3	+706	+1376	+1254	-122
GT-4	-63	+607	+790	+183
GT-5	-74	+596	+241	-355
GT-7	+47	+717	+591	-126
GT-6A	+315	+985	+631	-354
GT-8	-153	+517	+408	-109
GT-9A	-242	+428	+808	+380
GT-10	-159	+511	+792	+281
GT-11	-147	+523	+538	+15
GT-12	-51	+619	+869	+250
				Mean +10 1S ±237

3. Predicted pitch program bias effects included (different from vehicle to vehicle)
4. 42 lb Stage II skirt ablation effects included rather than the effects due to the 20 lb ablation used prior to GT-8
5. Payload contribution due to nominal seasonal winds included (different from vehicle to vehicle)
6. The values of Redundant Engine Shutdown System propellant consumption and total impulse used for GT-8 through GT-12 rather than the interim values used for GT-3 through GT-7.
7. Final values of Stage I tailoff and Stage II ignition propellant consumption and total impulse rather than the values previously used (for vehicles flown after deletion of shutdown level sensor operation)
8. Mixture ratio optimization at the beginning of the launch window rather than the times previously used
9. Stage I and Stage II mixture ratio biases included to match the GLV flight mixture ratio and outage history
10. Stage I outage indicator increased to the 2.6% value currently used
11. Propellant temperature control and monitoring capability results for GT-1 and GT-2 adjusted to be compatible with those for subsequent missions
12. Incorporation of a zero outage region on Stage I
13. Incorporation of increased Stage I velocity gain, from shutdown to separation, for fuel exhaustion compared to the velocity gain for the normal oxidizer exhaustion shutdown.

The achieved payload capability margins are all based upon attaining the desired SECO velocity rather than the actual SECO velocity, and upon usage of items 6) through 13) above.

The normalized predicted and achieved payload margins, as well as the spacecraft launch weights, are shown in Figure II. D-10.

The normalized achieved payload margin differences from nominal are plotted against probability in Figure II. D-9. As may be seen in the figure, the normalized points compare well with the prediction curves. The prediction curves are those which were developed for use with GLV-11 and GLV-12. It may be seen in Table II. D-13 that the normalized achieved payload margin differences from nominal ranged from -355 lb (GT-5) to +380 lb (GT-9A). This spread may be compared with the predicted dispersion from mean (53% probability) to minimum (99.4% probability) for GLV-12 of -670 lb. It may also be observed that, of the twelve missions, payload margin surpluses were achieved in seven, while payload margin deficits (below predicted nominal) occurred in five.

The list of normalizing factors provided above is not exhaustive of the normalizing factors which are possible. For example, further normalization could be achieved by using consistent engine model simulations for all launch vehicles. However, the overall effects of the other normalizing factors are believed to be relatively small and would not materially change the results shown in this section.

If the payload margin differences from nominal in Table II. D-13 are handled as though their distribution were Gaussian, an interesting comparison can be made with the statistical data for the non-normalized payload margin data in the preceding section of this report. The mean payload margin difference from nominal is +10 lb. The standard deviation for the twelve Gemini flights is 237 lb. At the 75% confidence level, the estimated 3-sigma (99.87% probability) dispersion assuming Gaussian distribution, is <u>867</u> lb. At the 90% and 95% confidence levels, the estimated 3-sigma dispersions are 1020 and 1131 lb, respectively. These data are significantly different from the data developed from the non-normalized payload margin summary. The normalized data agree better with the predicted distribution curves shown in Figure II. D-9, as would be expected. From these curves, the disperson from the nominal capability (53% probability) for the nominal propellant temperature case to the 99.87% probability minimum point on the -2 deg propellant temperature case is: <u>840</u> lb.

(6) Trajectory Analysis

Post-flight analysis of each Gemini Launch Vehicle trajectory was conducted to define the reasons for deviations from nominal and to determine thereby what changes should be made in predictions for subsequent vehicles. Trajectory analysis in one form or another was carried out by the Martin Company, Aerojet-General Corporation, General Electric Company, as well as Aerospace Corporation and NASA.

The Aerospace effort was accomplished largely by the Post-Flight Reconstruction Program (PFRP) which is described elsewhere in this report. This reconstruction program not only reconstructed the shape of the achieved launch vehicle trajectory, but also determined the vehicle parameter values which accounted for the shape of the trajectory. The PFRP results were compared with the nominal trajectory to determine deviations from the nominal.

Actually, there were 3 "nominal" trajectory sources to be considered. There was first the Aerospace Corporation STO trajectory for non-rendezvous missions and the Post-Flight Reference Trajectory (PFRT) for each rendezvous mission. This latter trajectory was run utilizing the guidance constants actually set into the guidance computer for the flight. The Post-Flight Reference Trajectory, therefore, was based upon the actual final velocity, launch azimuth, and Stage II yaw steering required of the launch vehicle, rather than upon preflight nominal values. Occasionally, slight changes were also made to vehicle parameters to improve the trajectory prediction.

The second source of "nominal" trajectory data was the Range Safety trajectories. These were prepared prior to the flight by the Martin Company and were always somewhat different from the Post-Flight Reference Trajectory. However, they were the trajectories which were used in real time by Range Safety personnel at ETR to determine whether the launch vehicle trajectory was deviating significantly from the nominal so as to be a hazard to human life along the projected flight path. The third source of nominal trajectory data, differing from both the PFRT and Range Safety trajectories, was the Slow Malfunction Detection trajectories. These, prepared by the Martin Company prior to each flight, were used in real time to determine if vehicle malfunctions were occurring which would endanger the crew. For post-flight analysis, gross trajectory information was obtained by comparing the achieved trajectory with the Range Safety and Slow Malfunction Detection charts, while detailed comparisons were made between the Post-Flight Reference Trajectory and the trajectory determined by the Post-Flight Reconstruction Program.

The trajectory analysis for each vehicle showed that each achieved trajectory was relatively close to the predicted nominal. There were few times when Range Safety and Slow Malfunction Detection chart monitors had cause for alarm. There were no instances of imminent destructs by Range Safety or aborts by Slow Malfunction Detection personnel. Each launch vehicle trajectory satisfactorily permitted attainment of launch vehicle flight test objectives. There were no instances of trajectory parameters at significant times in the trajectory being outside of their predicted 3-sigma limits. The significant times in the trajectories referred to are maximum dynamic pressure, BECO, SECO, and SECO +20 sec.

A table comparing significant parameters at liftoff and each of the significant points in the trajectory was provided in each Aerospace final post-flight report. Some of the achieved differences from nominal are provided in this report in Table II. D-14. Qualitative explanations for the trajectory time history differences from nominal were also provided in terms of deviations due to vehicle and environmental parameters. Specifically, the effects of winds, engine thrust, loaded propellant weights, pitch program bias, pitch gyro drift, and other parameters were commonly included in the discussion.

Table II. D

Time in Trajectory	Parameter	GT-1	GT-2	GT-3
Max q	Time from Liftoff, sec	+4	-7	0
	Dynamic Pressure, psf	+49	-62	-40
	Mach Number	+0.15	-0.26	+0.08
	Altitude, ft	-	-5206	+3267
	Angle of Attack, deg	-	-	-
	Relative Velocity, ft/sec	-	-	-
BECO	Time from Liftoff, sec	+0.77	-1.82	-1.70
	Inertial Velocity, ft/sec	-58	+154	+95
	Inertial Flight Path Angle, deg	-0.42	+1.24	+1.73
	Geocentric Range, ft	-580	+12,742	+14,637
	Yaw Velocity, ft/sec	-12.0	-56.1	-109.5
	Out-of-Plane Distance, ft	-	-	-
	Radar Range, ft	-	+11,370	+8218
	Radar Elevation Angle, deg	-	+1.11	+1.84
	Ground Range, n mi	-	+1.4	+1.9
SECO	Time from Liftoff, sec	+3.44	-4.34	-4.68
	Inertial Velocity, ft/sec	-	+5	-9
	Inertial Flight Path Angle, deg	-	0.00	-0.01
	Geocentric Range, ft	-	+75	-309
	Yaw Velocity, ft/sec	-	+0.1	-5.7
	Out-of-Plane Distance, ft	-	-	-
	Radar Range, ft	-	-28,120	-47,727
	Radar Elevation Angle, deg	-	+0.15	+0.25
	Ground Range, n mi	-	-4.0	-7.9
SECO +20 sec (Nominal Insertion Time)	Time from Liftoff, sec	+3.44	-4.34	-4.68
	Inertial Velocity, ft/sec	+18.5	+7.5	-16.9
	Inertial Flight Path Angle, deg	-0.125	-0.01	+0.04
	Geocentric Range, ft	-2424	-1104	+376
	Yaw Velocity, ft/sec	-79.5	-4.5	-4.5
	Out-of-Plane Distance, ft	-	-	-
	Radar Range, ft	-	-28,162	-48,110
	Radar Elevation Angle, deg	-	+0.12	+0.21
	Ground Range, n mi	-	-4.0	-6.9

NOTES: 1. The paramete
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-14. Flight Test Trajectory Parameters Dispersion

Observed Parameter Difference from Predicted Nominals									Sample Mean ±3σ
GT-4	GT-5	GT-6A	GT-7	GT-8	GT-9A	GT-10	GT-11	GT-12	
-1	+2	+5	-4	+5	-1	+3	+1	+1	+1 ±14
-24	+18	-15	-51	-66	-36	+10	0	-19	-20 ±129
-0.04	+0.19	+0.19	-0.22	+0.16	-0.07	+0.15	+0.04	-0.03	+0.03 ±0.58
-56	+4823	+4629	-4213	+5547	-643	+3706	+1498	+303	+1241 ±13,742
-0.98	+0.03	+0.36	-2.28	+0.67	-2.26	+0.06	-0.26	-0.27	-0.55 ±4.26
-33	+144	+143	-199	+143	-71	+115	+25	-26	+27 ±475
-0.94	-1.29	+0.83	+0.25	+0.91	+0.84	-0.01	+0.40	+1.37	-0.03 ±4.18
-78	-153	-6	+112	-39	-27	-131	-129	+8	-21 ±373
+1.11	+0.90	-0.39	-0.13	-0.13	+0.05	-0.43	+0.11	-0.26	+0.28 ±2.83
+6413	+5159	-3322	+1832	-2089	+1704	-4303	-165	-913	+2593 ±22,752
-48.2	-111.0	-189.3	-49.0	-114.1	-6.1	-38.9	-17.0	-21.1	-64.4 ±207.7
-	-	-5328	-1533	-5608	-844	-224	-2978	-971	-2498 ±9124
-3050	-5070	+4747	+9914	+4955	+3828	-4024	-1074	+5382	+3200 ±21,703
+1.41	+1.44	-0.99	-0.62	-0.89	-0.09	-0.38	+0.14	-0.72	+0.20 ±4.00
-1.1	-1.7	+1.2	+1.6	+1.1	+0.5	-0.3	-0.2	+1.1	+0.5 ±4.5
-2.07	-3.65	+2.11	-1.60	+1.95	+0.80	+0.83	+0.62	+3.78	-0.23 ±11.10
-12	-3	-9	-5	+9	-14	-3	-4	-11	-5 ±27
+0.08	0.00	+0.02	+0.06	-0.03	-0.13	+0.01	-0.01	0.00	-0.01 ±0.20
+195	-447	+232	-384	-314	-364	-1470	-59	-672	-320 ±1809
-1.7	+1.1	-1.2	-14.6	-9.4	-15.3	-2.6	+5.1	+1.1	-3.9 ±25.2
-	-	-24,690	-5029	-16,994	+406	-4004	+2044	-2676	-7278 ±40,765
-40,526	-62,962	+24,821	+19,177	+24,658	+3983	+819	-15,778	+43,160	-7136 ±130,856
+0.20	+0.32	-0.12	-0.09	-0.13	-0.03	-0.04	+0.07	-0.23	+0.03 ±0.68
-7.1	-9.6	+3.8	+3.1	+3.8	+0.3	-0.1	-2.9	+6.9	-1.2 ±20.8
-2.07	-3.65	+2.11	-1.60	+1.95	+0.80	+0.83	+0.62	+3.78	-0.23 ±11.10
-13.0	-2.1	-11.6	-11.0	+9.0	-16.0	-7.0	-8.3	-15.0	See Section II.D.1.b regarding these four parameters.
+0.065	-0.01	+0.05	+0.05	-0.04	-0.11	+0.008	+0.008	+0.0002	
+1252	-583	+476	+758	-264	-2127	+61	-202	-894	
0.0	+3.4	-6.7	-12.9	-11.4	+0.5	-6.0	+5.1	+4.1	
-	-	-24,841	-5020	-17,266	+333	-3519	+2087	-1647	-7125 ±41,604
-40,694	-63,190	+24,650	+19,169	+25,019	+4371	+1933	-15,973	+43,009	-7089 ±131,394
+0.25	+0.25	-0.10	-0.07	-0.11	+0.04	+0.04	+0.06	-0.18	+0.05 ±0.57
-6.5	-9.6	+3.4	+3.1	+3.8	+0.3	-0.1	-3.0	+6.7	-1.2 ±19.9

r differences are not normalized in any manner. If all of the predicted parameter values had been
; a consistent trajectory prediction technique, many of the earlier predicted values would have been
wever, as noted elsewhere in this report, prediction techniques evolved throughout the Gemini Program.

values are evaluated at the 75% confidence level. They apply to predictions for the entire Gemini
do not apply to a single launch.

SECTION D. REFERENCES

1. General Electric Mod III Gemini Look Angle Restraints, TOR-469(5126-27)-3, S.H. Fabian, Aerospace Corporation, June 1965.

SECTION II. D

DEFINITIONS

AGE	Aerojet-General Corporation
BEEP	Best Estimate of Engine Performance
DEU	Data Exchange Unit
GAATV	Gemini Atlas Agena Target Vehicle
GE	General Electric
GLV	Gemini Launch Vehicle
GT	Gemini Titan (Launch Vehicle Mission)
IBM	International Business Machines
IGS	Inertial Guidance System
M/B	Martin Baltimore
MCCH	Mission Control Center - Houston
MOPS	Missile Operation System
MR	Mixture Ratio
MVS	Modular Vehicle Simulation
NASA	National Aeronautics and Space Administration
PAFB	Patrick Air Force Base
PFRP	Post Flight Reconstruction Program
PFRT	Post Flight Reference Trajectory
RandD	Research and Development
RESS	Redundant Engine Shutdown System
RGS	Radio Guidance System
RSS	Root Sum Squared
SECO	Sustainer Engine Cut-Off
TARS	Three-Axis Reference System

E. SPECIAL STUDY EFFORTS

1. GENERAL

In every program, a time comes when the design must be frozen to the point of committing to "cut" hardware. The design at that point represents an acceptable compromise of a configuration that is adequate to do the required job. It therefore represents an acceptable starting point. History has shown, however, that no design is ever truly frozen. New requirements arise, unknown design deficiencies manifest themselves, and improvement areas are discovered. All of these potentially or actually effect design modifications. Unfortunately, they are the areas that can neither be accurately forecast nor conveniently scheduled, and they represent an expenditure of considerable effort which takes place during the actual flight program.

Although some of the special study items have been previously mentioned under their respective subsystem descriptions, a number of the more significant ones that occurred during the program are presented in this section. Those covered are by no means all that were evaluated, but they do represent the ones most worthy of note from a magnitude or program impact standpoint.

These special studies have been divided into three rather general categories: those that could have influenced flight safety; those that resulted in performance improvement; and those which come under the category of system development.

2. SAFETY OF FLIGHT

a. Escape Environment

In order to assess flight crew hazard, in the event that ejection seat abort was followed by launch vehicle breakup and explosion, an investigation of the resulting environment was undertaken. The anticipated environment was modeled in terms of the four principal hazards: heat, overpressure, fragmentation, and toxicity. Due to the entirely uncontrolled conditions and the limited data available from real occurrences, a controlled test program was funded by NASA and implemented by Aerojet-General Corporation (AGC), Downey. These tests were also undertaken to establish a relationship between the conditions resulting from cryogenic and hypergolic explosions. This technique permitted utilization of Titan II half-scale test data, as well as actual data from Vanguard, Navaho, Thor, and Atlas failures to establish the boundary conditions of the escape environment[1].

b. Switchover/Switchback Studies

With the incorporation of a redundant flight control system, a detailed system evaluation was conducted to re-assess the vehicle airframe, switchover logic, and sensor limits. The evaluation indicated that the initial selection of sensor limits, structural safety factor, and switchover logic did not result in optimum switchover capability. The concept and evaluations that were conducted are described in references at the end of this section.[2,3,4]

It became apparent that a switchover during Stage I flight from a loss of hydraulic pressure would result in flying on the backup flight control system throughout Stage II flight. This could have resulted in discarding a good, reliable primary flight control system during Stage II flight. To alleviate this situation, the capability of switching back to the primary system was incorporated. It was planned that switchback would only be actuated in the event the switchover was initiated by loss of hydraulic pressure and would be activated between staging and guidance enable.

The switchover flight loads during the high maximum dynamic pressure region were found to be in excess of the structural design criteria. Consequently, the concept was optimized by selecting the sensor limits that maximized crew safety. A corresponding hardware change was made to reduce the angular rate switch settings. The structural load carrying capability was re-evaluated in the light of probability considerations which resulted in a reduced factor of safety for switchover from 1.25 to 1.10.

c. Rate Switch Setting Change

The Gemini rate switch settings were constrained on the high side by vehicle strength considerations and on the low side by normal nonmalfunction rate excursions during flight. The higher the switch setting, the greater the chance of vehicle breakup prior to switchover. The lower the switch setting, the greater the chance of inadvertent switchover from normally incurred vehicle turning rates.

The choice of settings for the malfunction detection system (MDS) rate switches was originally made at a time when the structural strength capabilities and malfunction switchover loads were not completely defined. Also, over-conservative estimates of the maximum normal vehicle rates and actuator excursions were used since there was only limited Titan II flight test data to give confidence in the selected settings.

The additional accumulation of statistical data from the Titan II flight program permitted a reduction in the low rate pitch and yaw switch settings prior to the first manned flight, which increased crew safety protection by decreasing the time to sensing a malfunction condition.

d. Redundant Engine Shutdown System (RESS)

A redundant method of shutting down the Stage II engine was a crew safety requirement of the Gemini program to minimize the possibility of capsule overspeed caused by malfunction of the primary shutdown system. The RESS was developed under the Augmented Engine Improvement Program. The

RESS was a squib-actuated valve located in the oxidizer bootstrap line, which upon actuation terminated oxidizer flow to the gas generator. This system was operated simultaneously with the thrust-chamber valve/pressure sequencing valve over-ride (TCV/PSVOR) utilizing the same command. The system was ground-tested, formally qualified and successfully used on all the manned flights.

Some of the salient design objectives achieved by this backup system were:

1. Both cutoff systems were actuated by any cutoff command.
2. Failure of either cutoff system would not disable the other.
3. A component failure in the redundant system would not result in engine cutoff.
4. The backup system was isolated from the primary system to assure complete redundancy.
5. Operation of the backup system did not result in engine damage.
6. Tailoff characteristics were repeatable.

e. Compartment Five (Stage I Engine) Protection

The AGC review for the Abort Panel of engine incidents, such as hot gas leaks and fuel leaks resulting in engine compartment fires, indicated a requirement for study of fire detection and/or protection techniques. AGC accepted the task of modeling a Compartment 5 thermal environment comparable to that experienced during the flight of Missile N-20 when the entire first-stage engine compartment was afire.[5] Martin Company, Baltimore (M-B) investigated pre-liftoff and inflight fire detection techniques and insulating materials.[6, 7] The study concluded that detection techniques were too expensive and complex; however, adequate insulating materials were available. To minimize the weight penalty, the addition of insulation was recommended only for those Compartment 5 components where proper operation was required for successful abort.[8] Effectivity of this modification was GLV-3 and subsequent vehicles per ECP GLV AJ 149R and ECP GLV-MM 373R1. (See Section VII.)

f. Re-Evaluation of Mode I Abort (Seat Ejection)[9, 10]

Because of the unsatisfactory results obtained from seat-ejection tests performed at velocities in excess of Mach 1, NASA directed the launch vehicle contractor team to develop alternate abort techniques in the time region between 50 and 100 seconds of flight. The primary failure modes requiring abort were determined to be single-engine thrust failure or partial thrust failure. By limiting the investigation to the primary failure modes, it was possible to determine the relative performance of the various escape techniques proposed. Because of the large aerodynamic loads during the 50-to 100-second period, immediate Mode II abort (retro abort) was not feasible.

The various techniques investigated were as follows:

- Feasibility of controlled flight on one engine only
- Low altitude staging
- Delayed Mode II Abort

Single-engine flight control was eliminated after the study showed that even to obtain a short period of vehicle stability, a significant autopilot redesign was necessary. Low-altitude staging was eliminated since the second-stage autopilot was unable to control the large initial rates resulting from first-stage thrust failure.

The technique that finally evolved from this study was called the Delayed Mode II Abort Procedure. This procedure was to "shut down the GLV when failure was recognized (on engine failure events, shut down immediately after overrate light illuminates). Delay for short 2-second count after overrate light and retro salvo off the GLV." To further enhance the safety of this technique, high strength NAS bolts were installed in the manufacturing splice at Station 923 in place of the existing AN bolts in GLV-9 and subsequent vehicles, to force the potential structural breakup to occur between stages rather than between the first-stage fuel and oxidizer tanks.

g. Hold and Shutdown Parameters

The master operations control system (MOCS) at Complex 19, ETR, used the Radio Corporation of America/MILGO Mod III sequencer. This sequencer had the capability of activating automatic hold-fire circuits ON and OFF for any period between T-35 minutes (start of automatic countdown) and T+10 minutes. Circuits activated prior to T-0 were called "holds" and the detection of any malfunction resulted in stopping the countdown until the malfunction was corrected and the count resumed by the test conductor. Circuits activated after T-0 and prior to 0.2 seconds before liftoff were called shutdowns and the detection of any malfunction resulted in stopping the launch and sending a shutdown signal to both Stage I and Stage II engines. The hold-down time span was nominally 3.2 seconds for all vehicles.

Hold parameters consisted of those pressure, level, and position parameters required to ensure that all GLV systems were operating correctly and that the engines could be started for launch. Certain parameters that would have required complicated automatic monitoring devices were monitored by launch personnel with manual switch hold capabilities.

Shutdown parameters were a minimum number of critical system monitors that had to be operating correctly to allow a successful launch. A shutdown signal aborted the launch. Since these parameters were extremely critical, the following criteria were used to develop the parameters:

1. Primary and secondary systems were operating properly at liftoff.
2. Pilot safety requirements were provided.
3. The possibility of a catastrophic failure during the first stage flight was minimized.
4. Maximum assurance of mission success existed at liftoff.

A study effort was initiated to determine what parameters would be used for holds and for shutdowns and implemented for the launch of GLV-1. Some 29 shutdown parameters were ultimately used.

After the launch of GLV-1, the shutdown parameters were re-evaluated and some deleted. Subsequent re-evaluations and studies resulted in selecting 9 shutdown parameters that were used for GLV-4 and the remainder of the program. These are documented in M-B specification 424-1430-002.

This method of automatic system monitoring with the capability of automatically shutting the engines down operated successfully on two vehicles. GLV-2 was shut down as a result of a primary system hydraulic failure and GLV-6 was shut down when an electrical plug disconnected prematurely. Both malfunctions occurred after engine ignition and prior to 0.2 seconds before liftoff.

h. POGO at Low Fuel Tank Pressure

In order for the vehicle structure to support steady state acceleration loads at booster engine cutoff (BECO), the Stage I fuel tank pressure abort limit was set at 8 psia for the period from 105 seconds to BECO. As the onset of POGO during this time period was quite likely at low fuel tank pressure, the 8 psia criteria was questionable from a crew safety standpoint. It therefore became necessary to evaluate the effect POGO oscillations might have on the vehicle structure and crew while the pressure was decaying to the 8 psi abort limit.

Considerable effort was expended in defining a workable approach for attaining a more realistic abort constraint; however, because of the lack of POGO data for the time period and condition in question, the Martin and Aerospace analytical models were not sufficiently complete to yield reliable data. Since positive definition of a serious problem was not possible, the Abort Panel was reluctant to recommend revision of the abort procedures. All agencies were therefore informed of the flight risk and the effort was terminated.

i. Titan II Anomalies

At mid-point in the Gemini flight Program a re-examination of Titan II, Titan III, and GLV anomalies was conducted to determine whether there was a trend or commonality in the problems. [11] As a normal procedure, each Titan flight had been examined in detail during the program for any impact on the Gemini hardware. This special task team effort of re-examination was conducted jointly by SSD, Aerospace Corporation, Martin-Marietta Corporation, and Aeroject-General Corporation, to ensure, in light of additional knowledge and experience accumulated subsequent to the flights, that the original problem resolutions were still valid. Additionally, it assured the program that some problem or trend had not been overlooked which would degrade the confidence in the Gemini Launch Vehicle.

The investigation was initiated by each contractor and customer task team forming a separate list of anomalies by flight. This list was obtained from contractor post-flight reports, raw and cal comp data, and flight working group reports. The flight results of the operational training missions were obtained from the Air Force Logistics Command, Norton Air Force Base, Ogden Air Materiel Area reports, Strategic Air Command reports, and raw data. The listings were then compared to develop an assimilated single list. The final list of approximately 300 anomalies were then categorized and assigned to the respective vehicle subsystems for system review.

The study indicated that the Titan Family problem review had been very thorough. There were no trends; however, there were some problems occurring randomly such as Stage II engine skirt break-up, post-Stage I venting, minor engine leaks, and Stage II engine "green man". These were nuisance items and did not appear to be flight safety items. There were a few unexplained engine flight problems that were investigated in detail. The conclusion from these special investigations indicated that no hardware changes were required.

j. Staging Event (Tank Venting)

High-speed long-range camera coverage of GLV-10 indicated an apparent anomaly consisting of a large orange-red cloud appearing from Stage I shortly after staging. A detailed review of the films revealed that the oxidizer tank vented approximately 1.2 seconds after Stage II ignition. A study of Stage II telemetry data revealed no indication of this event. Stage I telemetry was inoperative at this time having been disabled 0.7 seconds earlier. A thorough study of all possible causes of the tank rupture was conducted by Martin and Aerospace. This study isolated the three most probable causes of tank rupture (and tank venting). These were: 1) Stage I turning after separation and resulting in the Stage II engine subassembly exhaust impingement and burn-through of oxidizer tank barrel, 2) Breaking of the ablative coating on the oxidizer tank dome, due to dome flexing caused by high local pressures at Stage II engine start, resulting in dome overheating and subsequent structural failure, and 3) Dome or tank barrel penetration by transportation section debris.

A review of the staging films on Titan II flights revealed similar occurrences on seven Titan II flights. The same anomaly occurred on GT-12; however, this occurrence was followed by the apparent rupture of the Stage I fuel tank and the breakup of Stage I just forward of the Martin/Aerojet interface.

The results of the study and a review of all available Titan II and GLV flight data showed no detrimental effect on mission success or crew safety due to this event.

3. PERFORMANCE IMPROVEMENT

a. Improved Gemini Launch Vehicle

Early in the program, a review of the Mercury Spacecraft weight growth history indicated that if the Gemini Spacecraft followed the same trend, it would exceed the performance capability of the Titan II ballistic missile as the launch vehicle. As a result, the Martin Company conducted a study concerned with Gemini Launch Vehicle performance improvement.[12] The study proposed various techniques, some of which were major hardware changes to the propellant tanks and the Stage I engine.

The items considered were decreased hold-down time, further reduced Stage I and II tank ullage volumes, change to a lightweight telemetry system, redesign of Stage II engine support cone, removal of 15-percent repair factor criteria for Stage II tanks, reshaping of Stage I trajectory, a lower insertion altitude, Stage II tank stretch, and the increase of Stage I engine expansion ration to 12/1. These items were studied to a level sufficient to make cost estimates, availability schedules, and magnitude of payload gains. Because of the magnitude of change associated with each of the above, these items were not selected for incorporation. Instead, the program selected other means, such as shorter launch windows, increased engine specific impulse based on flight data, and NASA's choice of launching with less than 3 sigma payload margin, depending on S/C propulsion for insertion if necessary.

At the time of this study there were in process, however, many changes to the basic Titan II to gain payload performance capability. These were weight reduction by eliminating unused hardware, Stage I shutdown by propellant depletion, optimizing of the engine mixture ratio, use of cold propellants, reshaping of the trajectory by allowing a negative guidance look angle, and Stage I and II reduced ullage volumes.

b. GE Radio Guidance Negative Look-Angles

Performance improvement studies[13] by Aerospace and M-B indicated that a reshaping of Stage I trajectory would increase the payload capability by 60 lbs. The GLV Stage I trajectory was lofted above the optimal ascent trajectory to avoid the Stage II pitch zero look-angle constraint imposed by the Mod III airborne antenna. Aerospace and General Electric (GE) prepared an antenna test program to measure antenna patterns in the negative pitch look-angle regime. This was implemented by Technical Directive GE 2140-8 dated 3 November 1964. The pattern test data was measured at the GE Pittsfield, Massachusetts antenna range using a full-scale mock-up[14] and analyzed by Aerospace to determine trajectory design and loss of lock constraints.[15, 16]

c. Primary Antenna Redesign

The GLV primary antenna was redesigned from a 6-inch to a 4-inch slotted configuration to change the antenna pattern for better look-angle coverage. The antenna requirements study was accomplished in 1963 and documented by GE.[17] The prototype units were built and tested by GE utilizing a full-scale half-shell mock-up. The 4-inch antenna was flown on all Gemini launches.

d. Low Altitude Insertion

A low altitude insertion study indicated that it was desirable to extend the present Mod III Radio Guidance System noise model to lower than 5 degrees in elevation angle. Discussions with GE indicated that they had more available flight data that could be included which was not in the current model, and that the model could be extended to 2 degrees in elevation angle. This was accomplished in October 1965 and an updated report issued.[18]

e. Deletion of Propellant Level Sensors for Shutdown

A new propellant level sensor design was selected for GLV in place of the Titan II sensor and, for that reason, qualification and flight testing were required prior to usage in a close-loop circuit for engine shutdown.

Since the required flight testing on the new design sensors had not been accomplished prior to the GLV-1 flight, studies were conducted with respect to propellant depletion shutdown in lieu of level sensor shutdown to support the first flight. The studies indicated that there were no adverse staging or spacecraft separation problems evident from a propellant depletion shutdown. Engine combustion chamber damage from a fuel depletion shutdown was not deemed detrimental to mission success.

After the GLV-1 flight, sufficient flight experience was available to utilize the level sensors for engine shutdown with confidence; however, the additional payload margin available from propellant depletion made this a more attractive mode of shutdown. In addition, the risk of an early shutdown from level sensor failure outweighed the risk of a hazardous engine propellant depletion shutdown from a crew safety/mission success standpoint.

The Gemini propellant level sensors were used for instrumentation purposes only.

f. Cold Propellants[19, 20]

In order to increase vehicle performance, it was deemed desirable to load the vehicle with relatively cold propellants, i.e., from 35° to 40°F. Before implementing this procedure into the Gemini program, it was necessary to investigate engine durability and operation using the low temperature propellants. A series of hot-fire engine tests were conducted at AGC utilizing propellants conditioned to a temperature of from 35° to 40°F. Analysis of the data from these tests failed to reveal any significant departure from the nominal start transients or system performance associated with propellants at ambient temperatures. It was therefore concluded that utilization of the colder propellants would produce no deleterious effects upon engine performance, including the performance of the autogenous system.

g. Reduced Ullage[21]

Early in the Gemini Program, it became evident that projected payloads would probably require more efficient utilization of the available propellant tank volumes. The primary objective of the minimum-ullage * test program was to determine if reduced fuel and oxidizer tank ullages would have any adverse effects on engine performance. Secondly, it was of interest to note whether the engine starting characteristics differed significantly from those of Titan II, and if so, to determine the consequences of the differences. The "minimum ullages" were:

1.	Stage I Oxidizer	48 cu ft
2.	Stage I Fuel	39 cu ft
3.	Stage II Oxidizer	63 cu ft **
4.	Stage II Fuel	16 cu ft

The Martin Company analyzed the effects of reduced ullage on the pressurization system (propellant tank pressure) and the ability of the pressurization system to provide sufficient pressure, during the start sequence and under steady-state conditions, to satisfy the minimum net positive suction head (NPSH) requirements of the first-stage engines. In addition, the effects of reduced ullage upon the response characteristics/time of the pad malfunction detection system (PMDS) were also scrutinized. Although the major effect was that of reducing the warning time to failure, this reduction in margin was not sufficient to cause concern from a pilot safety standpoint. AGC, through its test program, substantiated the Martin findings.

*This is a misnomer since the actual minimum tank ullage was not evaluated.

**AGC test = 65 cu ft

Early in 1965, additional payload capability was sought through a further reduction in ullage, particularly Stage I ullage. An evaluation of a proposal to utilize maximum tank volume by reducing the ullage to the physical minima of 15 cu ft in the fuel tank and 17 cu ft in the oxidizer tank was undertaken. After considerable investigation by Martin and Aerospace it became evident that any further significant reduction in Stage I propellant tank would result in a high probability of tank implosion prior to the time that the PMDS could sense the impending catastrophic condition and initiate engine shutdown in the event of an autogenous system failure. Besides the fact that a certain number of engine tests would have been required to demonstrate satisfactory engine operation under the more severe conditions of true minimum ullage, the aforementioned risks to the safety of the crew created problems and uncertainties that necessitated rejecting this method for increasing payload capability.

h. Stage I and II Mixture Ratio Evaluation

Launch vehicle postflight evaluation indicated that on vehicles 9-12 there had been greater than predicted oxidizer outages. Data evaluation verified that both Stage I and II average mixture ratios were lower than predicted by a considerable amount. On Stage I, in particular, the magnitude of the mixture ratio shift exceeded the log-to-launch three-sigma value being utilized for trajectory analysis/payload capability dispersion. Even though the oxidizer outages were other than predicted, the overall launch vehicle capability had not been lowered.

A task team consisting of Aerospace Corporation, SSD, AGC, and M-B was established following the launch of GLV-9, the primary objective being to study this particular phenomenon and, if possible, make recommendations towards either correcting the cause or better predicting the anticipated effect on payload dispersion. A detailed investigation was conducted into the following general areas:

1. Vehicle propellant loading accuracies
2. Performance determination and trajectory evaluation
3. All facets of engine acceptance and data evaluation
4. Propellant characteristics

Results of the investigation of the first three items indicated the mixture ratio shift did exist and it was not the result of problems associated with propellant loading, performance determination, or basic engine calibration. A review of the flights of GLV 1-10 indicated a distinct difference in mixture ratio dispersion/outage on GLV groups 1-8 and 9-10. The first group did not indicate a definite bias and had dispersions within expected tolerances, while the second group indicated a definite bias with dispersions greater than those anticipated. As an expediency, the log-to-launch mixture ratio tolerances were increased for the flight of GLV-11 to better determine minimum payload capability.

Because the problem appeared to be common to both Stage I and II engines, the investigation centered about the propellants being used. This investigation initially indicated that the oxidizer (NTO) used for GLV's 1-8 differed in its age and vendor procurement from that utilized on the later series of flights. Computer evaluations indicated that the most probable cause of the mixture-ratio shift was some peculiarity in the oxidizer which manifested itself in a bi-phase flow, or reduced density of the oxidizer passing through the engine pump. Although no conclusions evolved from the detailed investigation into the oxidizer characteristics (including dissolved gases), the problem appeared to be associated with increased cavitation of the oxidizer pumps resulting in a lowered mixture ratio. This increased cavitation most probably resulted from dissolved gases coming out of solution at the pump inlet. This would result from a combination of gas content in the oxidizer and of suction conditions. Both the Stage I and II engines were acceptance-tested (balanced) at a considerably higher suction pressure than normal during flight. It was expected that this problem would concern the Titan II Weapon System and the first liquid stage of the Titan IIIB vehicles. The Titan IIIC vehicle has an additional augmented oxidizer tank pressurization system which apparently alleviates the problem. The final summary (including all generated data) results of the entire mixture ratio anomaly, as studied by the Gemini Directorate, was turned over to Titan II/Titan III for their review and action as required.

4. SYSTEM DEVELOPMENT

a. Interface Dynamic and Structural Analysis

The structural integrity of both Martin and McDonnell structure in the Spacecraft/Launch Vehicle interface area, under the influence of fluctuating local pressures caused by transonic buffeting, was a matter of concern because of lack of knowledge of the behavior of the higher frequency ring modes when excited by fluctuating pressures, and the catastrophic nature of failures in this area. Aerospace conducted an analysis [22] which indicated that substantial local stresses did result from local buffeting pressures. These values resulted in positive margins of safety on the M-B forward oxidizer tank skirt, but on the adapter, when coupled with the early McDonnell analysis, negative margins resulted. A more refined structural analysis run by McDonnel incorporating better phasing of internal pressure loads and airload body bending loads, as well as the results of a better dynamic analysis patterned after the Aerospace analysis, showed the necessary positive margins of safety.

b. Structural Integrity Program

On GT-2 and GT-3, special structural instrumentation was carried to assess the validity of the design loads used in structural design. The instrumentation consisted of four strain gages and a temperature sensor in compartment 2, two strain gages and a temperature sensor in the interstage structure, and four strain gages and a temperature sensor in compartment 5. All gages were mounted on continuous stringers. Data was successfully recovered from all sensors.

M-B, using conventional reduction techniques, was unable to obtain correlation because of the inability to define effective cross-sectional areas of the somewhat complex semimonocoque structure at the instrumentation stations. Aerospace, by using a calibration procedure with the known loads at pre-ignition and pre-BECO, was able to establish good correlation with pre-launch predictions and post-flight reconstructions, except for certain portions of the GT-3 flight where the wind data was uncertain (i. e., wind conditions were changing rapidly and the exact conditions during ascent are not known). The two gages in the interstage area proved to be useless, since a minimum of three gages per station were found to be required. Static and dynamic analysis of the resulting data fully validated the design loads and pre-launch prediction techniques used on the GLV. [23]

c. Evaluation of Defective Conduit Welds

Prior to the flight of GLV-2, visual inspection (later confirmed by x-ray evaluation) disclosed minute transverse cracks in the longitudinal conduit seam welds on a Stage II conduit. Inspection of other conduit assemblies revealed similar defects. Metallurgical examination of the defective welds revealed that the cracks were on the edge of the weld on the inner surface of the conduit, and did not go through to the opposite surface. Investigation of the manufacturing processes revealed that the cracks were caused by poor welding procedures during fabrication, prior to convoluting the conduit.

An environmental life test program was initiated to investigate the structural adequacy of the defective conduits, since they were installed in all vehicles up to and including GLV-8. The test program consisted of a 450-cycle pressurization test and a 7.5-minute random vibration test at 11.5 g's rms. After completion of the test program, the specimen was cut up and microscopically examined. These examinations, plus visual examination, x-ray examination, dye penetrant check, and helium vapor emission testing, showed no evidence of crack propagation. The environmental life testing program demonstrated the structural adequacy of the defective conduits then installed in vehicles GLV-1 through -8. The defective conduits were flown satisfactorily. New conduits, built under more stringent quality control and with improved weld procedures at the vendor level, were procured for GLV-9 through -12.

d. Malfunction Detection Thrust Chamber Pressure Switch (MDTCPS) Task Team

This team was organized by Aerospace Corporation during September 1964 to investigate pressure-switch diaphragm corrosion, repetitive switch failures due to current overload, and relocation of Stage II MDS switches from the hot gas to the fuel system. The team was composed of technical representatives from Martin-Baltimore and -Cape, Aerojet-Sacramento, SSD, and Aerospace. In the course of these meetings, other areas such as switch hot-fire requirements, MDTCPS shutdown circuitry, and the M-B/AGC interface specification 62-190A were also reviewed.

The MDTCPS task team was formally dissolved in June 1965 with the following concrete achievements to its credit:

1. Development of test methods and criteria for detection of diaphragm corrosion.
2. Determination of causes for MDTCPS current overload and revision of procedures to prevent recurrence.
3. Definition and review of testing necessary to approve Stage II pressure switch relocation.
4. Re-definition of pressure switch hot-fire requirements.[24]
5. Agreements for revision of Interface Specification 62-190A.
6. Definition of system requirements for the MDTCPS shutdown monitoring circuit.

e. Stage I Shutdown Oscillations

A review of GLV-9 flight data revealed vehicle oscillitory motions during Stage I shutdown in the frequency range from 16 to 50 cycles per second. While the condition had been noted on previous flights, it was more pronounced on GT-9 and was of sufficient magnitude, as recorded off vehicle rate gyro outputs, to initiate switchover.

Using frequency response data furnished by M-B on the Titan II rate gyros and Gemini rate switch package, special filters were constructed in the Aerospace laboratory to simulate the damping characteristics of the rate switch package (RSP). The rate gyro output data from GLV-9 was then played through the special filters and it was shown that the high amplitude, high frequency oscillations were damped to 50% of the nominal RSP switchover setting.

The special filter data was made a part of the normal Gemini flight data analysis for subsequent flights.

f. MDS Hold-Kill Parameters and Slow Malfunction

Considerable effort was directed toward establishing meaningful criteria for the parameters monitored after engine ignition, to assure proper vehicle systems status before liftoff. Primary and secondary hydraulic system pressures were used as one of the hold-kill parameters and the functions were monitored with pressure switches.

To establish pressure levels at which a malfunction could be assumed at the optimum parameter sampling time, a hydraulic-system malfunction analysis was made of all applicable Titan II and GLV flights, summarizing startup data under flight conditions. Tests were also run in the Martin-Baltimore airborne systems functional test stand (ASFTS) test area to look at engine pump pressure characteristics under simulated engine start conditions.

A philosophy was established that all secondary (backup) systems must be operable at vehicle liftoff. On this basis, it was decided to monitor only the secondary hydraulic-system pressure as a kill parameter after engine start, while both system pressures were monitored as hold parameters during countdown prior to engine start. If failure occurred after engine start, some primary system failure modes would result in a switchover from primary to secondary systems. This would automatically initiate engine shutdown, provided the launch vehicle holddown bolts had not been blown. The pressure-switch activation point for vehicle shutdown was set at 2500 ± 50 psia, well below pressure dips experienced on Titan family vehicles, and predicted by ASFTS test.

Hydraulic-system pressures were monitored as slow malfunction parameters by NASA personnel acting as booster monitors at the Mission Control Center in Houston. Pressure level envelopes for various stages of flight were supplied to NASA for use as guides in evaluating real-time data of booster performance. As a backup to the pressure data, it was recommended that system reservoir fluid level be monitored. Correlation data between pressure and reservoir levels along with expected reservoir levels at various stages of flight were supplied.

g. Range Safety Command Control Receivers

Early in the Gemini development program there was a concerted effort for weight saving. The development of a new light-weight solid-state range safety command receiver was a natural area for consideration, but funding was limited for this type of development. Concurrently, Titan III was in the process of developing a new "man-rated" range-safety receiver, primarily because the standard government-furnished equipment (GFE) vacuum-tube type receiver did not meet the environmental requirements.

An additional requirement imposed on the receiver was the intent to standardize and simplify the flight termination system for other SSD programs, with attendant savings in hardware, testing, and logistics. This new receiver development was made to order for Gemini, and subsequently the Gemini Program was able to utilize this development program. Gemini was able to participate in the bid and evaluation program of vendors' proposals for a new receiver, which was finally awarded to Advanced Communication Incorporated (ACI). The Gemini/Titan III receiver was basically a Model R-420 ACI Receiver which was developed for the White Sands Missile Range (WSMR) and built to their rigorous specifications. This receiver had already had an impressive record of performance on several programs at WSMR.

The development of a new command control receiver resulted in several benefits to the Gemini Program. The most significant was the incorporation of a time-delay circuit between the engine shutdown and the destruct commands. This was to assure the astronauts sufficient time to escape from the launch vehicle, if necessary. By means of connector coding, the delay circuit could be by-passed for unmanned flights.

The dual ACI receiver installation in Gemini saved about 20 pounds per vehicle over the old receiver installation which was used only on GLV-I. The 28-vdc power consumption was decreased by about 100 watts per vehicle. The new receiver employed a unique command tone detector which demonstrated a higher degree of protection against response to spurious signals. This added protection was especially important on Gemini, since the range safety command receivers had to share the same rf link with the Spacecraft Digital Command System. There were no significant problems in the development of the R-423A receiver for the Gemini/Titan III programs.

h. Electrical and Electronic Interference (EEI) Compatibility Testing

The EEI compatibility testing was accomplished by means of a comprehensive program involving the vertical test facility (VTF) at Baltimore and Complex 19 at ETR. Tests were first performed on the aerospace ground equipment (AGE) at both installations and then on the complete vehicle, during combined systems tests at VTF and Complex 19. A spacecraft simulator was utilized to effect the interface from the GLV to the space capsule.

Performance of all tests was planned for the first three vehicles. However, due to the achievement of an interference-free system after wiring and circuit corrections on GLV-1 and -2, further testing on GLV-3 was considered unnecessary. The equipment for EEI evaluation was maintained in operating condition, should extensive system and/or wiring changes be required. Such was not the case.

GLV-1 underwent tests at VTF covering all aspects of countdown and simulated flight in both primary and secondary modes, including switchover. As each out-of-tolerance condition appeared, it was corrected and the pertinent test repeated. Six tests were required to verify that interference levels were in tolerance, or caused no transients derogatory to systems operations. 58 points were monitored: 48 were launch vehicle circuits, 10 were AGE circuits. A summary of test results appears in LV-335-3.[25]

Similar tests were performed at Complex 19, using the AGE and similar EEI monitors. Measurements were taken for most points through breakout boxes and/or cables on the umbilical tower. The 57 monitor points comprised the following: 38 launch vehicle circuits, 8 spacecraft interface circuits, 3 AGE circuits, 8 power bus circuits.

The results of the EEI tests at ETR were considered to have met the requirements of MIL-E-6051 C as interpreted by the EEI Control Plan.[26] In particular, the GLV had a safety factor of 6 db or better above the level which would cause system degradation or malfunction.

Concern arose over the existence of high frequency transients having energy in the spectrum to 5 MHz. Such transients, appearing on the lines from the spacecraft inertial guidance system, could possibly affect the spacecraft computer operation or memory circuit. A series of tests were performed on GLV-1 at VTF, wherein magnetic tape recordings were made with a frequency response up to 1 MHz, in addition to the conventional lower frequency oscillograph system. The results were presented to NASA for evaluation since no specification existed for conducted EEI in this frequency range. Report LV 335-4a provides details of these tests.[27]

GLV-2 underwent essentially the same tests as those performed for GLV-1. However, procedures were changed somewhat and partial testing was performed to verify correction of several out-of-specification conditions. The testing at ETR was performed with the permanent type equipment installations which consisted of much of the original EEI amplifiers, oscillographs, cables, etc., in a thoroughly integrated system. The results, after necessary corrections and retests, satisfied the vehicle system requirements. The findings are reported in LV-335-6.[28]

In addition to the initially scheduled test aforementioned, post-mate EEI tests between the GLV and spacecraft were performed on GLV-2 and GLV-3 by McDonnell Aircraft personnel. As a result of the thorough testing of the launch vehicle, the post mate interface testing was accomplished with a minimum of difficulty. Some changes were incorporated in the spacecraft due to the evaluation of test results and no further action was required after GLV-3.

i. Complex 19 - Electromagnetic Incident Investigation

During the functional testing of GLV-2 at ETR, there was an occurrence which has been called alternately the "Electromagnetic Incident" or the "Lightning Strike Incident". A thunderstorm in the area had interrupted launch vehicle testing on the evening of 17 August 1964. Several observers reported that lightning struck on or near Complex 19, although no actual strike could be proved.

The vehicle was connected to the AGE through the regular umbilicals. The AGE power monitor system received out-of-tolerance alarm indications from three a-c monitoring points; no d-c monitoring point indicated over-voltage. The flight control system of the launch vehicle had been undergoing tests prior to the cessation caused by the approach of the storm.

The vehicle was thoroughly inspected and systems were tested. No physical markings of any kind were uncovered on the vehicle. However, there were failed components in the vehicle. Seven of eight malfunction detection system (MDS) pressure transducers showed calibration shift; the eighth had zero output. Also, the voltage regulator within the adapter package failed, thereby applying excessive current through transistors on six spin motor rotation detector (SMRD) circuit boards.[29] The failure in the adapter package was responsible for failure of solid state devices in AGE units CP 2858, CP 2859, CP 2631, and CP 2632. Other AGE units affected were a seven-inch drag sphere which had erratic output, and five open galvanometers in CP 0400.[30]

The nature of the failure indicated induced voltage in excess of 50 volts, but actual magnitude could not be determined, except indirectly by comparison with items that were not affected. Therefore, it became necessary to develop a hardware replacement and retest plan.

The salient features of the plan prescribed the replacement of all packages containing semiconductor elements and complete visual inspection of the vehicle for physical damage as well as the engines for corrosion. All tests were to be repeated as though the vehicle had just arrived at ETR. New calibrations were established for telemetry measurements. Special telemetry data acquisition runs were performed on each subsystem for comparison with data acquired during VTF testing. Similarly, the combined systems acceptance test results were compared with the CSAT data from VTF. Any data point falling within the last 20 percent of the tolerance band was to be questioned by M-C engineering representatives and resolved before proceeding to subsequent tests.

The actual test plan was rewritten as a Cape Test Operation,[31] and was successfully accomplished. The electrical harnessing, probably the most critical area, was subjected to special high voltage breakdown testing and passed all tests. The vehicle then underwent post-mate testing and was judged to be ready for manned spacecraft launch.

The details of schedules, retest sequences, and prior and post incident vehicle history are covered in a separate report.[32]

j. Recovery of Telemetry Data During Staging Blackout

Telemetry reception from the GLV was lost momentarily during booster staging as a result of the second-stage engine flame pattern. The blackout was caused by the highly ionized gases traveling forward and surrounding the telemeter antenna located in the area between the propellant tanks. The blackout generally occurred at approximately 0.95 seconds after second-stage ignition (91 FS1) and its duration was in the range of from 0.30 to 0.45 seconds. Several parameters were in the peak transient phase during this time interval. The significant parameters lost were the thrust-chamber pressure and hydraulic actuator position measurements, highly desirable data.

These functions were recovered by means of an on-board magnetic tape recorder, which recorded the data and reproduced it over a separate telemeter r-f link on launch vehicle-1 through -4. The auxiliary system was removed as a weight-saving measure, inasmuch as the systems operation was to have been thoroughly evaluated by the time four flight tests were accomplished. However, there still remained the question of whether, on any individual vehicle, the structural loads generated by the engine and/or hydraulic system transients would affect the trajectory and whether, in case of structural failure, the cause could be identified.

On-board equipment changes to add the storage capability were considered but abandoned because of their extensive weight and the time required for requalification of the system. However, it had been observed that the signal from the spacecraft telemeter was attenuated very little during booster staging, and it was reasoned that the flame attenuation was less at the forward end of the vehicle. Therefore, it was postulated that a receiving station ahead of the vehicle would have a better chance of data recovery. The basis of the supposition was the fact that sometimes the station at Grand Bahama Island (GBI) had a shorter blackout period than the Tel-2 station at ETR, even though GBI was but slightly forward and off to the right of the normal trajectory at time of staging.

In comparing an airborne telemetry receiving station with a shipboard station capability, it was apparent that the ship antenna would have ten times the antenna power gain. With an aircraft at roughly 25,000 feet altitude and the staging altitude of 209,000 feet, the slant range would be little different between the aircraft and ship location, and a shipboard downrange station was recommended.

Actually, both stations were utilized in attempts to recover data; aircraft only was used on GLV-9, and both aircraft and instrumentation ship were used on GLV-10. Both trials were unsuccessful and no further effort was expended.

k. G80 Test Set Investigation

The G80 test set was manufactured by AGC for use in checkout of the landline and telemetry instrumentation system transducers with associated wiring. It was utilized at AGC, M-B, and Complex 19 ETR during propulsion subsystem testing.

During the course of vehicle testing, particularly at VTF, failed pressure transducers presented a major problem in the instrumentation system. None of the failures occurred during operation of the test; they happened either with the application or removal of power and were attributed to over-voltage transients. On one occasion three transducers were inoperative after power application; on another, four transducers were inoperative. Moreover, both vehicle stages were affected.

The AGC engine system tests were performed concurrently with other vehicle tests. Because no indication of excessive voltage was uncovered on the vehicle power bus monitoring oscillographs, nor was the over-voltage alarm tripped, the G80 became suspect as the cause of destructive transients.

The functions of the G80 were selected by means of crossbar switching. Measurements made included hy-pot at 50 volts, continuity, zero, and 50 percent stimulus checks and frequency checks for the frequency-to-d-c converters. Possibility existed that the 50 volts might somehow have been impressed on the 28 volt supply line.

Considerable effort was expended reviewing the circuitry of the G80, its various cables and interlocking devices in the crossbar switching. Tests were made using the G80 on a test engine at AGC, Sacramento, wherein the test procedures were deliberately performed in random sequences. Cables to the vehicle were connected in various combinations and the hy-pot test again performed and compared with initial values. Grounding of the test set to various positions on the engine frame was attempted. There was no deterioration of any of the transducers or wiring as a result of these tests.

The cables from the test set to the engine junction boxes are so configured that when excitation power is obtained from the test set, vehicle power cannot be applied to the same junction box and associated transducers. The aforementioned tests were performed primarily to detect sneak circuits or possible ground loops. None was detected and the G80 test set was absolved as the problem source.

l. Augmented Engine Improvement Program (AEIP)[33, 34]

This program, originally initiated and managed by the Titan II Program Office of the Ballistic Systems Division under the purview of the Gemini Directorate of Space Systems Division, consisted of component redesign/development of marginal areas and the basic Titan II engine system. The objectives included minimizing the amount of post-fire disassembly, rework, and replacement of various engine components/subsystems, thereby increasing the inherent reliability of the overall system. The planners of the program envisioned improvements in the engine static and dynamic seals, hot gas lines and fittings, TCV lipseals, lubricating/cooling oil, PSV, thrust chamber pressure sensor (TCPS), solid start cartridge, gas generator valves, and a variety of selected turbopump assembly (TPA) modifications. Also, the Gemini RESS, for Stage II, was an AEIP development item. To complement the basic program, and to comply with the available time constraint, the following go no-go program ground rule was established:

If a particular design approach failed to meet its pre-established requirements during an evaluation cycle, no redesign or new designs would be initiated, which, of themselves, would require another evaluation sequence. The only exception to this would be when failures occurred early enough in the design/evaluation cycle so that sufficient time and funds (within the overall time planning of the program) existed to make such a recycle worthwhile. The major improvement desired from the AEIP were the TPA tasks, which, in the case of Stage I, were cancelled in November 1964 because of lack of adequate development success in the new lubrication and seal material areas. Stage II TPA tests showed significant improvements in lubrication capability and bearing temperature control, but the results of this effort did not show a promise of probable elimination of the teardown and inspection of the Stage II TPA.

Although some items in the program exhibited satisfactory results, a decision to withhold incorporating these "improvements" was necessitated by, and predicated upon, the failure of other key improvement tasks to be satisfactory (go no-go criteria) as well as by the decisions to delete GLV's 13-15 from the Gemini Program and to incorporate additional procedures to allow completion of program objectives without the need for major hardware changes. Certain select Engineering Change Proposals (ECP's) resulting from the AEIP development, as well as some baseline ECP's, were incorporated. The RESS development under this program was incorporated in GLV-3 and subsequent vehicles. Subsequently, much of the experience accumulated on the AEIP has been/will be incorporated on Titan III.

m. Stage I Gearbox Failures[33]

During Titan II development, the Stage I gearbox encountered two problems of major significance. The first and most critical was the failure of the idler gear rim, which resulted in catastrophic failure of the turbopump assembly. The cause of the failure was found to be a gear-rim resonance at operating speed, which resulted in fatigue failure at the rim/web radius. One problem solution was demonstrated to be a design improvement, which thickened the gear rim and web areas, improved metallurgical properties (9310 vacuum melt alloy vs. 4620 alloy), shot peened surface treatment to improve resistance to stress risers, and improved gear-tooth profile. Similar design improvement features were also incorporated into the fuel and oxidizer gears. Only GLV-1, of the 12 Gemini vehicles, was launched with gearboxes without the modification.

The second problem of major significance was incipient failure of the turbine shaft high speed bearing. This problem was originally isolated to a bearing design unique to one manufacturer. Resolution of the problem at that time was to restrict bearing usage in this application to a single manufacturer whose product had a substantially higher load-carrying capability. However, during Acceptance Testing of a Stage I Gemini engine in July 1964, failure of the No. 6 bearing caused premature termination of the test when an abrupt drop in the performance of SA-2 was noted. This malfunction precipitated a major analysis to find the cause of the failure. The catastrophic results of this failure mode of the engine system dictated that the problem be defined, a solution generated, and appropriate corrective action implemented prior to any manned flight on the Gemini program.

The subsequent failure analysis revealed many areas where redesign would be desirable. One item, however, was selected as the most probable cause of the failure. It was hypothesized that initial failure of the turbine phosphor bronze interstage labyrinth seal had precipitated the events leading to the bearing failure. Acceptance of this hypothesis led to the incorporation of a new free-floating stainless steel labyrinth seal. Furthermore, the history of failures indicated that if a No. 6 bearing were going to fail, the probability of failure on the first acceptance firing after gearbox assembly was very high. Since Gemini was using only selected bearings in this application, and since all Gemini turbopumps were required to be hot fired, it was concluded that the probability of having a faulty condition was low and that the probability of detecting a faulty bearing by virtue of the hot-fire requirement was high.

n. Gemini Stability Improvement Program (GEMSIP)

One of the major concerns in man-rating the Titan II Stage II engine was the possibility of combustion instability during the Stage II start transient. The original Stage II engine utilizing the production quadlet injector was classified as statistically stable but dynamically unstable.[35] This meant that the quadlet injector had a demonstrated instability incident rate of about 2 percent during ground tests. Even though this incident rate appears low, the resultant effect of an instability during flight could be quite severe. Instabilities to date have occurred during ground tests; none during flight.

The demonstrated marginal stability of the quadlet injector in conjunction with the Gemini program philosophy resulted in the DOD/NASA decision to develop a dynamically stable Stage II injector. (Capable of accepting limited pulsing without instability.) The development of the GEMSIP injector consisted of analytical/design evaluation of several injector types, which were screened by thrust-chamber assembly tests. The injector screening tests, as conducted, consisted primarily of newly developed bomb/pulsing techniques derived to establish instability triggering thresholds.* The selected prototype injectors were then tested at the engine level for system compatibility. A final candidate injector then underwent a modified qualification test program. This program was integrated into the AEIP modification verification test series. To give further assurance as to the adequacy of this injector for manned flight, it was flight tested by a Titan IIIC vehicle.

The significant differences between the Titan production quadlet and the GEMSIP were: GEMSIP had seven radial baffles, while the Titan quadlet had a center hub and six radial baffles; GEMSIP baffles were approximately 1-1/2 inches shorter than the Titan quadlet, and the GEMSIP thrust per element was 200 lbs, while the Titan quadlet was 100 lbs.

The GEMSIP injector was incorporated into Gemini on GLV-8. Gemini was the only program to utilize this injector; however, the Titan IIIM program does plan on incorporation.

o. Ablative Skirt Demonstration Tests

The predicted maximum 3σ GLV Stage II burn durations exceeded the qualification duration of the Titan II ablative skirt. Therefore, a test program consisting of seven firings was conducted to demonstrate the adequacy of the skirt design for the GLV durations.[36] This program was established with a go no-go success criteria, namely that all skirts complete the planned durations intact. The criteria was successfully accomplished with all seven skirts enduring durations ranging from 198 to 215 seconds. All skirts also underwent detailed post-test examinations. The conclusion derived from this test series was that the basic Titan II ablative skirt, as controlled by GLV fabrication and inspection procedures, was satisfactory for the Gemini missions.

*Contract No. AF04(695)-517.

An additional program of two tests was conducted to verify that the GEMSIP injector with its resultant higher heat load would not degrade skirt life.[37] The skirts of this program also had incorporated special instrumentation to measure honeycomb pressure, to provide data for the Green Man/Brown Man studies. (Reported later in this Section.) These tests were also completely successful.

p. Engine Frames[38]

Early in the Gemini program it was discovered that discrepant tubing may have been used in the fabrication of several Stage II Gemini engines, as well as on other Titan family frames. The cracks were first detected on the turbopump horizontal fuel-side support during a routine radiographic inspection. The results of the ensuing investigation were as follows:

The cracks originated during tubing manufacture. They propagated when the tubing was subjected to the severe stresses of welding and heat treatment during subsequent processing, as well as to the stresses of the static load test of the frame after assembly.

The problem occurred during a certain period of time within the manufacturer's history, and records indicated that all the defective material was isolated to a single heat of raw stock. AGC was able to document those frames which had utilized this particular raw stock.

Finally, it was concluded that none of the discrepant tubing had been used on the early (GLV-1 and GLV-2) Stage II engine frames.

Although no Gemini frames had been affected, it was concluded that this incident indicated a possibility that minute material imperfections could pass the manufacturer's inspection procedures, and result in propagation of cracks during subsequent frame assembly and checkout procedures. Therefore, it was decided that, in addition to the radiographic inspection of welds and the magnaflux inspection of tubing, ultrasonic and magnaflux inspection of all fabricated Gemini engine frames would be required after the static load test had been performed.

It should be noted that, through the Pilot Safety Program, an Aerospace metallurgical expert was also required to review engine frame x-rays before acceptability of the hardware could be determined.

Titan II experience had revealed that the Stage I engine compartment was subject to base heating, i.e., a high temperature environment due to convective heating. Titan II had insulated various components for this reason. Because of some minor differences in design between the 87-7 frame and the 87-5 (Titan II) frame, and the Gemini philosophy of requiring a greater knowledge of marginality than that for the weapon system, both thermal and structural analyses were performed on the Stage I engine frame.[39] As a direct consequence of this review, steps were taken to require that insulation protection be provided for the frame and specific MDS components.[40, 41, 42]

q. Start Cartridge Conditioning

Early in the Gemini program, it became apparent that there was a problem associated with the reliability/adequacy of the Stage I and II start cartridges utilized for the flight test program. The basic problem centered about the variability in burning characteristics of the grain being used. This variability was further accentuated by the effects of temperature and aging. For example, under the possible operating temperature range of from 35° to 100°F, a cartridge could drive the engine unstable through the mechanism of a low-step chamber pressure, or fail to start the engine because of shortened burning time resultant from the shape of the burning curve. A reasonably extensive analytical study backed up by limited emperical data only served to verify the overall lack of understanding with respect to grain composition, temperature affects, and aging affects. Because of the amount of time and cost that would be required to develop a cartridge grain less sensitive to the above problems, it was decided to design flight-type thermo conditioning equipment that would maintain the grain within narrow temperature limits thereby providing a reasonable assurance as to the expected cartridge operation during flight. In effect, this was avoiding the real problem of variability by eliminating the major item contributing to this variability. The airborne start cartridge conditioners, along with the ground support equipment, were designed, developed, and used quite satisfactorily on GLV's 2-12.

r. Dust Cap[43, 44]

The first attempt to launch GLV-6 on 12 December 1965 was automatically aborted after ignition when an electrical tail plug accidentally became disengaged. Subsequent routine investigation of the engine-generated data revealed that the thrust (Pc) of one subassembly was decaying prior to the commanded shutdown. The problem was diagnosed as a restriction in the gas generator circuit of SA-2, which would probably have caused an engine shutdown at 87FS1 + 2.2 seconds. Subsequent disassembly of the system revealed a protective dust cover in the gas generator oxidizer injector inlet port. (See Figure II. E-1.)

A complete review was made of all engine associated rework procedures, as well as a review of the rework history of all engines. Evaluation was made of a proposal to redesign dust caps, as well as standardizing the type of protective cover to be used for particular components in each work area (such as AGC, Sacramento, M-B, and the ETR).

The situation was resolved by the institution of a procedure whereby a complete accounting of the number of closures installed and subsequently removed from a system was required. As far as physically inspecting other Gemini engines for the same discrepant condition, the historical review indicated the one engine that had suffered the dust cap episode was unique; no other Gemini engine had been subjected to that particular procedure in the field.

s. Oxidizer Pressurant Pressure Switch (OPBPO) (PoPoi Autogenous System Orifice)

A part of the prelaunch Malfunction Detection System logic requires that the OPPS register a "pressure up" condition ($\geq$ 445 psia) at 87FS1 + 2.2 seconds, in order that a "go" condition exist for liftoff. Review of data from early Gemini flights revealed that the attained pressure had been marginal at the interrogation point (87FS1 +2.2 seconds) on three flights, i.e., an abort was narrowly averted on GT-4, 5, and 7 even though the engine (the autogenous system in particular) was operating satisfactorily.

Investigations revealed a disparity in pressure levels between Acceptance Test values and flight values, with the flight values being lower and resulting in the marginal condition. It was further discovered that the configuration used by AGC during the Acceptance Test series to simulate the pressure drop in the flight system had a higher resistance than the actual flight system, and therefore produced higher values in pressure during the Acceptance Test than would be attained on the subsequent flight.

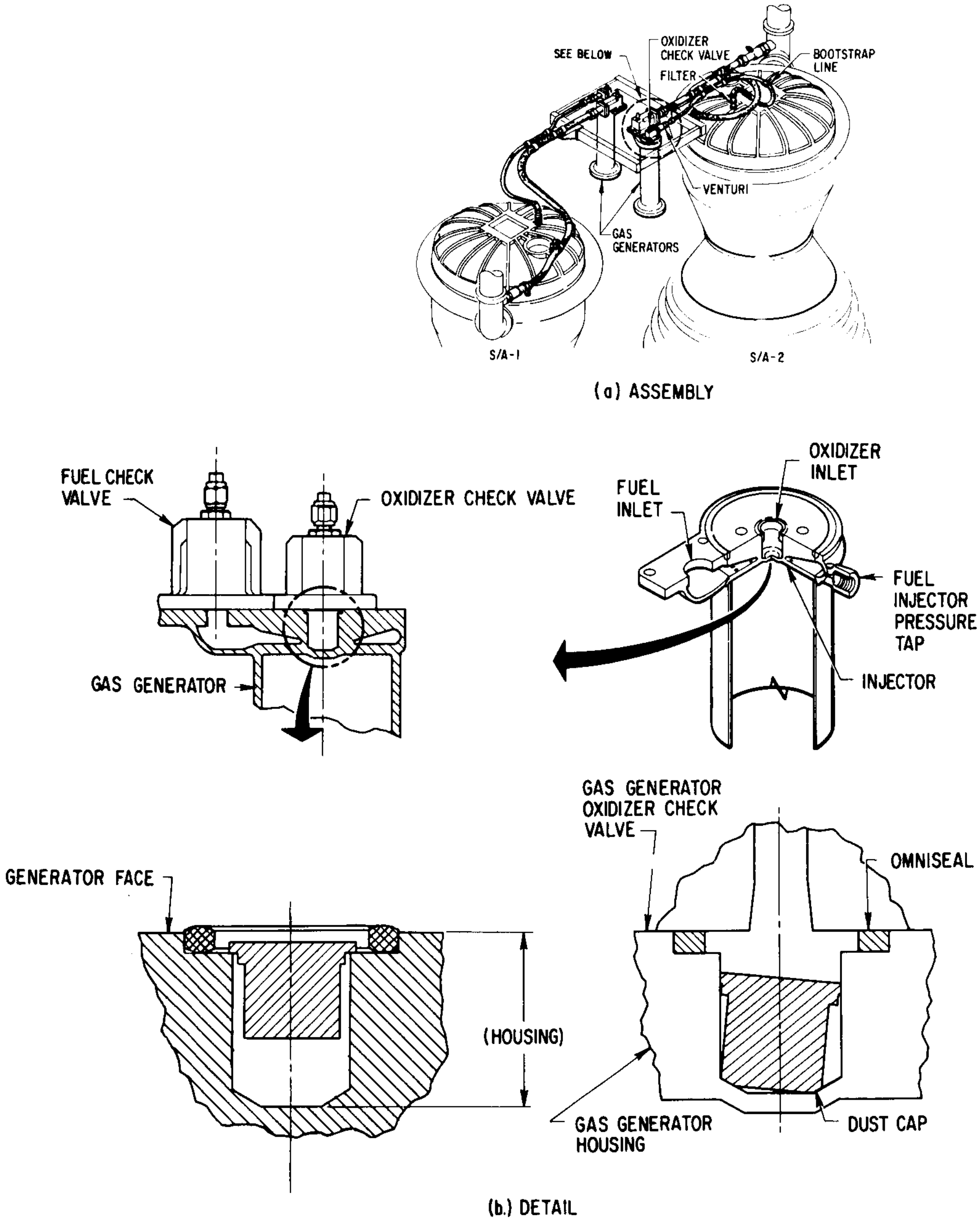

Figure II. E-1. Dust Cap Incident

A detailed review of T-II/Gemini data, utilizing different orifice diameters, was conducted. In order to correct the problem, it was decided that an orifice (PoPoi) of 0.460-inch diameter should be installed on all Gemini flight engines. This size was calculated to produce a pressure great enough (445 psia) to give a "go" cue to the PMDS at the interrogation point, while maintaining the maximum pressure below that level (600 psia), which would put the line in danger of distortion, with the autogenous system functioning properly.

t. Engine Driven Pump Compensator Change

At the inception of hydraulic-system redundancy on GLV, a review of Titan II experience indicated a possible problem area associated with the "soft start" of the secondary system engine driven pump. It was suspected that, with the hydraulic system initially at zero pressure and the engine pump compensator at full flow position, a rapid pump startup would result in a high pressure spike, due to pump overshoot, before the compensator could reduce flow to the normal steady state value. Titan II solved this anticipated problem by setting the electric motor pump discharge pressure at 3200 psi and operating the pump throughout Stage I engine startup. This resulted in the engine pump compensator remaining in a fully feathered position during the rapid acceleration startup, and thus no high pressure excursions were experienced.

With the use of two hydraulic systems on Stage I of GLV, it was necessary to start the secondary system engine pump with zero system pressure. Therefore, a test was devised and conducted during Phase I of the ASFTS program to determine if a pressure overshoot problem existed. This test was considered to be more severe than the actual Stage I rocket engine startup. The test led to the conclusion that the system proof pressure of 4500 psi would not be exceeded.

However, during SCF on GLV-1 the secondary system pressure peaked at approximately 475 psi during engine start. This level of pressure transient exceeded the proof pressure limit of the hydraulic system, thus jeopardizing the integrity of the system and affecting the safety of the launch vehicle crew. The pressure dip, which occurred after the peak, dropped to values below the pressure switch setting utilized in the hold-kill circuits for vehicle shutdown. As a result, the switch was locked out of these circuits for GLV's 1 and 2.

At that time, a method of improving pump compensator response was given prime consideration. It was decided to change the pump configuration from a short differential design to a standard differential design, in an effort to reduce the pressure transient to acceptable levels.

The short-differential compensator used a small control orifice and a tapered plunger while the standard differential used a larger control orifice and nontapered compensator plunger. The short-differential compensator, with the necessarily smaller control orifice, had relatively poor transient response characteristics as compared to the standard differential. Thus, the standard differential could maintain more constant levels of system pressure during transient flow demands of the actuators.

The standard-differential compensator pump was successfully flown on the Stage I secondary systems of GT-II and subsequent missions. No excessive pressure overshoots were observed and the subsequent pressure dips did not approach the final pressure switch setting of 2500 psia, which was used to detect a system malfunction prior to liftoff. Pressure transients settled out to a steady state at least 1.5 seconds before vehicle liftoff in all cases.

u. Propellant Discharge Lines

A Titan II propellant line was discovered with a dislocated knuckle in a tripod/bellows assembly. Although the exact cause was not determined, it was probably handling damage (i. e., dropped or shifted loads). The possibility existed that this discrepancy could result in a catastrophic failure of the line prior to liftoff due to starting loads, or in flight due to gimballing loads. The requirement was established to inspect, by X-ray, all GLV propellant discharge lines after erection of the vehicle on Complex 19. This inspection was conducted and several discrepant lines were discovered. No redesign was conducted to eliminate this problem for Gemini; however, it is probable that redesign may be conducted for other programs, as this is a Titan family problem.

v. Propellant Tank Level Sensor

Because of the development and reliability problems encountered with the Titan II level sensor and the criticality of the level sensor in the GLV application as a performance-measurement tool, a study was undertaken to find a reliable replacement for the Titan II unit in the GLV application. The study resulted in the selection of a new design level sensor, due to its inherent accuracy which conformed to the stringent GLV performance requirements. The Titan II sensor utilized optics to sense the presence of fluid. An internally generated light source was either reflected or not reflected to a solar cell, depending upon the presence of liquid on the sensor prism. The component history throughout the Gemini program showed many quality control problems, but few functional design problems.

A sensor chattering and recover characteristics was observed with fuel level sensors during Titan II piggyback flights and early GLV flights. The problem was defined as autogenous gas condensation on the sensor prism, which caused a false covered signal. The problem was solved by providing a shield for the sensor prism to eliminate the collection of condensate on the prism. The problem was not discovered until the flight test phase of the program. This was due to the fact that the engine firings at AGC, Sacramento utilized cold-gas tank pressurization rather than autogeneous hot gas when the level sensor evaluation tests were made.

Further problems were encountered on GLV-3 with cracking and crazing of the bond between the sensor's aluminum housing and the prism. To prevent introducing contamination into the propellant and loss of structural integrity of the bond, an aluminum retainer disc was bonded over the area of suspect cracking.

A problem of loose sensor shields under vibration environment was remedied by improvements in the roll pin design and installation used in connecting the shield to the sensor housing.

The level sensor engine shutdown capability was removed from the Gemini Vehicle, after a long study to verify flight safety with propellant exhaustion shutdowns so that the additional performance potential could be achieved.

Redundant level sensor capability was removed from the high sensor positions for GLV-5 and subsequent vehicles, and from the outage sensor positions for GLV's-9, -10, -11 and -12 as a weight-saving item.

Detailed information concerning level sensor development and data may be found in M-B, LV-234.[45]

w. GLV-2 Tandem Actuator Problem Investigation

The first GT-II launch attempt was scrubbed on 9 December 1964 when the primary servo valve body on actuator 2_1 failed during the engine start transient. Subsequent loss of primary hydraulic pressure resulted in an automatic switchover to the secondary systems, which in turn initiated the automatic shutdown and cancellation of the launch attempt. (See Figure II. E-2.)

The efforts to correct the problem with the actuator were initially undertaken in two directions. The first step, based on initial observations, was the decision to strengthen the servo valve body lugs to the point that they would take the higher pressure loads that must be occurring during engine start. This action was implemented immediately on a priority basis. It was thought that, if the investigation showed this to be the only modification necessary, fixed actuators would be ready to fly with a minimum of program delay. The second step was an investigation into the cause and level of forces resulting in the failure. This entailed a metallurgical and stress analysis of the failed part, and an extensive impulse-test program on the GLV and Titan II booster actuators. Later, an analog study, an intensive review of existing flight and ground engine start-test data, and a complete review of the actuator design were added.

The metallurgical analysis of the failed servo body revealed a relatively coarse grain structure for a small die forging and a marginal heat treat, as evidenced by the presence of eutectic melting along the grain boundaries. However, the strength properties were high enough that a failure of the type experienced should not have occurred with internal loads predicted at the time. Therefore, further investigation to identify all the internal actuator conditions under impulse loads became a requirement.

In an attempt to reproduce the actuator failure and to study the actuator response to impulse loads, an impulse test setup was devised in the Martin-Baltimore facility. A high impact shock machine was modified for the test program, such that either tension or compression axial impulse loads might be applied to the actuator. The concept of the impulse test program was to allow free fall of a pendulum, imparting Kinetic energy to its weight, which represented equivalent gimballed engine weight reflected through a striker arm into the actuator. The spring rate of the fixture was designed to represent vehicle backup structure. The fixture induced strain, plus damping on the actuator, produced an impulse of 0.030-second duration during rise and a return to zero within approximately 0.050 second at medium-impulse levels, closely simulating the observed conditions under actual engine start conditions.

In laboratory tests, the exact GLV-2 failure mode was reproduced twice with Gemini actuators, failures occurring at 42,250 and 35,900 pounds actuator load under 11,200, and 8,070 in.-lb. of impulse energy, respectively. It was estimated that the GLV-2 failure occurred at an actuator load of 34,700 pounds. Two Titan II booster actuators were also tested to destruction to provide a comparison with the tandem actuator. Throughout the T-II history, there had been no evidence of damage to the actuators, and thus their failure point was of prime interest. One unit, at a load of 49,500 pounds or impulse energy of 16,200 in.-lb., suffered a burst flapper flexure tube and yielded bearing straps, while the other unit exhibited a buckled piston rod at a load of 47,500 pounds or impulse energy of 12,200 in.-lb. The impulse failure levels of the GLV-2 specimens were below the failure levels of

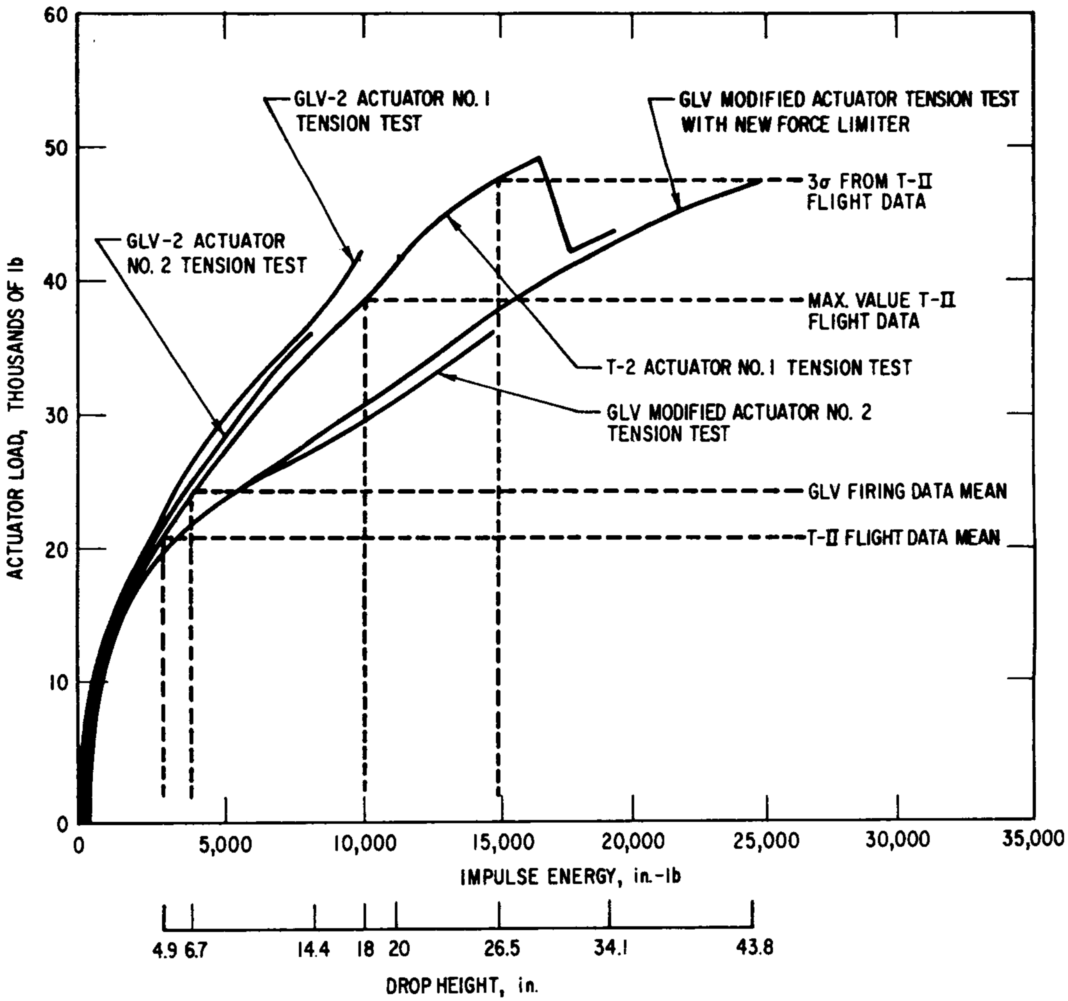

Figure II. E-2. Test Data Summary - Actuator Load versus Impulse Energy the Titan II specimens, because the tandem actuator was less compliant and the servo valve housing was weakened by miniaturization for repacking, and by variations in material process control which allowed the pressure generated loads to cause failure.

Actuators installed during early engine start tests at AGC recorded maximum loads of 14,500 pounds, well below the computed GLV actuator failure load.

Examination of the tandem actuator failure load in terms of the early T-II test data showed the need for a re-evaluation of engine start loads. From flight data Titan II and GLV-1 and 2 start transient loads were recalculated using piston position telemetry measurements, which were converted into loads and impulse magnitudes by utilizing the load versus velocity information generated on the respective actuators by the impulse testing. The calculated GLV-2 actuator impulse loading was greater than the previously accepted maximum-impulse data derived from early T-II engine test programs at AGC in Sacramento. The calculated GLV-1 and 2 impulses were higher than the Titan II mean, but were below the maximum Titan II impulse. The probability distributions and maximum values of the recalculated Titan II tension and compression impulses were used as a basis for evaluation of the modified GLV actuators and their qualification.

A series of impulse tests were made with AGC-supplied stiff links, to provide a comparison of stiff-link loads to actuator loads for a given impulse. The equivalent stiff-link load at engine start impulse levels was much greater than anticipated from early AGC engine tests at Sacramento, and higher than the specification value for design of the tandem actuator.

The analysis of actuator internal pressure data during impulse testing indicated a much higher than expected pressure level in the return pressure port of the servo valve. The pressure level was equal to that recorded in the supply ports of the valve. At this point it was evident that more than a weak servo body section was involved, since the return pressure within the valve was considerably above values used in the initial stress analysis of the actuator assembly. A detailed design analysis of the servo actuator indicated several areas for further study. The force limiter discharge was ported into the return pressure line upstream of the flow limiter; therefore, the back pressure on the force limiter was a function of flow limiter response. The normal flow limiter response allowed high back pressure buildup on the force limiter, thus raising its cracking pressure and increasing return pressure within the actuator. It was felt that the force limiter flow capacity was not great enough to allow rapid dissipation of peak internal pressures.

Utilizing the actuator and load information generated through the impulse testing, design and modification was started with the following constraints. 1) The loads produced by established impulse requirements must be bounded by inherent strength limits of the actuator, its attachments and backup structure, 2) modifications must be relatively simple and minor, so that they could be fully developed and evaluated to minimize delay of the program. The forged housing strengthening program was continued, as well as enlarging and rerouting actuator internal passages (and increasing the flow capability of the force limiter design) to reduce the backpressure on the force limiter.

Strengthening of the 2014-T6 forged housing was accomplished.

Existing fluid passages were enlarged and one new fluid passage added in the primary section of the main body to permit connection of the discharge side of the force limiter assemblies to the return manifold downstream of the flow limiter. This change permitted operation of the force limiter to maintain cylinder pressures and return pressure at the desired threshold, independent of the back pressure generated during operation of the flow limiter.

The main objective of the force limiter design modification was to increase its flow capability, compatible with actuator rates experienced during the engine start transient. To accomplish this, the damping orifice in the main flow path was removed to open up the passage. Since previous experience had shown that deletion of the orifice resulted in valve instability, the control or sensing area was separated from the main flow path and a damping orifice was inserted in this section.

Two actuators, modified as described in the preceding paragraphs, were impulse tested in the Martin-Baltimore laboratory. The first failure mode encountered on the modified GLV actuator was similar to the Titan II tension failure mode, but occurred at a 50-percent greater amount of impulse energy. At pressure levels above the normal hydraulic-system burst pressure level of 7500 psia, incipient failures developed in the servo, the first of which was distortion of the flapper flexure tube. The flexure tube distortion mislocated the flapper, resulting in a servo null shift. An actuator load of 36,000 lbs was set as criteria for allowable servo null shift, without running the danger of a switch-over due to actuator offset. This actuator load level was approximately 30 percent higher than any observed on GLV or Titan II flights. A complete functional evaluation test series was run on the modified tandem actuator in the Martin-Baltimore ASFTS test area. The tests were conducted to prove that the actuator performance characteristics had not been altered by the modifications.

A requalification test series was set up for a third modified actuator using a series of impulse tests, each test followed by a functional checkout of the unit. Impulse tests were conducted simulating a flight test mean impulse plus 1- and 2-sigma and maximum levels. No problems were encountered until impulse levels reached a point where actuator null shift appeared.

The modified actuators were successfully flown on GLV-II and subsequent vehicles.[46]

Conclusions

The tandem actuator failure on the GLV can be attributed to three primary engineering factors. First, there was no impulse test included in the qualification program for the tandem actuator. It is not now clear why, but it was felt by all engineering parties concerned that the snubbing test, which was part of the qualification requirements, did demonstrate the capability of the force limiters to absorb an impulse load into the actuators. An impulse load was included in the design specification which had a considerable margin above the predicted three-sigma stiff-link peak value that had been observed in the early T-II engine test, but the energy content or the shape of the impulse was not defined.

The increase in starting loads was the second contributing factor to the actuator failure. It became apparent in reviewing the history of the first-stage engines that a revision to the injector and manifolding was responsible for the change in start characteristics of the T-II engine, because of the correlation with available engine data. A series of tests on the revised engine for start loads or the evaluation of actuator pressure traces on early T-II flights could have exposed this change in start load characteristics.

The third consideration revolves around the design and development of the tandem actuator. The study or the development testing of the actuator for its response to externally applied impulse loads would have exposed the interaction of various components within the actuator and the fact that they could limit the effectiveness of the force limiters in relieving reaction loads to forcing inputs. The T-II actuator operated successfully mainly because it was more compliant due to configuration differences from the Gemini actuator.

x. POGO Suppression Device Development

The first flight of a Titan II vehicle in March of 1962 revealed a longitudinal oscillation problem that resulted in high acceleration levels at the payload. Accelerometers at the payload recorded 5 g's peak to peak at a frequency of around ten cycles toward the end of first-stage flight. The observed oscillation was later established to be a closed-loop instability between the structure, propellant feed system, and the engines and was dubbed with the title POGO. Ballistic Systems Division (BSD), Space Technology Laboratories (STL), and the rest of the Titan II contractors established an investigation program to evaluate and solve the problem. Later in the year a formal committee composed of the involved contractors was set up to direct, control, and integrate all of their action under BSD. It was through this committee and its meetings that all information was distributed and technical recommendations for action made.

Two significant parameter effects were noted in the evaluation of the first two flights. One was the apparent presence of a feed-line resonance of the same frequency as the first longitudinal mode at the time of oscillation occurrence, and the second was the apparent amplification of the fuel feed line pressure fluctuations across the fuel pump. The third flight was made with increased fuel tank pressure to force the fuel pump operating point further away from the cavitation point with a resultant improvement or reduction in oscillation level. The next significant fix attempted was the incorporation of a tuned attenuator in the ox-feed lines just above the two Stage I ox-pumps. This attenuator was in the form of a standpipe tuned to the observed resonance frequency of the feedline. A standpipe fix was flown early in December of 1962 with effectively disasterous results. The standpipe-fixed vehicle produced oscillations that reached 10 g's peak to peak at which point the pressure fluctuations in the engines were great enough to effect their shutdown.

The failure of the standpipe fix resulted in an extensive reevaluation of the problem by the weapon system contractors and the initiation of efforts by Aerospace and Martin-Baltimore of the Gemini team to evaluate and understand the problem. The evaluation of the anomalies and previous flights, by the weapon system contractors and the Gemini contractors working on the problem, did not reveal any basic errors in the analytical model or hypothesis upon which the standpipe fix had been proposed. Evaluation of the flight did reveal, however, that the amplification between the suction line fluctuations and the resulting discharge pressures in the fuel system was much greater than that which had been used in early analysis, and that these gains when incorporated in the analytical model did predict the instability with the standpipe fix. It was determined by the interested parties that the only logical approach to fixing this problem was to institute a series of tests and evaluations that would give parametric data and basic understanding of the dynamic behavior of the propulsion system. The means for obtaining this data were to be a series of ground tests on a complete engine at Aerojet and a pump drive assembly (PDA) test at the Martin-Denver cold-flow facilities, with provisions for dynamically perturbating suction pressures and, in the case of the PDA tests, the discharge pressures. Both test setups were extensively instrumented. Further evaluation led to the conclusion that active steps had to be taken in the fuel side of the propellant system, either upstream or downstream of the pumps, to suppress or attenuate induced pressure oscillations in the frequency range of the problem, as well as to keep the standpipe fix on the ox feedside. During this evaluation phase, several Titan II flights were made with increased fuel tank pressure, but also with a change in the oxidizer feedline from a stainless steel configuration to an all-aluminum configuration. These vehicles with the aluminum feedline and the increased fuel-tank pressure showed a significant decrease in oscillation levels. These levels, which were running around 1 g peak to peak, were well within levels that the weapon systems payloads could tolerate and it became evident that from a weapon systems point of view there was no need for further activity or expense in solving this oscillation problem.

The expected levels of 1 g peak to peak were not consistent, however, with requirements of the manned version of the vehicle which had been established as a maximum of plus or minus 0.25 g's by NASA. It was at this point that the Gemini team through and with SSD took a more active part in solving and obtaining fixes for this problem. Because the weapon systems contractors already had the facilities and were geared up to investigate the problem, the decision was made for these people to continue their active pursuit of the problem, with the Gemini team providing inputs in steering the efforts to achieve a satisfactory solution for the Gemini Launch Vehicle. From this point on, the program effort was basically divided into two categories: 1) The active pursuit of an adequate analytical model, and 2) the development of various attenuator configurations to suppress oscillating pressures in both the ox and the fuel feed systems.

Active pursuit of an analytical model was being undertaken by STL, Martin-Denver, Martin-Baltimore, and Aerospace. The basic approach being used by all four was similar, but each of the participants was actively reviewing some of the subtler aspects that they felt might be the key to obtaining a good match between recorded data from the flights and the models. The tests of the fuel pump assembly at Martin-Denver and the full engine at Aerojet were to provide some of the key parameter data necessary for a complete model. The major problem with the analytical model was that it did not predict the return to stability before BECO that was observed during the flight.

In order to provide backup for the two most favorable fixes for the ox-feed system and the fuel-feed system, several alternate fixes were picked for further study and some development. In the ox-feed system a "lossy line" and an acoustic horn were proposed as alternates to the standpipe, in order that pressure oscillation suppression could be achieved over a broader frequency range than was possible with the standpipe. In the fuel-feed system, evaluation was made, not only of a simple mechanical spring piston accumulator, but of a number of types of accumulator in both the suction and discharge lines from the pump. A large effort was initiated at Aerojet to redesign the fuel pump for improved suction characteristics with the constraint of allowing only modifications to the inducer.

Although the analytical model did not predict the cessation of oscillations, a good margin of stability was predicted with the use of both a standpipe in the ox and a mechanical accumulator in the fuel suction side. The four contractors making detailed analytical studies agreed on this conclusion and the decision was made to try the standpipe and accumulators on vehicle N-25 by BSD. Prior to the flight of N-25, data became available from the transfer function tests on the engine at Aerojet, which showed and confirmed the presence of a fuel-side resonance with a frequency which would enable the analytical models to predict not only the onset of oscillation, but the cessation. This information provided additional confidence that the forthcoming flight of N-25 would indeed prove successful, since it was this inability to predict the cause of the cessation of oscillation that had caused the greatest concern as to the accuracy of the analytical model. The flight of N-25 did indeed show that the standpipe and accumulator combination was the key to controlling the POGO oscillations of the Titan II vehicle.

Further analytical studies and development tests indicated that some of the alternate approaches such as the "lossy" ox feed line and a different type of accumulator on the fuel-feed side would be beneficial and provide a better overall solution in controlling this closed-loop problem. The availability of R & D flights in the terms of development lead time made it impractical to pursue these other approaches, however, and the continuing success of other test flights with the standpipe and accumulator demonstrated that the NASA criteria of ± 0.25 g's could be achieved with this hardware, and thus the alternate approaches were never developed completely or incorporated in the Gemini program. The redesign of the fuel pump was completed at Aerojet and production-type pumps were developed. The pump development was continued through to completion and two pumps were scheduled for one of the last T-II R & D flights. This was done to provide a backup capability in case the fuel-side accumulator was not sufficient by itself to suppress fuel-side pressure fluctuations with low fuel tank pressures. This modified fuel pump did fly on one of the T-II R & D flights, but did not show the performance benefit that was expected. This occurred, however, after a low fuel-tank pressure flight had been made with both the standpipe and accumulator, and had demonstrated the adequacy of these two fixes even with the low fuel-tank pressures, thus obviating the need for the improved pump.

The piston-type fuel accumulator and oxidizer standpipe were successfully flown on all Gemini Launch Vehicles. Several changes were made to the original accumulator design for the GLV application, including changes in seal and bushing material to reduce piston friction, and hardcoating the piston shaft to reduce wear and the particles generated by wear. Problems with the accumulator piston position potentiometer were solved by changing lead wires to a heavier gauge, placing lead wire solder connections inside the potentiometer case, and, beginning with the GLV-4, protecting both the accumulator and potentiometer from recirculation heating with a shroud.

Only on GLV-5, where a problem was encountered in properly tuning the oxidizer standpipes, was the ± 0.25-g limit exceeded. On this flight, the target level was exceeded by a slight amount for a period of 16 seconds. No adverse flight conditions were experienced. Some oscillation at the end of Stage I flight was observed on all flights, but they were of the short burst, low amplitude character rather than the sustained type encountered without the surge chambers.

A detailed study of the POGO problem along with the analytical studies and corrective action will be found in the references at the end of this section.[47-54]

y. Oxidizer Standpipe Remote Charge System

The method for suppressing longitudinal oscillations (POGO) on the Gemini Launch Vehicle made use of tuned hydraulic resonators, or peak-notch filters, installed in the Stage I propellant feed lines. The resonators in the oxidizer feedlines were standpipes, which utilized an entrapped gas bubble to provide a soft spring for the oxidizer mass in the standpipe to act on.

Because of initial trapped gas quantity variations in the standpipe, a charge process was required during launch countdown to ensure proper sizing of the bubble and thus correct tuning of the standpipe. The charging process was initially done with manual tools; however, this function proved unsatisfactory because of safety problems, and because it necessitated early opening of the oxidizer prevalves and exposure of the engine to oxidizer. Initial propulsion operating constraints resulted in considerable system servicing if a launch scrub occurred after prevalves were opened, which in turn lengthened the recycle time required before another launch could be attempted. To remedy this situation, a remote charging system was developed which allowed the prevalve opening and standpipe charging operation to be moved up in the countdown to any convenient time prior to initiation of the automatic count at T-6 minutes.

The system was designed and developed to the following design criteria.

1. The charging system was to be designed so that a minimum of modification would be required on the airborne side of the interface.
2. The system was to be capable of remotely controlling all operations from the blockhouse.
3. Flight-proven hardware was to be used as applicable.
4. All new hardware was to be subjected to qualification to GLV requirements.
5. Redundancy was to be used in systems design where practical, in an attempt to increase reliability.
6. The charging system was to provide enough capacity to adequately clear the standpipe of propellant and then ensure that the system would allow adequate propellant bleedback to clear the standpipe of excess nitrogen. This also ensured that the correct size gas bubble was then trapped in the standpipe.
7. The system was to have the capability of sensing an improper charge in the event of a charging system malfunction.

The component and systems design was finalized by a series of test runs on the operational configuration. These tests were designed to prove system repeatability, reliability, and the ability of the system to sense a charging malfunction.

The design criteria were achieved. The airborne hardware was modified only to the extent of adding some tubing, a ball-type shutoff valve, and a quick disconnect. The AGE design fulfilled the rest of the objectives. The quick disconnect was the same as that used in the topping system, but the ball valve was new and required qualification.

The system was successfully used for GT-4 and subsequent launches. The POGO problem on the flight of GT-5 resulted from a design deficiency in the manual charging system. The remote charging system could not be used for this flight due to maintenance problems.

Detailed information concerning systems design and development may be found in the reference at the end of this section.[55]

z. Propulsion System Test Program[19, 20]

The purpose of this test program was to check and verify the operation of those portions of the Titan II/Gemini propulsion system that were peculiar to the Gemini Launch Vehicle. The primary objectives were:

(1) Vehicle

- Verification of the tankage malfunction detection system
- Demonstration of capability of the spacecraft display
- Verification of pressurization system and ullage reduction
- Verification of the tandem actuator system
- Development of operational history of the prevalve
- Accumulation of operational history on the propellant level sensors

(2) Engine

- Verification of the malfunction detection system
- Verification of the prelaunch malfunction detection system
- Verification of the Stage I engine frame
- Durability of cold propellants
- Verification of flight instrumentation
- Verification of low/critical NPSH
- Engine starting capability under conditions of reduced ullage

The test results demonstrated the ability of the Gemini propulsion system to meet its design goal. All vehicle and engine test objectives were satisfactorily achieved.

a'. Side Load Disturbances (Green Man/Brown Man)

Through the course of the Titan II, Titan III, and Gemini programs, disturbances of the flight control system/vehicle were experienced. The disturbances occurring post-sustainer engine cut-off (SECO) were entitled "Green Man" and were noted in flight control and dynamic parameters, but not seen on any engine parameters. The frequency of Green Man on GLV and Titan II was approximately 40 percent. Disturbances were also experienced on Titan II and Titan III during Stage II operation (pre-SECO). These were entitled "Brown Man" and included two cases of ablative skirt failure; one on each program.

Prior to the Gemini flight program, the Green Man phenomenon was studied in detail to determine if this phenomenon could cause recontact of the Gemini capsule and GLV Stage II. No cause was established during this study but it was concluded, mainly from flight-test data, that recontact due to Green Man was remote and the probability was consistent with acceptable Gemini philosophy.

A combination Green Man/Brown Man task team was established subsequent to the Titan IIIC ablative skirt failure. This task team concluded that both Brown Man and Green Man had the same cause. The team concluded the most probable cause was excessive pressure buildup in the honeycomb structure of the ablative skirt. The side forces were then generated by either pressure relief or loss of a section of ablative material. The pressure buildup was the result of hot gas and/or resin vapor. The hot gas could migrate to the honeycomb through liner cracks or porosity, the latter resulting from liner char. The resin vapor was the result of heating of liner or inner laminates.

Other pertinent conclusions of the task team were that the Titan IIIC skirt failure was at least partially caused by quality problems in addition to the above, and that Green Man/Brown Man did not endanger crew safety or mission success. Details of the study and conclusions can be found in the reference at the end of this section.[56]

b'. Enlarged Bellows ECP - TARS HIG 4 Gyros

The three-axis reference system (TARS) gyros had been found to acquire air bubbles in their flotation chambers, which could cause unpredictably large and erratic output signals during rotations such as the pitchover motion of a flight. The fault was traced to an improperly sized expansion bellows in the individual gyros. When the gyro was allowed to cool between operations, the bellows would completely collapse and thus permit the further contraction of the fluid to pull gases out of solution, or draw air in around the lead seals from the outside. Experience with this phenomenon on the Mercury program led to an evaluation of whether there was any associated flex-lead damage problem. Fortunately, the flex-lead problem on this gyro had been found and solved in the original development program. The ECP for enlarged bellows had been submitted at a time when the original baseline contract was being formulated. The ECP was reviewed in detail, and found to be adequate. The alternative of tumble testing gyros shortly before launch, to assure that excessively sized air bubbles had not formed, proved expensive and operationally unfeasible. The delayed approval of the ECP made it necessary to change out the TARS for GLV-1 at ETR in order to fly the improved gyros.

c'. Stage II CG Offset Study

The flight of GLV-1 indicated significant and unexpected attitude variation at staging and throughout Stage II in yaw. A study was initiated to investigate 1) the dispersion of cg misalignment between the combined Stage II and Gemini spacecraft for each mission, 2) the effect of a cg offset on radio guidance, and 3) the corrective action required to bias the Stage II actuator so that the cg offset would use up a minimum amount of available actuator motion (controllability). It was found that improved cg data was needed on the Stage II GLV, and that special data on spacecraft cg was required for each mission. A nominal yaw actuator bias adjustment was incorporated into the rigging procedure. These changes satisfactorily resolved this problem.

d'. Rate Gyro Studies

Failure analysis of a frozen rate gyro gimbal was traced to contamination stuck to the gimbal assembly. Further evaluation of the vendor's facilities and procedures revealed that cleanliness conditions (though considered adequate for making rate gyros for weapon system boosters or aircraft) were far from satisfactory for gyros manufactured for a manned launch vehicle. A major revision of manufacturing and inspection techniques was made, particularly in the final assembly area. The assembly facilities were reworked completely; the assembly rooms were reconditioned, the air cleaning system was upgraded, and laminar flow clean air assembly benches were provided. Cleanliness control on the cleaning solvent used for ultrasonic cleaning was initiated. Additional inspection steps were added with microscopic examination. The assembly technique and design of the gyros made cleaning of delivered units impractical. Only gyros of the post-contamination control type were used for Gemini launches.

A study was undertaken as a result of numerous rate gyros which could not meet specifications on maximum allowable run-up time. Failure analysis of these gyros indicated that in many cases the bearing lubricant was entirely depleted in the first three-hundred hours of operation. Studies showed the motor design was far from optimum and therefore the bearings were required to conduct the excess heat from the motor shaft to the case. Just as alternate source proposals were being considered, it was discovered that excessive preload forces were being inadvertently applied to the gyro assemblies because of inadequate preloading techniques. It was further learned that, when the preload was properly adjusted, the bearing life expectancy was up from 500 to 750 hours. Because an incremental coast-down test had been evolved which would preclude the use of any marginal gyro for manned launch, it was possible to avoid the expense and schedule difficulties of developing an alternate source, without compromising pilot safety or mission success probability.

e'. RGS/TARS Torquing Sensitivity

Guidance system interface discussion following a basic guidance loop gain change, which was made early in the program, explored the relationship between radio guidance system (RGS) commands and TARS response. As a result of the exploration, it was decided that the interface was complex and too difficult to analyze. Therefore, a laboratory study was initiated wherein the effects of known signal variations in conjunction with actual hardware sampling techniques could be tested. This study revealed significant nonlinearities in the low-level torquing range, which is the range over which most of the in-flight torquing would occur.[57]

The possibility of a hardware change to improve the linearity of the torquing sensitivity in the low range was studied. Evaluation of the effect of the increased variations in the torquing gain at low levels on the GLV insertion conditions revealed that the hardware change was not necessary.

The data resulting from this study was also used to provide a more accurate model of the guidance system used in the performance trajectory studies.

f'. TARS Attitude Gyro on-Pad Stiction Test Study

The attitude gyros used on the Mercury/Atlas and the Gemini vehicles had a problem from low-force level gimbal hang-ups, which were called stiction. The close tolerances between the gimbal float and case make such gyros extremely sensitive to small-sized contaminants in the fluid. Shock of assembly or handling often causes dimples in the thrust-bearing surfaces which can also cause irregular low-level restraint. Degraded fluid which separates (stratifies), congeals, granulates, or becomes charred from overheating will act like contaminated fluid.

The Atlas programs had provided means for running stiction tests in the factory, in the field laboratory and on the launch pad to screen out bad units. The Titan program initially had only some intermediate level gyro torquing tests in the factory at approximately five times the test levels required to find this problem. Martin proposed improved bench test facilities at Baltimore and ETR, which would perform low-level stiction checks. The long lead lengths and unique design of the launch pad AGE made on-the-pad stiction tests impractical.

A review of the launch facility also revealed inadequate assurance of proper heater monitoring. Heater monitors gave visual output (blinking lights) and the operator had to note that lights were no longer blinking to detect a heater monitor failure. There were incidents where gyros had overheated because of failure of the heater monitor circuits.

It was decided that the improved bench test capability and a required stiction test after transport to ETR covered the contaminants, deterioration, and handling areas of concern. An audible overheat warning was installed in the blockhouse to satisfactorily assure that the gyros being used were not inadvertently overheated. These steps provided the required confidence in the Gemini gyros.

g'. Dynamic Studies

The Gemini program originally required end-to-end system frequency response test data on only the first two vehicles for purposes of proving specification compliance and to enable correlation between preflight predictions and inflight performance. Components were tested for individual frequency response characteristics during acceptance testing. Because of the schedule costs and the complexity of the tests, the vehicle tests were to be discontinued after the first two flights.

The benefits accrued in the end-to-end vehicle tests, such as establishing subsystem interactions and component degradation, made it worthwhile to try and find a means for obtaining this data without the attendant schedule effects or the complexity.

An approach was established with Martin, after considerable evaluation, whereby transient response data would be obtained at VTF and ETR in lieu of the frequency response data. This data was analyzed by Martin and Aerospace, using computer techniques to determine the equivalent system frequency response. This data was used successfully to predict inflight performance throughout the remainder of the program.

h'. IGS Studies

A validation study was performed at Aerospace on the spacecraft inertial guidance system (IGS)[58] to ascertain the compatibility of the system with the rest of the secondary flight control system and the launch vehicle performance requirements. This study provided the launch vehicle team with the assurance that all objectives would be achieved in the event that a switchover occurred and the flight completed utilizing the secondary guidance and control system. As a result of making this study, it was possible to review the interface requirements and test procedures and ascertain their thoroughness and completeness.

The validation study helped fill the void in development testing on the secondary flight control system that was not present on the primary system. An IGS could not be made available for ASFTS systems testing such as all primary systems components went through. Testing and flight data were thoroughly reviewed on the secondary system to screen for any potential interface problems that were not or could not be caught by the study. Several unexpected performance phenomena were observed, evaluated, and accepted during the course of the program.

i'. Effect of Launch Drift of the GLV with Respect to Stand Clearance[59]

A study was conducted by Aerospace and M-B in 1965 as the result of updated IGS null contributions to the secondary flight control system (FCS) null and its effect on the null shutdown limits. The study was made to assure that, in increasing the limits to prevent an inadvertant shutdown, a stand clearance problem at liftoff would not be created. As the result of this study and a subsequent study involving the primary FCS nulls, it was determined that Stage I actuator excursions of up to 0.7 degrees when combined with all other contributing factors would not cause a flight hazard from contact with the launch stand. Factors considered were wind velocity toward the launch stand, engine offset due to actuator length adjustment, electrical system biases, and the launch vehicle's bending modes.

The change to the primary system limits was made necessary by the discovery that normal flight-control system and actuator adjustment tolerances, when combined with expected null shifts at engine start, could fall outside the 0.25 degree shutdown criteria. The change was implemented such that any of the four Stage I actuator position transducers indicating greater than 0.7-degree engine displacement, and a Stage I actuator not operating in the null land of the position switch, would cause engine shutdown. The parallel requirement provided protection against a transducer failure and the resultant unnecessary shutdown.

j'. Gyro Test Set and AGE Studies

The gyro test set was originally developed as an engineering tool and was not intended for production testing. It was meant to be used by engineers and highly skilled technicians. A decision was made to use this test set in Baltimore as a production test tool, and to construct a duplicate to be used at ETR for flight hardware checkout. The personnel experience level and the close proximity of the test facility to the engineering design group resulted in relatively smooth and effective operation of the Baltimore facility. However, at ETR, there were numerous cases in which TARS units were jeopardized because of test equipment malfunctions and lack of equipment familiarity and detail design knowledge on the part of engineers and technicians. As a result of an extensive investigation, many changes were made to the test procedures and test equipment to try to improve the facilities effectiveness and to reduce the testing malfunctions.

During this period of uncertainty, there was considerable pressure to delete the ETR test facility which, of course, would have eliminated any further ETR test equipment problems. This was considered unsatisfactory from a technical standpoint, since it was felt that the equipment should be checked out as close to flight as possible to check for component degradation and to screen out marginal hardware.

Considerable effort was also expended during the program to attempt to upgrade the AGE equipment at the launch complex. AGE malfunctions necessitated the addition of a backup temperature monitor for TARS in case of a no-go indication. AGE relays were found to contain contamination which caused malfunctions, and new type relays were required with special process control requirements. Relay changeout maintenance procedures were required and instituted because of life limitations. The ASFTS facility at M-B proved to be very useful in isolating these types of AGE malfunctions and in the checkout of AGE design changes.

k'. Stage II Thrust Overshoot

During the flight test data review of GT-1, it was noted that the thrust overshoot on the Stage II engine reached a level of 119 percent of steady state thrust, whereas the engine specifications listed a maximum overshoot of 112 percent of steady state thrust. Structural analysis conducted at that time showed essentially a zero margin of safety on the M-B engine cone for this overshoot value, considering maximum compartment 3A pressure, maximum engine gimbal angle, and minimum tank pressure. At the time, this value of overshoot was considered an upper bound. Pressure transducer problems tended to cloud the issue, plus the fact that lower values of overshoot were seen on GT-2, -3 and -4. After that flight the FM/FM transmitter and tape recorder were removed, and there was no further collection of data in this area during the Gemini program. Considerably later in the program, just prior to the flight of GLV-8, AGC presented data gathered from Titan II testing which showed a maximum expected overshoot of 122 percent of steady state thrust. This value of overshoot, using the above assumptions, yielded a negative margin of safety; however, these conditions were felt to be conservative, especially since the compartment 3A pressure data used was derived from flights of Titan II's where the transportation section was beefed up. It was decided at that time to fly the hardware as-is, since all the hardware had been built; however, had this data been available earlier in the program it is believed that a redesign of the M-B engine cone would have been desirable, especially since a cone design of lighter weight, but considerably stronger, was available. As far as can be ascertained from the limited instrumentation carried on the later GLV flights, this peak value of overshoot was never reached, and positive structural margins did exist on all flights since there is no evidence of cone failure or deformation.

l'. Development of Range Safety Data Package for Gemini Rendezvous Missions

The rendezvous phase of the Gemini program, beginning with GT-6, contained a number of peculiarities (i. e., variable launch azimuth, short duration launch windows, ascent flight yaw maneuvering, late definition of launch azimuth) which necessitated designing a different data package to satisfy the ETR regulations and to maintain the data at a minimum. The regulations were subjected to renewed interpretation to cover a rendezvous type mission never before conducted on the ETR.

A proposal delineating the elements of a rendezvous mission data package[60] was prepared and submitted in February 1965 for review to the Program Office, Martin-Baltimore, and Aerospace/ETRO personnel. Changes were incorporated before presenting the proposed package to the 6555th Aerospace Test Wing and to the Range Safety Office. The final version was accepted tentatively by the Range contingent upon a favorable error analysis and upon examination of the rendezvous mission range safety package for GT-6. Aerospace provided the requested error analysis in August 1965 and the Martin-Baltimore prepared data package was submitted to the ETR in September 1965, about one month prior to the planned launch day.

The AFETR Range Safety Office found the data adequate except for one objection based on a violation of safety criteria of the 3σ right instantaneous impact point dispersion at 140 seconds. The problem was worked with M-B, resulting in a revised 3σ right trajectory perturbed by a reduced wind profile. Although the reduced dispersion was not sufficient to remove the discrepancy, subsequent negotiation resulted in reshaping the destruct line in the Great Abaco region and subsequent removal of the objection by AFETR.

The rendezvous mission data package was designed to provide AFETR with range safety data applicable to all rendezvous missions. Except for mission specific maximum headwind and sidewind X, Y, Z data and a nominal trajectory prepared for each mission, the basic rendezvous and nonrendezvous data packages were adequate throughout the Gemini program.

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SECTION II. E

DEFINITIONS

AEIP	Augmented Engine Improvement Program
ACI	Advanced Communications Incorporated
AGC	Aerojet-General Corporation
AGE	Aerospace Ground Equipment
ASFTS	Airborne Systems Functional Test Set
BECO	Booster Engine Cut-Off
BSD	Ballistic Systems Division
CSAT	Combined Systems Acceptance Test
ECP	Engineering Change Proposal
EEI	Electrical-Electronic Interference
ETD	Engineering Test Directive
ETR	Eastern Test Range
FCS	Flight Control System
GBI	Grand Bahama Island
GE	General Electric
GEMSIP	Gemini Stability Improvement Program
GFE	Government Furnished Equipment
GG	Gas Generator
GLV	Gemini Launch Vehicle
GPIS	Gemini Problem Investigation Status
GT	Gemini Titan Mission, (Gemini Launch Vehicle)
HIG-4	Honeywell Integrating Gyro Number Four
IGS	Inertial Guidance System
IR	Inspection Report (formerly known as QRR, or Quality/Reliability Report)
JAN	Joint Army/Navy
LO	Liftoff
LV	Launch Vehicle
MARS	Martin Automatic Reporting System
M-B	Martin Company, Baltimore
M-C	Martin Company, Canaveral

MDS	Malfunction Detection System
MDTCPS	Malfunction Detection Thrust Chamber Pressure Switch
MHz	Mega Hertz, Megacycles/Second
MOCS	Master Operations Control System
NPSH	Net Positive Suction Head
OPBPO	Oxidizer Pressurant (Autogenous) Back Pressure Orifice
OPPS	Oxidizer Pressurant Pressure Switch
PDA	Pump Drive Assembly
PMDS	Pad Malfunction Detection System
POGO	Longitudinal Oscillation Peculiar to Titan II, which gave a Pogo Stick Effect
PSVOR	Pressure Sequencing Valve Over-Ride
RCA	Radio Corporation of America
RESS	Redundant Engine Shutdown System
RSP	Rate Switch Package
SA	Sub-Assembly
SCF	Sequential Compatability Firing
S/C	Spacecraft
SECO	Sustainer Engine Cut-Off
SMRD	Spin Motor Rotation Detection
S/N	Serial Number
SSD	Space Systems Division
STL	Space Technology Laboratories
TARS	Three-Axis Reference System
TCPS	Thrust Chamber Pressure Sensor
TCV	Thrust Chamber Valve
TPA	Turbopump Assembly
VTF	Vertical Test Facility

F. SYSTEMS PRODUCTION AND TEST

1. FACTORY ASSEMBLY AND TEST

a. Final Assembly

Final assembly, systems testing, and acceptance of the GLV took place in the Martin/Baltimore facilities. The propellant tanks, engines, guidance and range safety command subsystems were procured as described below.

The propellant tank and skirt assemblies for Stage I and II were manufactured, pressure tested, and calibrated at Martin/Denver and shipped by air or rail to Baltimore after Air Force/Aerospace review and acceptance.

The engines for Stages I and II were manufactured, hot-fired, and calibrated at Aerojet General, Sacramento and shipped by air to Baltimore after Air Force/Aerospace review and acceptance.

The guidance range safety command subsystems were also government-procured and delivered to Baltimore for installation and test in conjunction with the other systems.

Final assembly began with tank splicing and cleaning. Engine installation and buildup of hydraulic tubing and electrical harnessing followed. Equipment-truss, aerodynamic ducting and fairing and major component installations then took place. These were the autopilot, telemetry, malfunction detection, range safety, and guidance subsystems which were mounted on the equipment trusses.

Once fully assembled, and before vertical testing, the vehicle was subjected to tank and propellant feed system pressure leak checks, hydraulic leak checks, and electrical continuity and high voltage tests (DITMCO).

b. Vertical Testing

The vehicle was erected in two operations, Stage I and then Stage II at the Vertical Test Facility. This facility consisted of two vehicle cells and resembled the launch site test stand insofar as vehicle station work platforms and electrical and air-conditioning umbilicals were concerned. Two test cells were built because of an early program requirement for structural vibration testing which was later deleted.

Once the vehicle was erected, alignment and propellant sensor position checks were performed. Following detailed post-erection inspections of each subsystem for physical and mechanical integrity, power was applied to each subsystem singly through the use of power break-out tools rather than total vehicle power. This feature minimized the accumulation of power-on time and unnecessary component aging, and isolated power to the subsystem under test. It further reduced the possibility of widespread damage in the case of a defective component. The test cell had the capability of testing all vehicle functions short of engine fire-up. Ordnance for staging, liftoff, and range safety were simulated by fuse boxes.

Adjoining the test cell was a simulated vehicle launch control center using the same ground equipment, personnel functions, and procedures as those at ETR. Some several hundred yards away from the cell was a telemetry receiving station for r-f transmission through roof antennas and wave guides for instrumentation system check-out and test data collection.

After the subsystem tests, the total vehicle system was powered up for subsystems compatibility and performance. Final preparations were then made for the start of the Combined Systems Acceptance Test (CSAT). This was the final step prior to contractor presentation of the vehicle for Air Force Acceptance, and the culminating test in the progressive testing philosophy used throughout the program. The test schedule included an abbreviated launch countdown, engine start, and liftoff simulations, and flight, ending at Stage II engine shutdown (spacecraft insertion). Both primary and secondary (back-up) flight-control/guidance combinations were tested through the performance of two such test runs. Throughout the test, data was collected by AGE recorders and telemetry. This data was then reviewed by the contractor for vehicle acceptability. When launch vehicle integrity and performance were completely analyzed and assured, the vehicle was presented for final acceptance by the Air Force/Aerospace Vehicle Acceptance Team (VAT).

After acceptance, the vehicle was de-erected, re-inspected for final integrity, prepared for shipment, and flown to the Eastern Test Range (ETR).

The acceleration of the Gemini program resulted in the activation of the West Cell for functional test in order to reduce down-time. This meant that with two vehicles in the vertical position, one could be inspected and prepared while the other was under test. However, simultaneous testing was not possible since the two cells used the same power sources, AGE connections, and facility tooling. It was also decided that the Data Acquisition Test could be eliminated. The collection of instrumentation ambient data was substituted. Systems parametric variation and linearity analysis were usually accomplished after the CSAT.

Further, acceleration of the program lessened ETR capability to perform engineering modifications and other tasks which could be accomplished at the factory. Accordingly, modification periods for GLV's-4 through -12 were scheduled in VTF before subsystem tests to assure a finished and tested vehicle for delivery to ETR.

2. ETR TESTING

All vehicles were transported from Baltimore to the ETR by air. Shock and vibration instrumentation accompanied the GLV to assure that no environmental limits were exceeded during the flight.

When the vehicle arrived at the Cape, it was taken to a hangar in the industrial area for visual inspection and storage until preparation of Launch Complex 19 was complete. No functional testing of the vehicle was performed in the hangar, but certain packages were removed for laboratory tests. In the beginning of the test program, all packages containing gyro elements were removed; after the test program revision for GLV-5, only the TARS Package and the Stage II autopilots (rate gyro checks only) were removed for routine laboratory testing.

Erection of the vehicle was done in two stages, Stage I, first, then Stage II. Launch azimuth reference calibration and vertical alignment of the vehicle were done at this time.

Vehicle testing philosophy at ETR required first verification of subsystem operation, then the verification of all launch vehicle systems, and, finally, verification of all systems including the spacecraft, in a test simulating as closely as possible the launch and flight sequence of events. In this progressive buildup to launch configuration, a spacecraft simulator was used to check interface functions prior to actual mating with the spacecraft.

The test program at ETR started with the conservative approach, in that there was considerable redundancy of testing. For the last six vehicles, it was possible to simplify the test program resulting in the following sequence of tests.

- Subsystem Reverification Test (SSRT)

  Subsystem tests were performed to verify the readiness of the vehicle to begin system tests. These tests for the first four vehicles were the same rigorous tests used at the factory, but experience suggested that full confidence could be established with simpler reverification at ETR.

- Pre-Spacecraft Mate CST (PMCST)

  This was a Combined System Test, essentially the same as performed in the factory, and consisting of an abbreviated countdown and plus time simulation of flight events to establish confidence in the launch vehicle before interconnecting (mating) the launch vehicle and the spacecraft. A simulator was used to represent the electrical characteristics of the spacecraft.

- Electrical Interface Integrated Validation (EIIV)

  The purpose of this test was to confirm interface compatibility between the launch vehicle and spacecraft and to check out the redundant circuits connecting the interface.

- Joint Guidance and Control Test (JG&CT)

  This test established proper functioning of the secondary guidance system, and consisted of the spacecraft inertial guidance system and the secondary flight control system of the launch vehicle.

- Joint Combined Systems Test (JCST)

  This was the first system test using both the launch vehicle and spacecraft, and consisted of an abbreviated countdown and two plus-time simulations. One simulation exercised the primary guidance system, the other the secondary system.

- Launch Vehicle Propellant Loading Exercise

  In order to calibrate the loading instrumentation and to practice the procedures required to achieve the desired loading accuracy, a propellant loading exercise was performed. Prior to GLV-5, this was done as part of a complete countdown practice known as the Wet Mock Simulated Launch (WMSL). This change in the test sequence was made to accommodate the requirement for a simultaneous countdown practice with the Agena Target Vehicle.

- Simultaneous Launch Demonstration (SLD)

  Beginning with GLV-5, it was necessary to conduct a complete countdown exercise with the Atlas/Agena, the Range, and Mission Control Center. From the viewpoint of the GLV this was equivalent to the WMSL, except that it did not include tanking. The compromise of separating the tanking exercise from the countdown practice was made for the convenience of the program.

- Simulated Flight Test (SFT)

  For the launch vehicle this was a repeat of the JCST, but for the spacecraft it was a detailed mission simulation. This test established final readiness of the vehicle for launch.

- Launch

  The launch countdown followed the pattern established by the Titan family of vehicles. Fundamental to this operation was automatic monitoring of critical vehicle functions for hold and shutdown purposes during the last 35 minutes of the countdown, and the automatic sequencing of all events during the last three minutes of the count.

On the first two launch vehicles special tests were performed to identify any Electrical/Electronic Interference (EEI) problems. These were Combined Systems Tests with special instrumentation. After completion of the tests without evidence of trouble, it was considered that normal instrumentation would be adequate to identify any malfunction condition that would cause such interference.

GLV-1, as the first vehicle of the series, was subjected to even more special testing with the objective of validating the functional relationships between the vehicle, AGE equipment, and facilities. This included a static firing of both the Stage I and Stage II engines, known as a Sequence Compatibility Firing (SCF). For this test the two stages were mounted side by side. Subsequent to the SCF, the stages were mated in tandem and a Flight Readiness Firing (FRF) was conducted; during this test only the Stage I engines were fired.

One other special test, the Flight Configuration Mode Test (FCMT) was performed on the first six vehicles. This test was similar to other Combined Systems Tests, except that all umbilicals were dropped. It was then decided that the objectives of this test could be met by a normal umbilical drop during the premate CST.

Table II. F-1 shows in summary the manner in which the test program evolved as modifications were made to the content of the various system tests.

Table II. F-1. Summary of System Testing at ETR

Type of Test	Name	Vehicle											
		1	2	3	4	5	6	7	8	9	10	11	12
Subsystem Tests	SSFVT	X	X	X	X								
	SSRT					X	X	X	X	X	X	X	X
Pre-Mate Systems Tests	PMCST	X	X	X	X	X	X	X	X	X	X	X	X
	EEI	X	X										
Integrated Tests	EEI	X	X										
	AST	X											
	EIIV		X	X	X	X	X	X	X	X	X	X	X
	JA & C		X	X	X	X	X	X	X	X	X	X	X
	JCST		X	X	X	X	X	X	X	X	X	X	X
	FCMT	X	X	X	X	X	X						
Countdown Practice	WMSL		X	X	X	X	X						
	Tanking							X	X	X	X	X	X
	SLD					X	X		X	X	X	X	X
Final Status	SFT	X	X	X	X	X	X	X	X	X	X	X	X

NOTE: Does not include SCF Testing of GLV-1

Figure II. F-1 shows the overall operating times, in hours, for each vehicle. The operating times are more representative for checkout of the vehicle at the VTF than at the ETR since the ETR times include providing support for the spacecraft when testing was not necessarily required for the GLV.

During the testing of the GLV's at ETR, the flexibility and versatility of the test program were demonstrated many times. One outstanding example of this capability was the Gemini 7-6 mission.

In accordance with prelaunch plans, the launch of GLV-6 was postponed when the Agena target Vehicle failed to achieve orbit, and the contingency plan was put into effect. This called for the rendezvous of two spacecraft and the combination of a long duration mission with an accelerated launch cycle for the second spacecraft.

Since GLV-6 had been completely checked out on the launch pad, this approach appeared feasible. The vehicle was removed from the launch pad and put in humidity controlled bonded storage, with the expectation that revalidation after re-erection could be accomplished with the following tests:

- Subsystem Reverification Testing
- Abbreviated E I I V
- S F T

It was estimated that with an accelerated schedule, this testing could be done in seven days. Since the GLV-7 mission was planned for 14 days, this allowed a reasonable amount of time.

GLV-7 was erected on 29 October and was launched only 36 days later on 4 December. To achieve this schedule, the FCMT and WMSL were deleted in accordance with previous recommendations. Since no dual launch with the Atlas Agena was planned, the SLD was also deleted.

Pad damage after the launch of GLV-7 was minimal, so that GLV-6 could be re-erected on 5 December, the day following launch of GLV-7. Revalidation testing on GLV-6 progressed according to plan, with the launch countdown on 12 December. This attempt was aborted due to the premature disconnect of umbilical 3D1M, which caused engine shutdown.

The launch was recycled and successfully accomplished on 15 December. This streamlined operation was possible because of the flexible approach to mission objectives, permitting constant review and refinement based on test experience, and because of contingency planning of vehicle checkout requirements.

3. MODIFIED TEST PROGRAM

Toward the end of 1964 a comprehensive review of the test program was started, with the object of simplifying the program wherever possible. It appeared desirable from a program standpoint to accelerate the launch rate to 60-day intervals rather than the existing plan of approximately three-month intervals. Also, factory test experience on four vehicles had been accumulated and it seemed feasible to eliminate some repetitive testing and make other changes that would ease the test burden of the launch operation.

The target for implementation of the new test program was established as GLV-5, the next vehicle coming out of the factory for acceptance testing.

Early in the investigation, it became obvious that in order to support the faster launch rate, it would be necessary to activate the other vertical test cell since the factory sequence of testing and acceptance exceeded the two-month interval. This was done in such a manner that most of the AGE equipment could support either cell with only transfer of umbilicals required.

Major changes to the test program made as a result of this study are as follows:

Changes at the Vertical Test Facility in Baltimore:

- Deletion of the Data Acquisition Test (DAT): this test was largely redundant and had been included originally to insure readiness for formal system test.

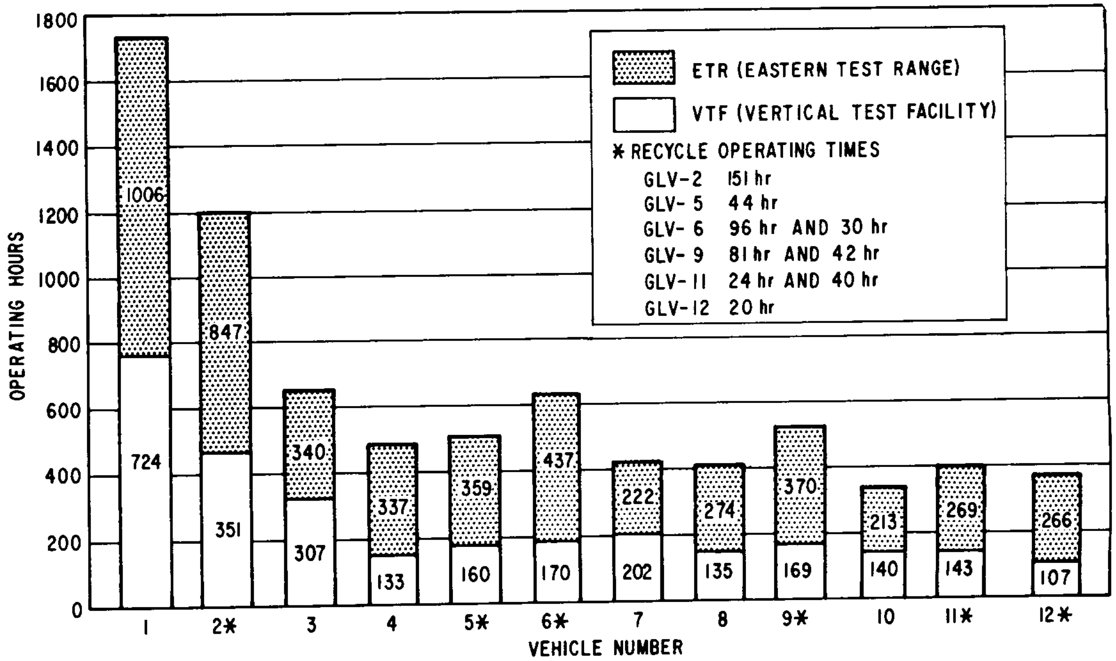

Figure II. F-1. Gemini Launch Vehicle - Operating Time Data

- Addition of a staging test to eliminate the requirement for doing it at ETR.
- Perform certain operations before erection into the test cell rather than after removal from the test cell. These operations were: leak checks, external marking on vehicle, and moisture proofing.
- Retention of the vehicle after acceptance, in the vertical test cell for purposes of retest in the event of required modification.
- Combination of the Vehicle Acceptance Review (VAT) and the "Rollout" physical inspection into a single exercise.

Changes at the ETR Launch Facility:

- Simplification of subsystem testing - Initially, subsystem testings at ETR were essentially the same as at the VTF. As a result of considerable simplification, the new tests were designated Subsystem Reverification Tests (SSRT).
- Elimination of electronic interference tests. On the basis of GLV-1 interference tests, it was considered sufficient to monitor other test data for the effects of interference.
- Elimination of the Flight Configuration Mode Test (FCMT). This was a redundant system test performed to drop umbilicals. After considerable analysis it was considered adequate to perform the umbilical drop as part of the pre-mate CST and eliminate the FCMT.
- Removal of components for periodic laboratory testing was reduced.

In addition to the changes made, several operational principles were emphasized as being essential to success in this acceleration of the program.

- All vehicle modifications should be completed and retest accomplished at the factory.
- All VTF testing should be done using flight hardware.
- All components should be shipped/installed on the launch vehicle.
- VTF test data should accompany the vehicle for reference by the launch operations group.

4. TEST CONSISTENCY

Within the basic concept for the Gemini Program, provisions were made for assuring continuity of vehicle checkout between the contractor's Vertical Test Facility (VTF) and Complex 19 at the Eastern Test Range (ETR). Individual items of Aerospace Ground Equipment (AGE) that were provided for the VTF were functionally identical with items provided at Complex 19 and identified by the same control point number. The vehicle support and test configuration were identical at both locations.

Launch vehicle integration tests performed at the VTF included performance of subsystem functional verification tests and combined systems tests similar to those conducted at ETR prior to launch. By duplicating the ETR checkout tests as much as possible during in-plant testing, a baseline was established early in the history of testing and carried throughout the program. Reference to this test baseline provided a systematic evaluation of testing both at ETR and at the contractor's facility. This approach was chosen to assure an adequate, unified, and coherent test program.

5. TREND DATA MONITORING

One significant item unique to the Gemini Program which insured consistent test results, was the use of trend data recorded during testing at the contractor's facility and compared to data gathered at ETR on a test-to-test and vehicle-to-vehicle basis. Trend data monitoring was the observation of selected parameters monitored at regular intervals during normal testing. This data was used to (1) express the normal operating drift and test-to-test characteristics of equipment for establishing realistic operating behavior and tolerances as a function of time and (2) identify incipient failures or engineering faults and allow replacement if necessary at a convenient point in the testing cycle.

A total of 13 hardware items were removed from vehicles due to data trend and, at other times, special tests were conducted which removed any doubt indicated by the trend. In such cases, the history of the unit (or parameter), as indicated by all previous testing on earlier vehicles, was researched and considered prior to package replacement. A typical data trend chart for the electrical system is shown in Table II. F-2.

The GLV data trend monitoring program has been of particular significance on two occasions: when GLV-2 was exposed to an electronic phenomenon and after de-erection and re-erection for a hurricane at Cape Kennedy. After the first occasion, a number of electrical and electronic components in both the AGE and airborne areas, some of which were known to be damaged and others which were thought to have been degraded due to over-voltage stress, were replaced. During subsequent retesting, an even more comprehensive data trend monitoring program was implemented to ensure that the integrity of the launch vehicle had not been impaired due to the prior events. All test data were reviewed and any peculiar or abnormal indication, or any data point falling in the last 20 percent of the tolerance band, was cause for a comprehensive review with hardware troubleshooting as required.

After the GLV-6A storage period at Cape Kennedy and prior to launch, all testing data was reviewed in a similar manner. Additionally, a digital computer program was used to print out the Simulated Flight Test (SFT) data points which differed between the pre-storage and post-storage SFT's by more than 60 mv. All such differences were reviewed and signed off when investigations were completed. Another benefit of data trend monitoring was the removal of the roll bias for GLV-9 and up. This was based on the data of the previous eight flights which reflected a slight thrust vector misalignment.

It is believed that the data trend monitoring program added materially to launch confidence by adding an extra dimension to test data analysis.

6. TEST CONTROLS

The basic method for controlling the testing conducted on the GLV was the contractual documentation that was implemented early in the program. This documentation encompassed all of the testing requirements, test description and definition, test location, sequence of testing, test configuration, interface requirements and definitions, and retest requirements for components, subsystems, and systems. These documents and their control were used by management as tools to prevent unauthorized changes to the test programs at the contractor's facility and at ETR. The following is a list of some significant test control documents and their contractual application:

- Type I Documentation - Required to be approved by contracting officer and negotiated as required to a mutual agreement and formally made a part of the contract.

Document	Prepared By
MB-1042, GLV System Test Specification	Contractor
ISCD-1, Gemini/Spacecraft Interface Specification	NASA
SSD Exhibit 62-195, GLV Pilot Safety Program	SSD/Aerospace
TOR-169(3126)-16, GLV Acceptance Requirements	Aerospace

Table II. F-2. Gemini Laun

				VIF TESTS	
Line No.	Meas No.	Parameter	Spec or Nominal Value And Tolerance	CSAT Date 6-25-65 Test No. 011/012	
				1	2
1.	 0800 0804	PS940300011 IPS Battery Volts IPS Battery Amps	27 to 31 VDC	-001 29.1 29.9	-001 29.8 26.9
2.	 0801 0805	PS940300011 APS Battery Volts APS Battery Amps	27 to 31 VDC	-001 29.7 34.3	-001 30.1 28.0
3.	 0802 0803	PS946000001 Static Inv Volts Static Inv Freq Serial Number	113 to 117 VAC 396 to 404 CPS	-007 114.3 399.4 R31	-007 114.4 399.4 R31
4.	 0726	CCI 9401A11 PWR Supply 25 VDC Serial Number	24.1 to 25.9 VDC	-1 25.1 170	-1 25.2 170

NOTES: *392.1 - 399.4 Variation--Substitute Access Doors Installed.
**Vehicle Access Doors Not Installed.

:h Vehicle No. 6 Trend Data Monitoring Electrical System

ETR TESTS							
Pre-SC Mate Date 9-16-65 Test No. 5547		EIIV (ETR) Date 9-20-65 Test No. 5750	JCST Date 9-23-65 Test No. 5751	FCMT Date 10-1-65 Test No. 5901	WMSL Date 10-7-65 Test No. 6000	SFT Date 10-20-65 Test No. 6260	
1	2	1	1	1	1	1	2
NICAD	NICAD	NICAD	NICAD			-001	-001
28.5	28.3	29.0	28.2	29.0	29.0	29.0	29.0
25.9	25.9	29.9	27.9	29.9	25.4	28.9	29.9
NICAD	NICAD	NICAD	NICAD			-001	-001
28.7	28.5	29.8	29.0	29.7	29.9	29.9	29.7
26.9	26.9	25.2	27.3	27.3	24.2	24.2	28.3
-007	-007	-007	-007	-007	-007	-007	-007
113.7	113.7	113.9	113.5	113.5	113.8	113.9	113.9
NO DATA**		399.8	399.4	397.7*	398.8	400.7	400.5
R31	R31	R31	R31	R31	R31	R31	R31
-1	-1	-1	-1	-1	-1	-1	-1
25.3	25.1	25.1	25.1	25.1	25.1	25.2	25.2
170	170	170	170	170	170	170	170

- Type IA Documentation — Documentation required to be submitted for unilateral approval of the procuring agency.

Document	Prepared By
424-1715007, GLV AGE Systems Test Specification ETR/VTF	Contractor
424-1020002, Launch Vehicle Acceptance Test Specification	Contractor
424-1430002, GLV Test and Checkout Specification (VTF and ETR)	Contractor

Supporting this effort were the Factory Working Group at the contractor's facility and the Gemini Launch Vehicle Working Group at ETR who reviewed all test efforts. In addition, the Vehicle Acceptance Team reviewed all of the vehicle test history and results prior to accepting the vehicle for shipment to ETR.

At ETR, the Pilot Safety Team was responsible for reviewing all test procedures and test procedure revisions prior to the actual use of the procedure for each test as well as also reviewing test results. Approximately two to three weeks prior to launch the Aerospace Program Office conducted "Scrubdown" meetings with all GLV contractors to review all testing, test results, problems, corrective action taken, and status of open problems and corrective action required. (See Section II. B.6 of this report.)

The final test control effort culminated in an SSD/Aerospace recommendation that the GLV was ready to launch at the Flight Safety Review Board Meeting. This assured the board that both SSD and Aerospace were satisfied that the testing and test results were not only successfully completed, but also were in accordance with the specifications and documentation required.

SECTION II. F

REFERENCES

None

SECTION II. F

DEFINITIONS

AGE	Aerospace Ground Equipment
CSAT	Combined Systems Acceptance Test
DAT	Data Acquisition Test
EEI	Electrical/Electronic Interference
EIIV	Electrical Interface Validation
ETR	Eastern Test Range
FCMT	Flight Configuration Mode Test
FRF	Flight Readiness Firing
GLV	Gemini Launch Vehicle
JCST	Joint Combined Systems Test
JG and CT	Joint Guidance and Control Test
PMCST	Pre- (Spacecraft) Mate Combined Systems Test
SCF	Sequence Compatibility Firing
SFT	Simulated Flight Test
SLD	Simultaneous Launch Demonstration
SSRT	Subsystem Reverification Tests
TARS	Three Axis Reference System
VAT	Vehicle Acceptance Team
VTF	Vertical Test Facility
WMSL	Wet Mock Simulated Launch

G. LAUNCH VEHICLE HISTORIES [1, 2, 3]

1. GLV-1 THROUGH GLV-12 SUMMARY

Figure II-G-1 presents the launch vehicle histories in chart form. Significant events at each phase in vehicle assembly, test, checkout, and launch are shown. In addition to major hardware and testing problems, management highlights are presented.

2. SIGNIFICANT ITEMS AND TRENDS - VEHICLE AS A WHOLE

a. Flight Configuration at Combined Systems Acceptance Test (CSAT)

A major objective in the factory test program was a launch vehicle in as complete a flight configuration as possible at time of final system acceptance test. This meant not only engineering modifications (ECP's) but corrective actions (CAD's) resulting from problem resolutions, qualified and flight-worthy components, minimum "recap" tasks for ETR accomplishment, closed discrepancy reports, failure analyses, and hardware waivers.

The degree of success in attaining this objective can be seen on Figures II.G-2 and II.G-3. In reviewing Figure II.G-2, however, it should be understood that a number of the hardware modifications (ECP's) were not known at time of the combined systems acceptance test. The solid curve II.G-3 shows ECP's accomplished at ETR prior to launch. The dashed curve represents numbers of approved ECP's at the time of vehicle acceptance. The difference between these curves represents the fact that the factory schedule was too rapid to allow incorporation of all changes prior to delivery.

The large hump on GLV's 3 and 4 in the dashed curve (Figure II.G-3) indicates a period of schedule pressures and a reluctant acceptance that ECP's had to be accomplished in the field. It is significant to note however, the very considerable improvement with GLV-5 and subsequent vehicles as a result of management attention to this area.

The obvious advantages of a flight configuration at final factory acceptance test are as follows:

1) Reduced manpower levels in field
2) Reduced time for launch preparation and checkout
3) Identification of problems for design engineering resolution at the factory
4) Earlier identification of problems
5) Higher confidence in vehicle reliability

b. Component Failures in Test at Factory and at Launch Site (ETR)

Figures II.G-4 and II.G-5 show numbers of significant components replaced for functional failures at the factory in Baltimore during testing and at the launch site at ETR during pre-launch checkout. Failures were defined to include:

(1) Out-of-Specification Performance

Any indication through telemetry, ground instrumentation, or AGE that a component was out-of-specification required replacement.

(2) In-Specification Performance But Degrading

One aspect of the man-rating concepts for Gemini was the recording of certain critical parameters at the various testing points such as vendor test, receiving test, subsystem and system functional tests at both VTF (Factory) and ETR (Launch Site). Systems engineers maintained continuous surveillance of these parameters. If there was an indication of degrading performance, the component or components were removed for failure analysis. Refer to the Trend discussion in Section II-F, 5.

(3) Transient Malfunction

This is defined as an apparent or suspected failure or anomaly occurring during system of component test which could not be repeated or safely explained and required replacement and failure analysis at bench level.

(4) Questionable History

Detailed logs were maintained on some eighty critical components. These logs were referred to as Data Packages and contained total test and calibration histories of each critical component. The packages were reviewed in considerable detail at several selected points in vehicle assembly, test, and checkout. When a question arose in these reviews that reflected doubt on component reliability, the part was replaced and reworked or in some cases scrapped.

With these criteria in mind, and noting that there were no component failures in flight, it can be concluded that the Gemini test and checkout philosophy contributed significantly to vehicle reliability. Refer to the discussion of the component Data Packages in Section II-B.

VEHICLE NUMBER	DENVER	HORIZONTAL
GLV–1	STG I Ox,dome patched (20 in. x 4 in.) to remove "oil-canning" resulting from weld repairs STG II Fuel, weld porosity cleared by stress analysis BALTIMORE DELIVERY: 10 Oct 62 (STG II Ox) : 1 Mar 63	STG II Ox ret'd to Denver replaced with tank; cracks in "Y" plates Dummy Eng used for tube and wire run d Flt Eng installed in May 63 Clamps discovered to be damaging to w replaced by "Thomas" clamps VERTICAL DELIVERY: 9 June 63
GLV–2	STG I Ox,rejected for crack in aft dome and inability to x-ray forward dome STG II Ox, rejected for damaged "Y" chord which was not repairable STG II Fuel,rejected for heat treatment cracks, new tank built Four add'l tank pressure cycles and 100% x-ray, new reqt GLV-2 and subs BALTIMORE DELIVERY: 12 Jul 63	All connectors inspected for contamina headers Attenuator pad wire nicking from steel blocks,teflon inserts were required, 9 External harness ladder fasteners misa replaced, tooling corrected Teflon coated wire longitudinal splitti screenable using hi-pot insulation resi: Interim STG I engine S/N 1002 installe serviceable one available
GLV–3	STG I Fuel,original build, aft dome failed unaer proof pressure,hand finished area 50% of B/P thickness; new build req'd STG I Fuel,truss misalignment, tooling, processes and engineering revised Tungsten inclusions in welds, industry survey and stress study Significant improvement of weld quality over GLV–1 and 2, weld acceptance criteria revised to meet man-rating reqts BALTIMORE DELIVERY: 13 Dec 63	3-dimensional harness assembly proce: meet configuration updates Propellant feedline and conduit x-raye lation, new x-ray reqts were too late f x-ray VERTICAL DELIVERY: 22 Jun 64

BALTIMORE				
VERTICAL TEST	CSAT/VAT	POST–VAT ACTIVITY	OPEN ITEMS AND WAIVERS ETR SHIPMENT	
contamination in hyd lines, all tubing replaced ir recleaned ontamination into system from AGE, N_2 system tter power amplifier tube problem, rework with to solder flux ronic Interface Tests resulted in rerouting, g of leads and components in AGE, SMRD, GS, MISTRAM, MDS, Secondary Autopilot, drop tests run for sequencing 4 hrs	CSAT 1 6 Sep 63/VAT 1 11 Sep 63 Connector contamination and cracked inserts Gyro cleanliness standards rigidized Autopilot SMRD malfunctions Pressure xducer torque sensitivity Attenuator pad wire nicks Tape recorder noisy Static inverter vibration integrity MARS and failure analysis procedures 2nd CSAT directed after Mod incorp CSAT 2 4 Oct 63/VAT 2 8 Oct 63 Airborne tape recorder noisy FM/FM subcarrier noise MISTRAM unlocks Guidance phase detector out of spec Rerun program seq tests/CSAT calib questionable VAT WALKAROUND SQUAWKS: 20	No significant tasks performed, vehicle expedited to ETR: schedule pressures	Shortages: PCM/FM xmitter-power amplifier tube mod FM/FM xmitter-power amplifier tube mod RGS airborne components Landline instrumentation, J-boxes Waivers: Ox feedline proof test reqt Defective components: 3 pressure xducers: calib recheck Airborne tape recorder noisy 16 defective connectors Unqualified components: 18 Recap tasks to ETRL: 60 GPIS open: 41 ETR DELIVERY: 26 Oct 63	Insul and m Stage idler Engin fir-tre Engin warpe Engin spike RSP PCM contr Exce prear Rate, and c contr OPP
corp ECPs 126R1, 127, 145, 181R1 and 135 veloped to isolate voltage spike from autopilot ulted in Mods to PCM encoder, AGE grounding, ated from voltage spikes, new torquer M dropouts 17 Apr 64 ABETS s damaged by Ditmco test 51 hrs	CSAT 22 Apr 64/VAT 27 Apr 64 MDS RSP replaced, retested during VAT RFI test on range safety system, destruct switch traces showed spikes EEI rerun for spacecraft interface Test personnel changes reviewed: OK System tolerances reviewed: some changes Data documentation reviewed: some changes VAT WALKAROUND SQUAWKS: 10 Eng injector holes plugged	EEI date indicated erratic instrumentation J-box capacitor failure, replaced Hyd disconnects found nicked by ground half disconnects, replaced STG I Eng SN 1002 (interim installation) replaced by SN 1003 10 Airborne ECPs installed	Shortages: RGS airborne components Waivers: STG I and II conduit x-ray requirements STG I Ox dome and fuel cone x-ray requirements Defective Components: Temp xducer calib questionable Connector integrity questionable Unqualified components: 13 Recap tasks to ETR: 41 GPIS open: 24 ETR DELIVERY: 11 Jul 64	Ligh comp Hurri Hurri Stag Stag and Stag 17 A MDT FM/ Viol Hydr mour
sition tests directed ures, replaced. , corrosion pitted nadvertently with APS de-energized, ECP fix te 42 ECPs d for out-of-tolerance e out-of-spec 214 (thru CSAT 1) 92 (thru CSAT 2) 306 hrs	CSAT 1 7 Aug 64/VAT 1 17 Aug 64 Strain gage open circuited Airborne tape recorder failed playback Electric hyd pump "start" data drop-outs 13 phase-sensitive demodulators changed, procedure error CSAT 2 30 Sep 64/VAT 2 7 Oct 64 Airborne tape recorder failed playback Autopilot end-to-end tolerances vs individual component tolerances studied Destruct primer simulator failed, loose connector pin in test tool "Gold Flake" diodes to be purged Solder-Ball contaminated switchover relays to be purged VAT WALKAROUND SQUAWKS: 17	VAT ROLL-OUT Inspection: 27 Oct 64 Roll nozzle (STG II) replaced, weld defect Turbine manifold replaced with hot-fired manifold STG I Ox tank support structure disassembled and cleaned of metal chips Coaxial connector problems cleared 16 add'l ECPs incorporated 6 CADs accomplished	Shortages: MDS (for ECP Mod) 8 Eng components (GG and GC systems) RGS airborne components Waivers: None Defective Components: None Unqualified Components: 13 Recap tasks to ETR: 29 GPIS open: 22 ETR DELIVERY: 23 Jan 65	Stag Tan Tan Eng bec Stag TA R–

ETR		
SIGNIFICANT EVENTS	FLIGHT ANOMALIES AND RESOLUTIONS	FLIGHT OBJECTIVES AND RESULTS
ation flaws in teflon electrical wiring, stock purge ore rigid controls I turbopump assemblies sent to Sacramento for gear rework ie turbopump seal leak, improper shim during rework; e braze cracks found on rotor ie turbopump, breakaway and running torque high, d manifold e captive firings; hydraulics showed severe pressure s during engine start cross-coupling problem, modified with solid state switches multiplexer, encoder problems; multiple failures, procurement ols reviewed and new capacitor switches added ssive noise on VDA due to 1600 cps signals saturating np, filtering added gyro run-up time excessive, lack of bearing preload controls ontamination; procurement controls reviewed, and preload ols instituted S late "MAKE" caused by orifice size	Post-SECO Oscillations Delay of 2 seconds in the initiation of OAMS rate control OAMS rate control by change of astronaut's procedures for S/C separation Fuel Sensor Recovering A hooded screen was designed to collect the residues and prevent recovering of the optics POGU pot malfunction Xducer was redesigned Yaw velocity error at insertion Aerospace modified the guidance equations to bias out the effects of vehicle cg shift	Evaluate GLV performance capability and structural integrity, qualify for manned flight GLV–1 Flight was highly successful in placing the Gemini spacecraft into required orbit, countdown was completed with no holds; all systems performed satisfactorily and the trajectory was within the predicted dispersions LAUNCH DATE: 8 Apr 64
tning strike: Resulted in extensive investigation, onent change out, and systems re-validation icane "Cleo": De-erected Stage II icane "Dora": De-erected Stage I & II e II turbine rotor corrosion e I Ox Tank Entry for metal particle contamination on 20 Oct on 5 Dec for piece of missing teflon from Stage I ox valve e II Ox Tank Entry for possible nitric acid contamination ug CPS "A" failures due to procedural errors FM xmitter center frequency problems ent booster engine movement due to IGS computer problem aulic actuator failure causing launch abort-servo-valve body nting lug failure	Stage I Hyd pressure decay at Eng start. Eng shutdown Gauss meter testing of hyd pump compensator stem and forward spring seat at zero system pressure to assure proper performance prior to launch. Higher than predicted thrust and ISP, see GLV–3	Evaluate systems performance; in particular: MDS, structures, secondary flight controls, and post-SECO motions All systems performed as planned; trajectory was within planned dispersions; minor anomalies occurred, but did not adversely affect the flight Pad shutdown required redesign of Stage I Hyd actuator primary servo valve housing LAUNCH DATE: 19 Jan 65
ge I Actuator rigging errors (3 actuators) dem Actuator 3_1 position xducer failure dem Actuator 4_1 null offset and slow response ine driven hydraulic pumps replaced for all subassemblies ause of suspected contamination ge I engine ox prevalve metallic contamination RS Package failed stiction tests S receivers delayed response to engine shutdown command	Higher than predicted vehicle thrust and ISP Investigation led to the utilization of empirical data from Titan II and GLV flights to more accurately predict engine performance	First manned flight, confirm GLV systems performance required to to place spacecraft into prescribed orbit, confirm performance of RESS and MDS The "Molly Brown" was successfully placed into the desired orbit, all systems performed satisfactorily; vehicle thrust and ISP were slightly larger than desired, but did not adversely affect the successful flight LAUNCH DATE: 23 Mar 65

Figure II. G-1. Gemini Launch Vehicle Histories

VEHICLE NUMBER	DENVER	HORIZONTAL	VERTICAL TEST
GLV–4	First tanks receiving high-pressure/leak sniff tests STG I Ox, aft dome wye chord erroneously installed as forward dome chord,accepted; however small weight penalty STG I Fuel, 2 weld cracks after hydro tests, repaired STG II Ox, crack in forward and Aft domes after hydro test, repaired BALTIMORE DELIVERY: 6 Mar 64	Phycomycetes mold growth on fwd skirt Stage I Fuel, probably deposited by workman's hand Eng xducers out-of-calib period, directed to proceed thru CSAT then calibrate STG I Ox tank support structure disassembled and cleaned of metal chips VERTICAL DELIVERY: 28 Oct 64	Gold plating on destruct receiver connecto Coax cable teflon insulation cut thru to sh MDS cable low resistance problem TARS pkg below configuration, directed to at ETR 2 Hyd Pump drain plugs "frozen" POWER-ON TIME: 133 hrs
GLV–5	STG II Fuel,cracked at cup weld,repaired STG II Fuel,cracked at four welds, repaired STG II Ox,sensor bkts mislocated, special fix BALTIMORE DELIVERY: 25 Jun 64	Skin corrosion, STG I outer skin, iridite, and procedures reviewed STG I Ox Fwd dome support disassembled and cleaned of metal burrs Harness configuration not compatible V-Band clamps general nick problem Mod period to incorp 15 ECPs (Pre-VTF) VERTICAL DELIVERY: 8 Feb 65	VTF procedures and criteria changed to m at Factory", philosophy STG II mated with STG I above normal ra crane controls too insensitive, hyd jack u Gyro overheat indication attributed to AG Loose potting in MDTCPS connectors Mod period to incorp 6 ECPs (Pre-CSAT) 4 CSAT attempts 15 Apr – 20 Apr 65 AGE, procedure and operator problems,M VPRF POWER-ON TIME: 160 hrs
GLV–6	STG II Fuel,2 cracked welds, repaired STG I Ox,weld defect caught at roll-out inspection, repaired BALTIMORE DELIVERY: 16 Aug 64	STG I dome support brackets disassembled and cleaned of metal chips Drain holes relocated on STG II Ox,drawing error Tandem actuator servo body heat treat Autopilot adapter circuit board wires not crimped VERTICAL DELIVERY: 15 Apr 65	First vehicle in VTF new West Test Ce Ditmco test damaged 25v power supply, Insert initiator safe-to-shortoff problem STG I Ox conduit bellows wall thickness CSAT attempt no. 1 25 Jun 65 MISTRAM false locked for 6 min., replac POWER-ON TIME: 170 hrs
GLV–7	STG II Fuel,dome/cone replaced for excessive weld defects and repairs 100% audit performed on STG I Ox welds and x-rays STG I Ox,first under new cleanliness reqts, result of dome support metal chips C–124 aircraft used first time to Baltimore BALTIMORE DELIVERY: 25 Feb 65	STG I Ox dome support structure reworked to obtain stringer clearance STG II Ox drain holes relocated, drawing error STG II Eng turbo-pump impeller; pit-corroded Mod period to incorp 19 ECPs VERTICAL DELIVERY: 28 Jun 65	Mod period: Zyglo'd TCV installed in engs Replace damaged guidance waveguides Incorp Flashing Beacons, STG II POWER-ON TIME: 202 hrs
GLV–8	STG II Ox, 2 weld cracks after hydro, repaired BALTIMORE DELIVERY: 15 Apr 65	VERTICAL DELIVERY: 28 Sep 65	2 ECPs installed New reqt to calibrate propellant level se Eng electrical connectors pull test Several FPDPS switch failures POWER-ON TIME: 135 hrs

BALTIMORE				
	CSAT/VAT	POST-VAT ACTIVITY	OPEN ITEMS AND WAIVERS ETR SHIPMENT	
r pins scraped back ield proceed with replacement	CSAT 25 Nov 64/VAT 11 Dec 64 Excessive test operations errors reviewed Accuracy and completeness of critical component data pkgs reviewed Spacecraft-to-launch vehicle scupper seal alignment procedures to be definitized Open ECPs and CADs scheduled post CSAT, criticized VAT WALKAROUND SQUAWKS: 11	Coax cable connector pin problem Destruct RCVR wrong tuning slug STG II Hyd disconnect leak, metal sliver, replaced Pictures taken of eng injector plates Level sensors removed for ECP Mods 27 Airborne ECPs installed, 7 CADs accomp	Shortages: 22 level sensors, Mod for shields RGS airborne components Waivers: Fuel accumulator, eng eval autogenous line interface tolerance Defective components: 4 Marman clamps, nicked and not fuel compatible Unqualified components: 3 Recap tasks to ETR: 32 GPIS open: 25 ETR DELIVERY: 23 Mar 65	TARS Packag Hydraulic res FM/FM xmitt Mistram xpon 8 B-nut leaks S/A-1 PSVO PTPS Circui Flowmeter ac
ieet "Flight Configuration te damaging aligning pin holes, sed in future E ISTRAM replaced for increasing	CSAT 21 Apr 65/VAT 26 Apr 65 IPS overvoltage alarm due to removal of FM/FM system and lower load AGE calibration procedures reviewed Vehicle stability margins requested Add'l testing to confirm liftoff-lite anomaly VAT WALKDOWN SQUAWKS: 24	In-configuration propellant level sensors installed In-configuration TARS installed Time delay relay replaced STG II Ox pressure xducer chgd, loose receptacle 4 Airborne ECPs installed 2 CADs accomplished	Shortages: None Waivers: None Defective components: Marman clamps nicked Unqualified components: 2 Recap tasks to ETR: 17 GPIS open: 36 ETR DELIVERY: 18 May 65	Review of da Erector failur control syster MOCS Sequen OPPS Diaphr 4 Tank press TARS Failed Tandem actua switches Hyd accumula
ll est configuration revised low ed	CSAT 25 Jun 65/VAT 7 Jul 65 Baltimore gas and electric power disturbance caused airborne power bus drops Eng quality condition reviewed CSAT considered well done Configuration audit: clean VAT WALKDOWN SQUAWKS: 10	Nicked marman clamps replaced In-configuration level sensors installed Thrust chamber valves replaced STG II Eng lube oil reservoir dye-penetrant 0 Airborne ECPs installed 1 CAD accomplished	Shortages: None Waivers: None Defective Components: None Unqualified components: None Recap tasks to ETR: 12 GPIS open: 32 ETR DELIVERY: 2 Aug 65	Engine ablati TCV Bolt ch Gas generato Broken stranc CVE malfunc Premature se second launc Decoder: sep
	CSAT 20 Sep 65/VAT 28 Sep 65 4 ECPs not affecting CSAT results performed pre-VAT Delamination of fire protection wrap-tape Air Conditioning for VTF caused airborne bus power fluctuations Vehicle very clean VAT WALKDOWN SQUAWKS: 10	Repaired damaged autogenous tubes 3 Eng connectors defective, replaced 0 Airborne ECPs installed 1 CAD accomplished	Shortages: None Waivers: Hyd Reservoir PMT vibration time Defective components: None Unqualified components: None Recap tasks to ETR: 10 GPIS open: 32 ETR DELIVERY: 19 Oct 65	Erector probl Signal condit Double interr on high powe instituted TCV Bolt ch
nsors	CSAT 8 Nov 65/VAT 16 Nov 65 Adapter cable omitted during CSAT Configuration audit showed good Vehicle deemed excellent WALKDOWN SQUAWKS: 10	Eng Pump discharge lines cleaned Natorq seal faulty, replaced STG II Eng roll nozzle bearing scored 2 Airborne ECPs installed 2 CADs accomplished	Shortages: None Waivers: None Defective components: None Unqualified components: None Recap tasks to ETR: 14 GPIS open: 22 ETR DELIVERY: 6 Jan 65	Oil flush hos Faulty Stage chamber tube Erector Prol box contamir Spacecraft/t Hydraulic pu

ETR		
SIGNIFICANT EVENTS	**FLIGHT ANOMALIES AND RESOLUTIONS**	**FLIGHT OBJECTIVES AND RESULTS**
.e problems ervoir N_2 leakage into fluid chamber ər frequency stability ders, numerous unlocks in engine R failure curacy problems	Roll transient at liftoff due to fuel-topping disconnect hangup Due to dust plug stowage position: dust plug assemblies will be removed prior to liftoff High Stage II fuel orifice inlet temperatures during flight Leak in the hot gas cooler, ECP prepared for installation of an internal bypass cooler	Evaluate launch countdown time and procedures for applicability in support of a rendezvous mission Mission anomalies did not adversely affect the success of the flight, erector malfunction delayed launch 1:15 min LAUNCH DATE: 3 Jun 65
mage controls between S/C and LV e during special flight crew egress, erector m refurbished, cycling tests instituted cer encoder failure during FCMT agm gold plate flaking problems ıre switches out of tolerance solder balls in capacitor ıtors replaced, suspected defective pressure ıtors replaced, possible contamination	POGO oscillations during Stage I flight Oxidizer standpipes were uncharged at liftoff, procedural changes were made	Demonstrate satisfactory performance of GLV to place a manned spacecraft into a prescribed orbit All vehicle systems performed as required, POGO oscillations occurred due to uncharged oxidizer standpipes prior to liftoff, these oscillations had little effect on on successful outcome of the flight LAUNCH DATE: 21 Aug 65
ve skirt wrinkle (soft spot), epoxy-filled ange: cracked bolts found in T-III : dust cap cover incident ls on CVE cable, cleared OK tion during SLD/Wet Mock, motor contact assembly paration of tail plug caused engine shutdown on h attempt (12 December 65) ıaration of capacitor plate from ceramic dielectric	Premature disconnection, umbilical 3 D1M, engine shutdown Suspect improper installation, metal fairing was cut for inspection, lock wire and index marks added Thrust decay on SA-2 prior to engine shutdown Plastic dust cover in gas generator injector T/M signal strength drop, Stage II Flight Antenna deformation aerodynamic heating, addition of antenna stiffeners GLV-7 up	Place spacecraft into orbit for rendezvous with Agena Agena failed to achieve orbit, GLV-6 launch rescheduled after GLV-7, rendezvous with GLV-7 was planned, premature disconnect of 3-D1M disconnect caused shutdown, GLV-6 rescheduled 3 days later, launch on schedule and rendezvous with GLV-7 ok LAUNCH DATE: 15 Dec 65
em, wiring discrepancy in hold-in relay ioner rejected, verdigris contamination ogation of Pulse Beacon; Mod III performed radiation test r during telemetry calibration, procedural controls ange	Later than desired "Make" of OPPS Due to the low level of oxidizer autogenous pressure and high actuation pressure of OPPS, orifice diameter was changed from 0.500 to 0.460 in. on subsequent vehicles	Demonstrate the satisfactory performance of GLV to place spacecraft into the required orbit for 14 day mission All vehicle systems performed satisfactorily and GLV successfully placed the spacecraft into an 87-177 n mi orbit LAUNCH DATE: 4 Dec 65
;e inadvertently connected to oxidizer cavity port drain II engine welds between crossover manifold and thrust ıs ılems: fail-safe malfunction during cycling test; switch ıation encountered during attempts to erect Stage I ank dome clearance mp failures, faulty compensator action due to wrong orifice	RGS pitch down (18%) maneuver at L/O +330 sec Low frequency noise in GE Mod III radar data, attributed to tropospheric condition	Place spacecraft into orbit for rendezvous with Agena Rendezvous and docking accomplished LAUNCH DATE: 16 Mar 66

Figure II. G-1. Gemini Launch Vehicle Histories (Continued)

[redacted]

VEHICLE NUMBER	DENVER		
		HORIZONTAL	VERTICAL TEST
GLV-9	STG I Fuel Fwd dome original build scrapped, dome chord mismatch STG II Fuel scrapped, undersize, GLV-10 reallocated to GLV-9 Tank calibration procedures improved First tanks for rail shipment BALTIMORE DELIVERY: 16 Aug 65	STG II Ox feedline shimmed, Fuel tank oversize VERTICAL DELIVERY: 10 Dec 65	3 Eng xducers failed Eng thrust chamber valves replaced pre-CSAT Mod period to incorp 4 ECPs, 5 CADs 16 Cut Shields on twisted wire, repaired POWER-ON TIME: 169 hrs
GLV-10	STG I Ox, mismatch cleared as OK-as-is STG II Ox, weld crack after hydro, repaired STG I Fuel, oversize truss bolt holes, special bolts BALTIMORE DELIVERY: 21 Sep 65	STG II Fuel rejected, returned to Denver, battery acid damage during rail shipment, GLV-11 tank assigned to GLV-10 STG I Eng injector dome leak, reworked and welded STG I Splice bolts sheared, torque reduced Damaged shielded wire discovered, complete reinspection all GLVs STG II Eng pump start and run torques high-sealer compound VERTICAL DELIVERY: 9 Mar 66	Electric hyd pump jammed by Hysol STG II Eng combustion chamber weld crack ins Critical component data inspection revealed ro out-of-spec and press-switch out-of-calib Mod period to incorp 5 ECPs and replace TCV POWER-ON TIME: 140 hrs
GLV-11	GLV-11 STG II Fuel reassigned from GLV-12 STG I Ox weld crack after hydro, repaired BALTIMORE DELIVERY: 3 Nov 65 (STG II Fuel) 16 Jan 66 (C-124)	STG II Eng exit flange weld crack repaired VERTICAL DELIVERY: 29 Apr 66	Component data pkgs reviewed pre-power-on Eng pump discharge lines cleaned Eng thrust chamber valves replaced PCM Encoder excessive problems Mod period for 7 ECPs and 2 CADs POWER-ON TIME: 143 hrs
GLV-12	STG II Fuel from GLV 10 reworked to eliminate acid damaged portions (Aft cone and dome) STG II Fuel, 4 weld cracks after hydro, repaired STG II Ox, 3 weld cracks after hydro, repaired BALTIMORE DELIVERY: 20 Jan 66 (C-124) (STG II Fuel) 12 Mar 66 (C-124)	STG II Eng roll nozzle cracked at exit flange, replaced STG II Eng adapter tube welds cracked, repaired Hyd Test selector valves cycled to vendor for rework VERTICAL DELIVERY: 22 Jun 66	Eng pump discharge lines cleaned Component data review revealed Staging Relay replaced Hyd test selector valve caused flow starvation POWER-ON TIME: 107 hrs

BALTIMORE				
	CSAT/VAT	POST–VAT ACTIVITY	OPEN ITEMS AND WAIVERS ETR SHIPMENT	
	CSAT 9 Feb 66/VAT 15 Feb 66 STG II Eng start signal drop-outs, test tool Gyro noise at null to be studied Eng quality at acceptance reviewed Component aging and availability reviewed VAT WALKDOWN SQUAWKS: 11 Roll-control nozzle gasket leak	STG II Eng crossover tube rewelded Roll control nozzle gasket leak 4 Airborne ECPs installed 3 CADs accomplished	Shortages: TARS (shipped to Honeywell for rework) Waivers: Clevis hole dimensions on eng Defective components: None Unqualified components: None Recap tasks to ETR: 11 GPIS open: 27 ETR DELIVERY 10 Mar 66	Stage II hot Stage II Eng S/A 2 Engin S/A 2 Fuel p SMRD Circui of printed ci Hydraulic te Stage II A/P ABETS Puls Mistram xpor
pection ll actuator bolts (corroded)	CSAT 14 Apr 66/VAT 26 Apr 66 Vehicle in excellent shape, quality and configuration VAT WALKDOWN SQUAWKS: 15	STG II Eng turbo-torque rechecked Accomp add'l engineering for staging redundancy Electric hyd pump leak, replaced 0 Airborne ECPs installed 1 CAD accomplished	Shortages: None Waivers: None Defective components: None Unqualified components: None Recap tasks to ETR: 10 GPIS open: 2 ETR DELIVERY: 20 May 66	Ball-joint mi Engine comb CVE Erratic Crazing of ur A/P failures Actuator 2_1 r Actuator 3_1 r Hyd selector MISTRAM xp Launch nut c Destruct init
	CSAT 9 Jun 66/VAT 20 Jun 66 Static inverter vs TARS Timing reviewed Diode assembly log showed low voltage, replaced Hyd accumulator strap clamp nick problem reviewed VAT WALKDOWN SQUAWKS: 17	Eng pump discharge lines x-rayed for cocking Retest after diode assy replaced Fuel autogenous line replaced for dents 1 Airborne ECP installed 0 CADs accomplished	Shortages: None Waivers: Clevis hole dimensions on Eng Defective components: None Unqualified components: None Recap tasks to ETR: 12 GPIS open: 14 ETR DELIVERY: 12 Jul 66 (Super-Guppy)	PSV Drain h Metal particl Oil cooler ou Stage I ox sp IPS Voltage
not cycle-proofed,	CSAT 29 Jul 66/VAT 9 Aug 66 ASSETS Program quarantees balance between reliability and logistics of remaining airborne hardware, GLV–9 and up VAT WALKDOWN SQUAWKS: 15	0 ECPs installed 0 CADs accomplished	Shortages: None Waivers: None Defective components: None Unqualified components: None Recap tasks to ETR: 11 GPIS open: 2 ETR DELIVERY: 3 Sep 66	Wrench dropp Ball-joint mi Extruded tefl Hydraulic pu Spurious SEC decoder Decoder inve Autopilot hac Replacement

ETR		
SIGNIFICANT EVENTS	FLIGHT ANOMALIES AND RESOLUTIONS	FLIGHT OBJECTIVES AND RESULTS
as cooler galled nut problem ie combustion chamber seal leaks crimped raco seal evalve o-ring imperfection inoperative due to improper installation :uit : selector valve anomalies pitch gyro failure modulator problem er failures	Excessive Stage I oxidizer outage Increase mixture ratio dispersions commensurate with GLV and Titan II system histories	Same mission as GLV–8 Atlas failed to achieve orbit. launch was rescheduled for rendezvous with ATDA; all GLV systems performed as expected; trajectory was nominal, enabling rendezvous to be successfully accomplished LAUNCH DATE 3 Jun 66
alignment of engine pump discharge lines stion chamber weld leak iverspeed circuit relay bilical 3D1E and 3D2E face plates ise problems II land alve slow actuation ider module failure ble potting defects tor rocker arm problem	Stage II Fuel topping disconnect break at liftoff Lanyard re-rigged on subsequent flights "Venting" of Stage I ox tank shortly after staging Probable causes: dome penetration by transportation section debris, cracking of ablative coating. or burn through of the tank side wall; no corrective action required	Same mission as GLV–8 All GLV systems performed as planned. GLV ascent trajectory was exactly as required to place spacecraft into prescribed orbit; "Venting" of Stage I ox tank shortly after BECO did not adversely affect the success of flight LAUNCH DATE: 18 Jul 66
;e leak contamination in propellant feed system et lines, teflon liner imperfections t weld leak, waterglass epoxy sealed ansient, auto pilot replaced	Excessive Stage I ox outage and eng mixture ratio shift The mixture ratio for GLV–12 was biased –1 2% to protect against repeating Pinhole leak in Stage I ox tank caused by corrosive action of ox Application of sodium silicate forced into pinhole sealed the leak	Same mission as GLV–8 Although an ox leak in GLV tank caused one day slip in launch schedule. GLV did successfully fulfill all its requirements; all systems performed as expected; trajectory again was as required for rendezvous LAUNCH DATE: 12 Sep 66
I in Stage I fuel. inspected ilignment engine pump discharge line. line replaced ı in fuel bootstrap line) intergranular corrosion problem signals during J-CSAT attributed to guidance igation for solder balls in 800–cycle inverter failure ıtopilot removed due to apparent drop in sync	204 lb overload of Stage I ox Lack of confidence in tank bottom temp probe proved to be unjustified Stage II ox pump inlet temp 10 deg high Caused by mechanical or electrical deformation during ox loading, one time malfunction Stage I ox and fuel tanks may have opened (see GLV–10)	Same mission as GLV–8 All GLV systems performed as required. no significant anomalies occurred and trajectory was exactly as planned. all objectives achieved LAUNCH DATE: 11 Nov 66

Figure II. G-1. Gemini Launch Vehicle Histories (Continued)

[redacted]

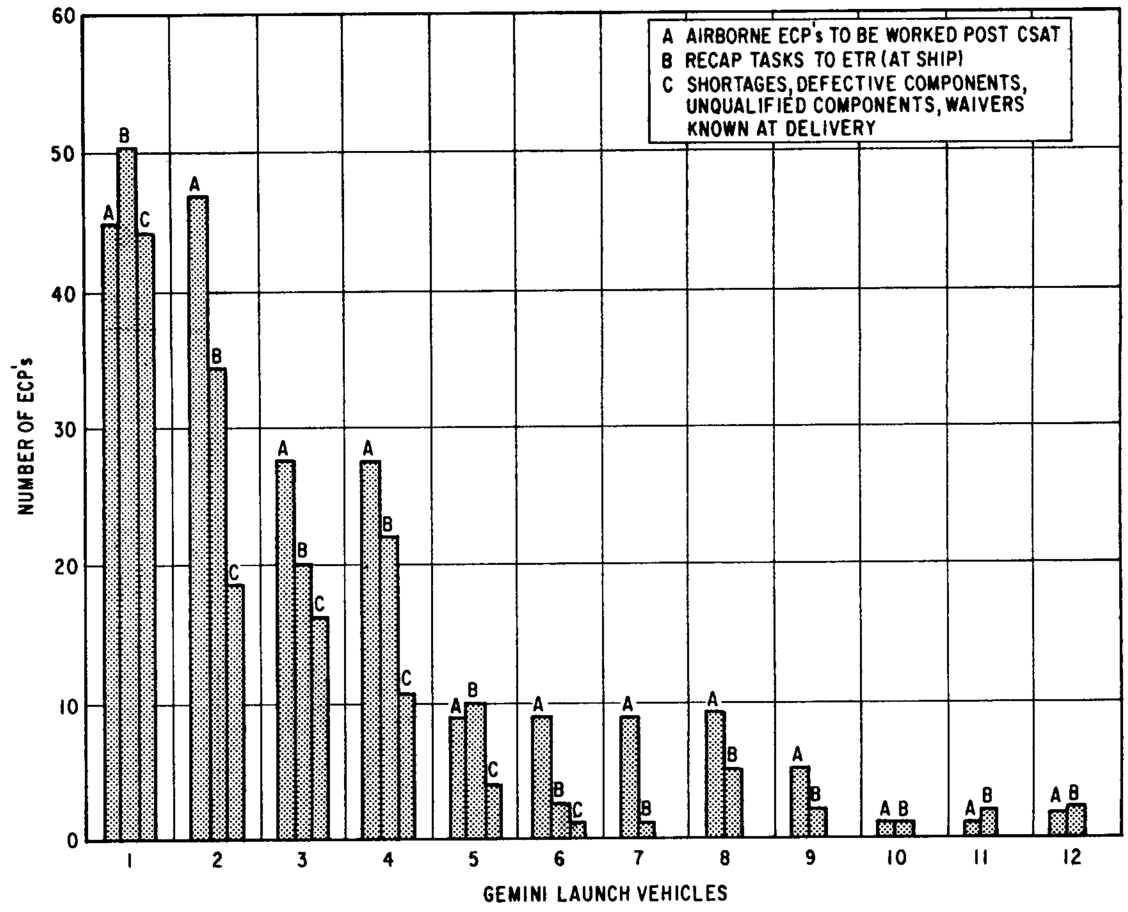

Figure II-G-2. Performance Indicators Flight Configuration at Factory

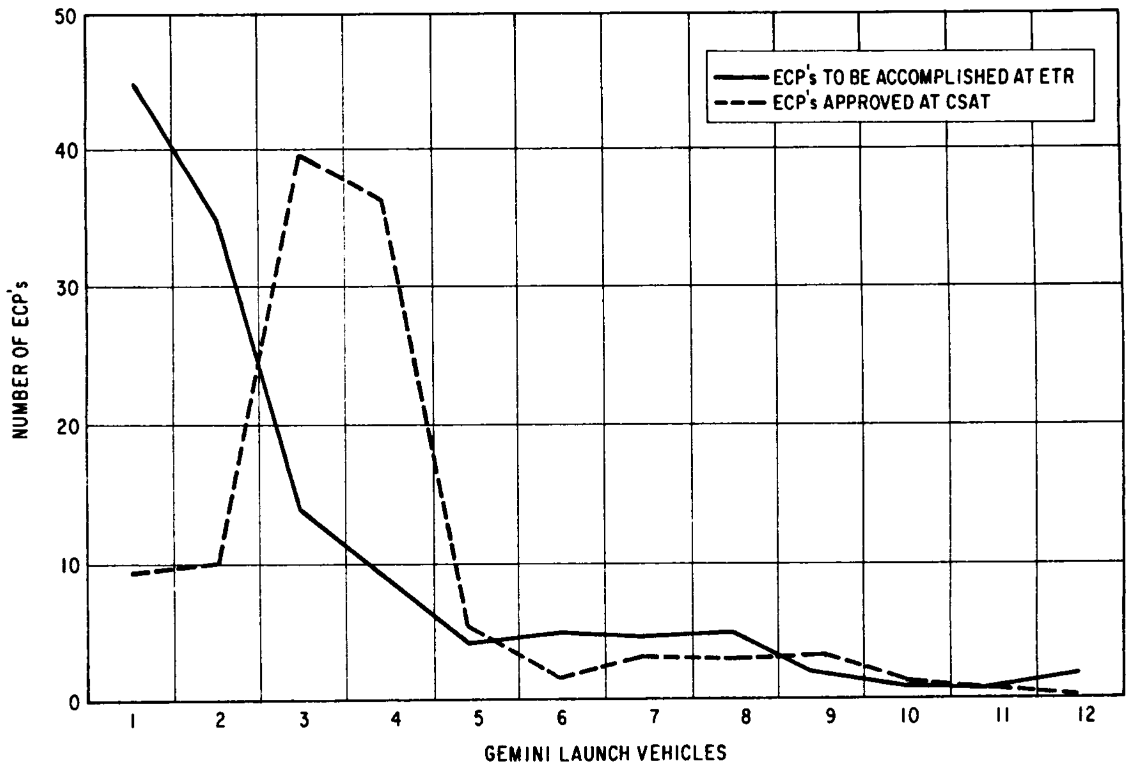

Figure II. G-3. Performance Indicators Flight Configuration at Factory (Continued)

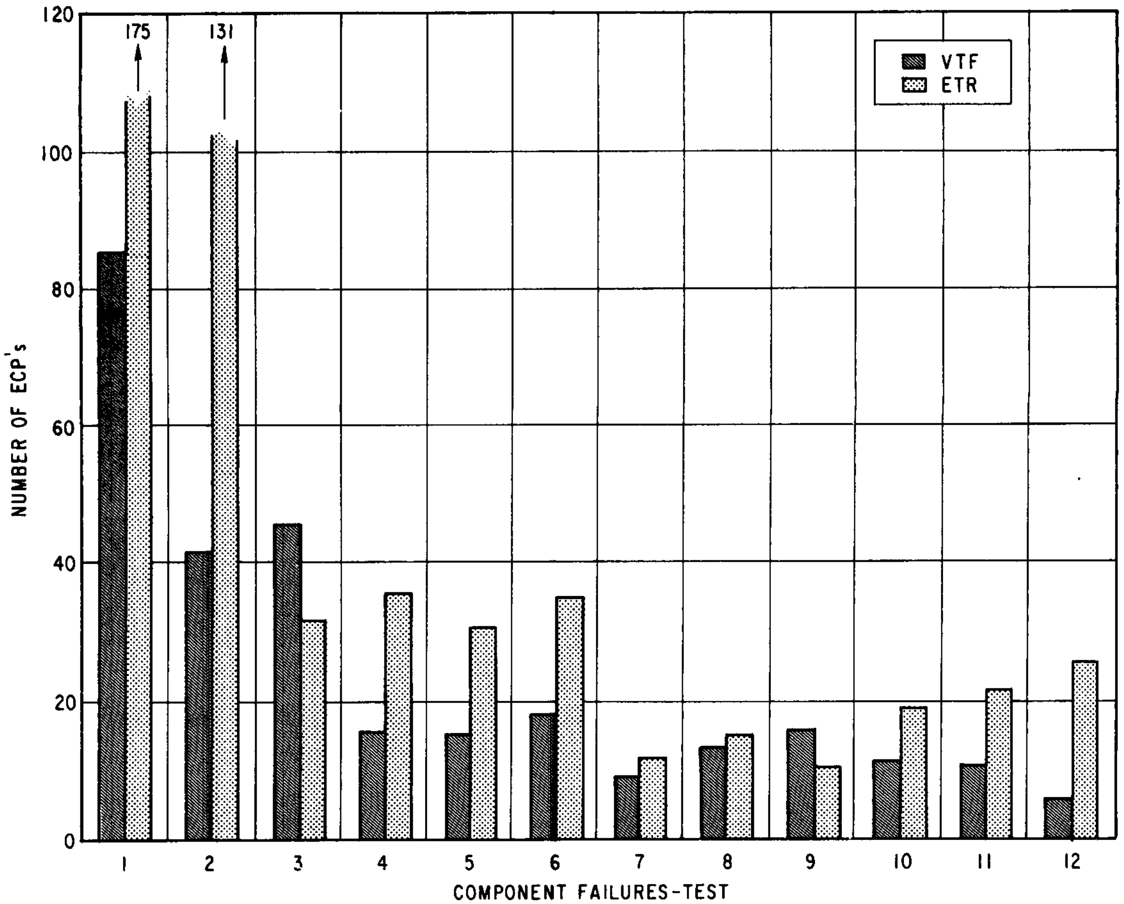

Figure II. G-4. Component Failures - Test (Total)

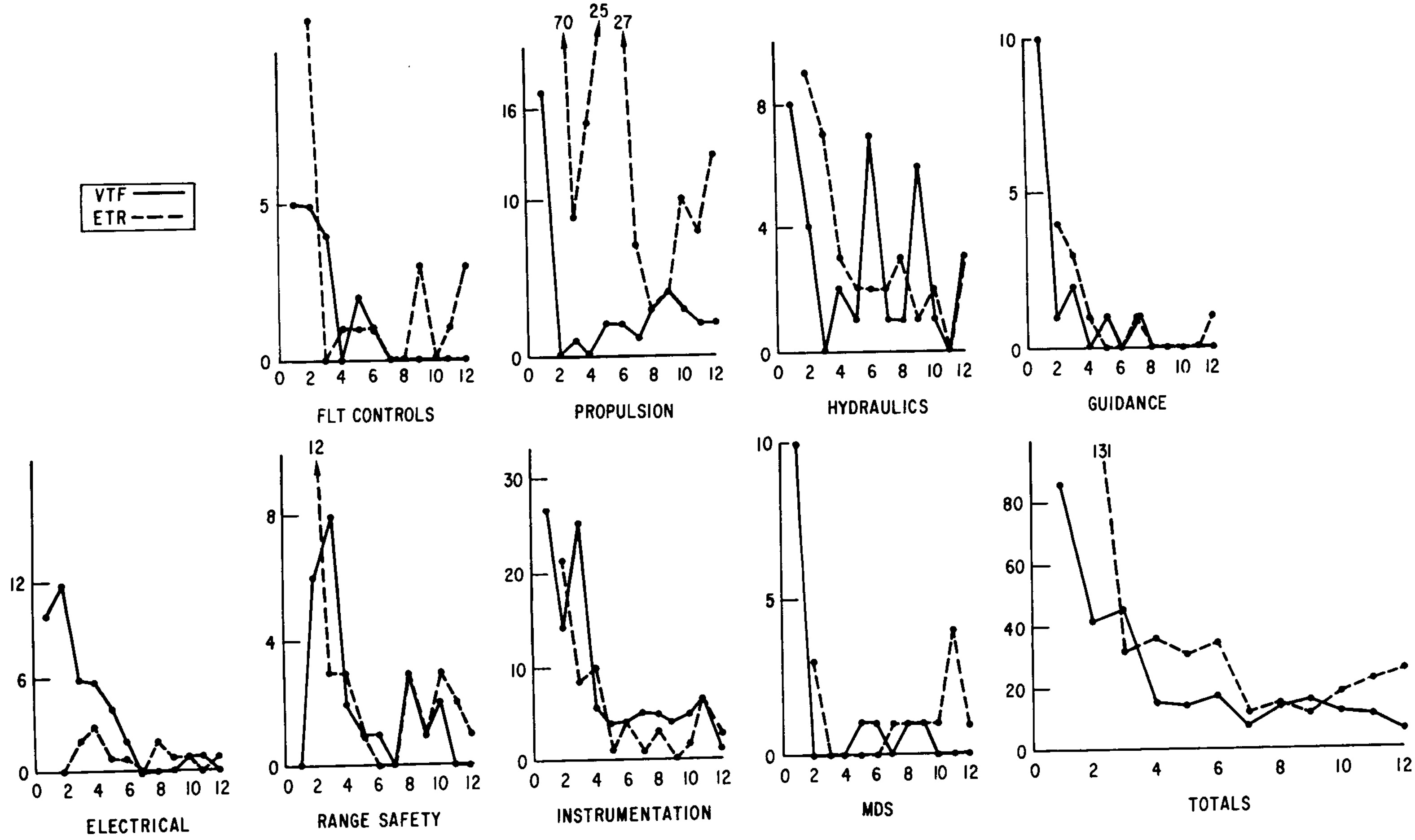

Figure II. G-5. Significant Component Failures - VTF/ETR Testing (By Subsystem)

SECTION II-G

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1. Fabrication and Test History for Gemini Launch Vehicle LV-344-1 through 12, Martin Company
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3. Vehicle Acceptance Report GLV-1 through 12, Aerospace Corporation

SECTION II. G

DEFINITIONS

ABETS	Airborne Beacon Test Set
AGE	Aerospace Ground Equipment
APS	Accessory Power Supply
ASSETS	Assets program to assure adequate and reliable spares through program end.
B/P	Blueprint
CAD	Corrective Action Directive
CSAT	Combined Systems Acceptance Test
Ditmco	Programmed electrical continuity and high voltage tester.
ECP	Engineering Change Proposal
EEI	Electronic and Electromagnetic Interference
Eng	Engine
ETR	Eastern Test Range
FM	Frequency Modulation
GLV	Gemini Launch Vehicle
GPIS	Gemini Problem Investigation Summary
IPS	Instrumentation Power Supply
MARS	Martin Automatic Reporting System (discrepancy report)
MDS	Malfunction Detection System
MISTRAM	Missile Tracking and Monitoring
Mod	Modification
PCM	Pulse Code Modulation
PSV	Propellant Sequencing Valve
RCVR	Receiver
RFI	Radio Frequency Interference
RGS	Radio Guidance System
RSP	Rate Switch Package
SMRD	Spin Motor Rotation Detection (gyros)
Stg I Ox	Stage I Oxidizer Tank
TARS	Three Axis Reference System
TRWT	Trouble Report Withholding Tag (discrepancy report)
VAT	Vehicle Acceptance Team

III. GEMINI AGENA TARGET VEHICLE

A. INTRODUCTION

1. NASA/SSD/AEROSPACE ROLE

SSD responsibilities for the DOD effort required on the NASA Gemini program were established by the NASA-DOD Operational and Management Plan for the Gemini Program (29 December 1961). In the plan, references to the Target Vehicle portion of the over-all Gemini program were quite general in relation to both management and technical objectives.

In March 1962, AFSSD was directed by Hq AFSC to initiate the Agena Target effort. Also in March 1962, initial funding was received on a NASA-DOD Purchase Request and working contact was established between the SSD Program Office and NASA Marshall Space Flight Center, program manager for NASA Manned Spacecraft Center for the Target Vehicle Program.

During January 1963, there was a realignment of responsibilities within NASA. It resulted in the elimination of Marshall Space Flight Center as a middle management office and the assumption by the Manned Spacecraft Center, Gemini Program Office (GPO), of direct responsibility for the Target program as a part of the over-all Gemini program management.

Because of the lack of early detailed program and technical objectives, Gemini Target management relationships and fundamental responsibilities evolved from work and coordination meetings between NASA GPO and SSD Program Office personnel. As might be expected, final management responsibilities were characterized by a blending of the philosophies of the NASA Manned Spacecraft Center and the Air Force Space Systems Division.

Following months of negotiations and coordination, the management responsibilities were finalized, in March 1965, in a formal document, NASA MSC and AFSSD Management Responsibilities Agreement for the Gemini Atlas Agena Target Vehicle Systems Program. This agreement clarified and supplemented the earlier NASA-DOD Operational and Management Plan and resulted in a system of cooperative program direction and problem reporting.

In the fall of 1964, Aerospace Corporation was put under contract by SSD for Technical Surveillance of the Gemini Agena Target Vehicle. The Aerospace responsibilities included close monitoring of the vehicles from the beginning of subsystem fabrication, through final vehicle systems test, and prelaunch and launch operations. Aerospace personnel also became deeply involved in special task-force efforts and in all coordination meetings with NASA/SSD/contractors. Some aspects (where applicable) of the Pilot Safety Program, developed by Aerospace for the GLV, were carried over to the GATV program.

2. PROGRAM MANAGEMENT

a. Space Systems Division (SSD)

Responsibility for what was to become the Gemini Atlas Agena Target Vehicle System was originally established within the Program Integration Division of the SLV-3 Directorate under the Deputy for Engineering. This office was assigned responsibility for Air Force support of all NASA Agena Programs, which included at that time the Ranger, Mariner, Nimbus, EGO, POGO, FIRE, OAO, ECHO, and Rebound programs as well as the Canadian S-27. When Gemini Target was added in March 1962, one project engineer (a captain) was assigned responsibility for the program.

During the first quarter of 1963, the USAF and NASA Headquarters reviewed their basic support agreement covering the unmanned NASA programs. This review resulted in the decision to transfer the Agena-peculiar modification and system integration contracts from SSD to the NASA Lewis Research Center, with the exception of the Gemini Target program, which would remain at SSD under the over-all program management of NASA/MSC.

As a result of the transfer of responsibilities, additional manpower was made available to the new Gemini Agena Division under the Agena Directorate (Figure III. A-1). The method of operation was similar to that used by all Air Force programs within SSD. In particular, the following provisions were made:

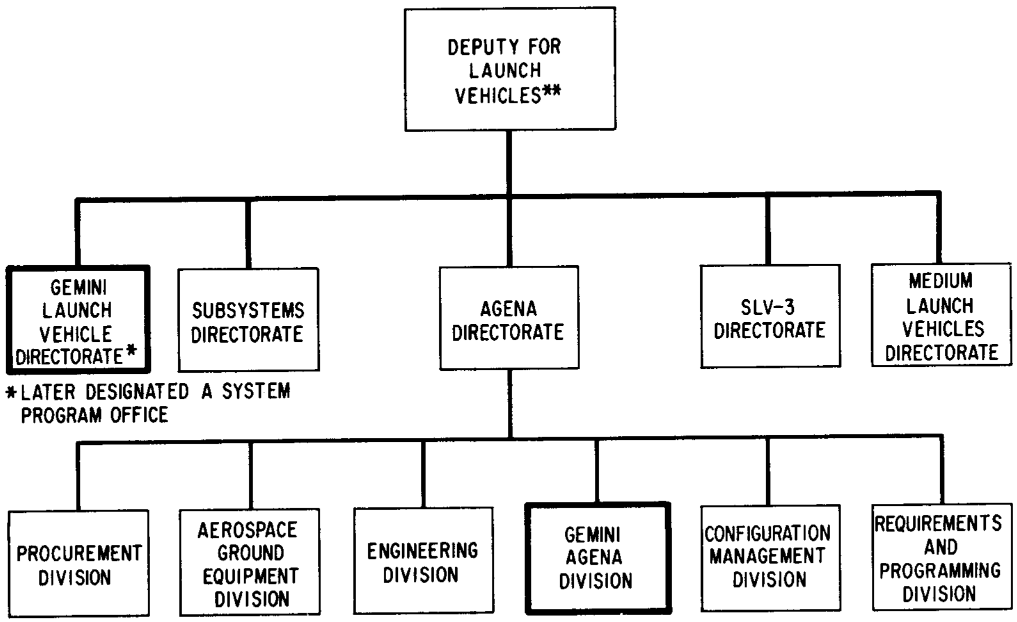
1) The Atlas (SLV-3) launch vehicles and associated launch services would be procured and managed by the SLV-3 Directorate upon receipt of requirements and funds from the Gemini Agena Division, as well as the conversion of Launch Complex 14 to an SLV-3 configuration. (By separate Headquarters agreement, USAF agreed to fund the launch complex conversion from Mercury Atlas configuration to a standard SLV-3/Agena D configuration.)

2) Guidance and ground computer support would be provided by the Subsystems Directorate upon receipt of requirements and funds from the Gemini Agena Division.

3) The basic Agena D vehicles and launch services for the final Gemini Agena Target Vehicles would be procured and managed by the Agena Directorate upon receipt of requirements and funds from the Gemini Agena Division. In addition, the Agena Directorate would provide technical consultant and specific subsystem engineering support to the Target Vehicle program, as for all Air Force space programs using the Agena D vehicle.

It became apparent, however, that the Gemini Agena Division was not only understaffed, but lacked experienced, ranking personnel. Recognition of this problem led Hq AFSC to authorize a significant reorganization and realignment of personnel. The new organizational structure (Figure III. A-2) was basically in accordance with AFSCM 375-3 with functions as follows:

- <u>Director and Deputy Director</u>. Managed and directed the development, procurement, and test activities for the Gemini Atlas Agena Target Vehicle System. Kept NASA Manned Spacecraft Center, Gemini Program Office fully informed of activities concerning the Gemini Atlas Agena Target Vehicle.
- <u>Program Control and Configuration Management</u>. Because the new Directorate was organized with minimum manpower, the Program Control and Configuration Management were not made separate divisions, but functioned instead as identifiable sub-elements of the Program Director's office.
  - <u>Program Control</u>. Managed the program control functions of preparation and maintenance of program documentation; acquisition and control of program resources; and management review, evaluation, and reporting of program progress.
  - <u>Configuration Management</u>. Managed the Gemini Agena Target Vehicle (GATV) configuration identification, control, and accounting system. Supervised contractor efforts in formulation, execution, and discipline of configuration control systems. Maintained records of approved and actual configuration. Managed the Reliability and Quality Programs for the GATV.



** TITLE CHANGED FROM DEPUTY FOR ENGINEERING IN JULY 1964

Figure III. A-1. Organization Chart - Agena Directorate, SSD

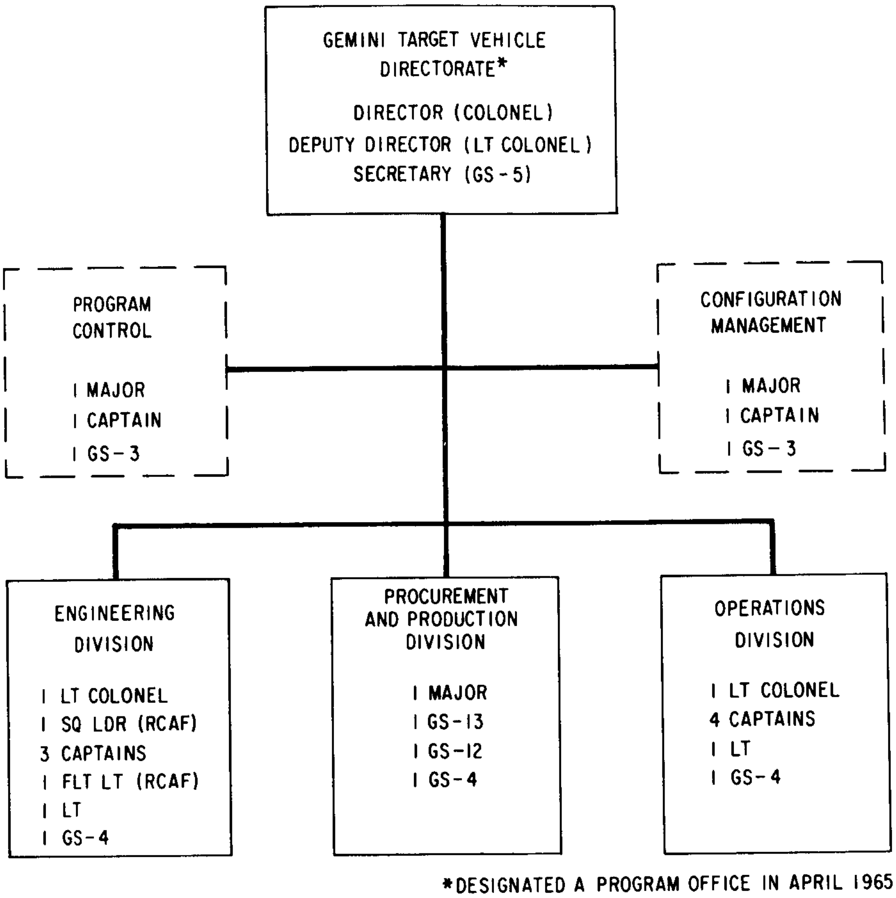


Figure III. A-2. Organization Chart - Gemini Agena Target Vehicle, SSD

- Engineering Division. Was responsible for over-all design, development, fabrication, assembly, and in-plant testing through factory acceptance of the GATV (less the NASA-procured Target Docking Adapter), and associated AGE; and for system integration of the complete Gemini Atlas Agena Vehicle System. Coordinated engineering effort with GPO.
- Procurement and Production Division. Was responsible for managing and coordinating all procurement actions for contracts under Gemini Agena Target Vehicle jurisdiction. Served as the focal point in all procurement matters and negotiated contracts and contract changes.
- Operations Divison. Was responsible for managing and coordinating all functions from Gemini Agena Target Vehicle and SLV-3 factory acceptance through GAATV launch operations; and for over-all supervision of operations analysis and support activities, guidance equations, mission planning coordination, and all associated documentation. Coordinated operations with GPO.

The value of the new organization became evident following the flight failure of the Target Vehicle during its first launch on 25 October 1965 - less than four months after the formation of the new Directorate. The failure investigation and top priority recovery test program were accomplished in what was considered an impossible time period. The success of the recovery program, ahead of schedule, was attributed to the strong program management, and to the knowledgeable experienced personnel assigned to the program.

b. Aerospace Corporation

When Aerospace was given the technical surveillance responsibility over the Agena Target Vehicle in the fall of 1964, it appeared logical that this function should be incorporated in the existing Gemini Launch Systems Directorate, which performed the systems engineering and had technical direction of the Gemini Launch Vehicle. This group of systems engineers and operationally-oriented personnel, who had gained a wealth of experience in manned space programs throughout the Mercury and Gemini programs, were best suited to take on this additional responsibility.

In order to cover both the GLV and GATV with a minimum number of personnel, the basic structure of the organization was maintained (Figure III. A-3). The Directors of Airborne Systems and Systems Test Operations assumed the responsibilities for both vehicles. The same functional systems areas of both vehicles were placed under the supervision of the single manager, with subsystems engineers assigned to the particular systems on each vehicle reporting to him.

In addition to best utilizing the available manpower, this organizational structure allowed for a ready cross-feeding of information between the two vehicle programs. In particular, any failures or problems occurring in the equipment of one program would automatically be fed into the other program. The systems manager, cognizant of both vehicles, could then take immediate action to determine the impact and necessary corrective action on both programs.

c. Lockheed Missiles and Space Company (LMSC)

(1) LMSC Management Structure

The LMSC management philosophy for satellite and space programs has consistently been to establish a sound technological base in stable line organizations and to assign a program manager to serve as the "adaption kit" or interface between these organizations and a specific customer with specific mission or contract requirements. Vital to the success of this approach is the additional Lockheed philosophy that the Program Manager must appropriately utilize the required skills existing in all of the Lockheed Aircraft Corporation or its large family of subcontractors and vendors.

In 1962, the Satellite Systems Division in LMSC was the product division designated to handle satellite programs, primarily for the Air Force. Early in the following year a new Space Programs

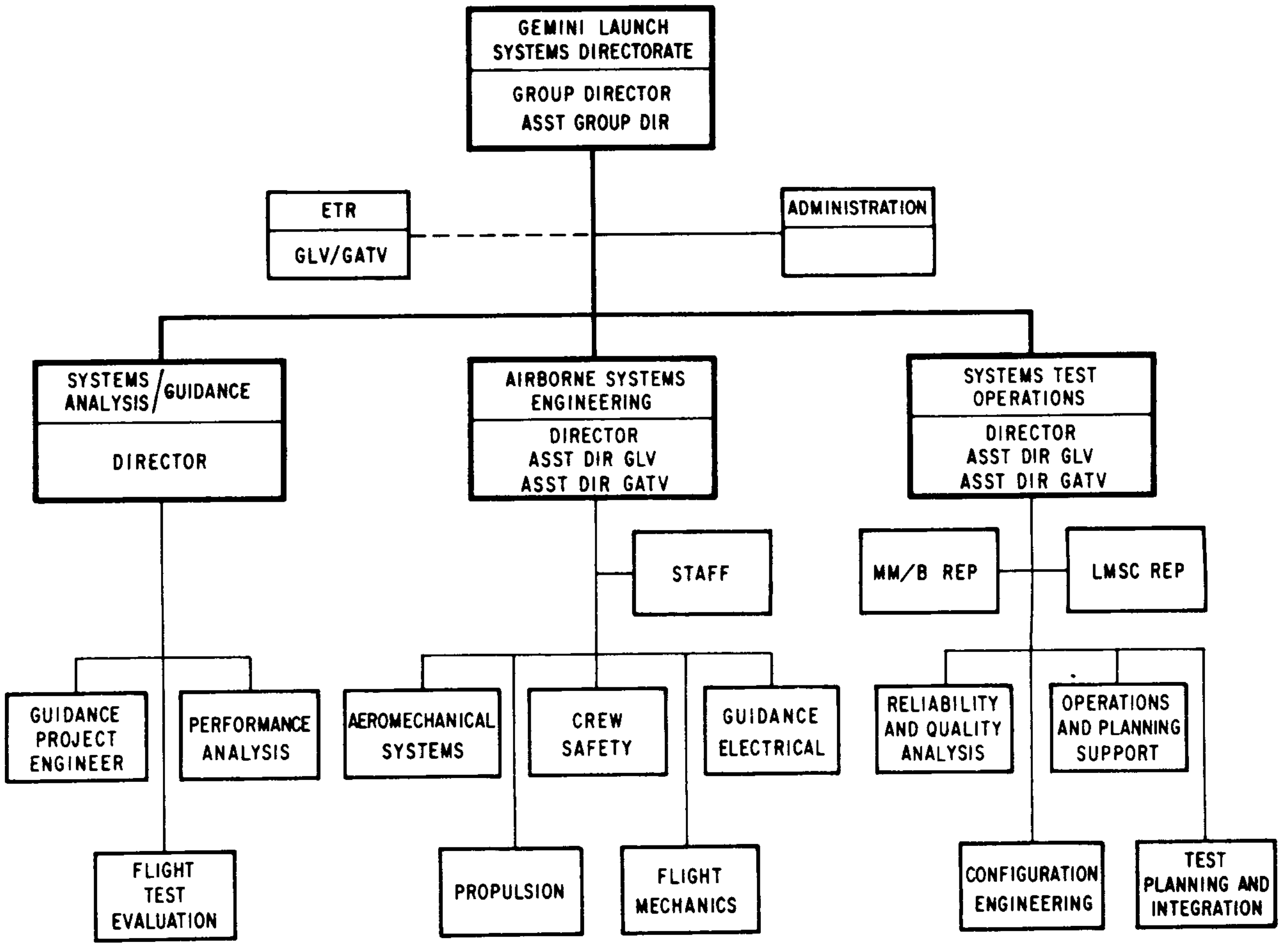


Figure III. A-3. Organization Chart, Aerospace Corporation

Division was created to concentrate on the NASA-mission-oriented programs, including the newly awarded RIFT Project (Reactor In-flight Testing) out of the Marshall Space Flight Center. The original Satellite Systems Division was renamed Space Systems Division, the name it still retains.

In mid-1964, with the partial termination of the RIFT contract, the remaining NASA mission-oriented programs that involved the Agena were returned to the Space Systems Division, and the Space Program Division was dissolved.. Each separate division in LMSC is headed by a Corporate Vice-President and each division adapts its policies and procedures to the needs of its particular customer. It was evident, therefore, that the significant changes in reporting levels and operating modes that occurred during these organizational shifts influenced the structure and performance capability of the program offices oriented to the fulfillment of NASA missions. Working relationships were further influenced by similar shifts in organizational alignments and assignments of responsibilities in Air Force and NASA agencies during this same period.

(2) GATV Program Management Evolution

There were three relatively distinct phases of management structure and associated control techniques throughout the history of the GATV Program. They were the initial intermediate and final phases, descriptions of which follow.

(a) Initial Phase. At LMSC, the Medium Space Vehicles Programs (MSVP) organization had been established in the Satellite Systems Division, and was headed by an Assistant General Manager (AGM) (Figure III. A-4). It was responsible for the adapting the Agena to missions for NASA. Within MSVP, the Gemini Program Office had been set up under a manager, with two subordinate supervisors. One supervisor was responsible for vehicle development, and the other for ground support equipment and facilities. The Manager reported through an intermediate level of management (the Programs Manager) to the MSVP AGM. All correspondence with the customer was signed by the AGM so that he, in effect, became the primary technical and management interface with customer agencies.

At this time, discussions were taking place between the Air Force and NASA as to the desirability of transferring all NASA missions to direct NASA contracts, without the intervening Air Force contract management activity. This was subsequently done on all MSVP programs (transfer to the NASA Lewis Research Center) with the exception of the GATV Program, which was retained on Air Force contract.

Within MSVP, a Gemini Engineering Department was established to handle GATV structural design and modifications, electrical circuitry and modifications, and instrumentation and guidance system modifications. Since the GATV was a peculiarization of the Agena vehicle, this department had an interface and integration group that worked closely with Agena Engineering at LMSC and with McDonnell Aircraft Corporation, which was responsible for the Target Docking Adapter. Propulsion systems (both primary and secondary) and command and communications (C&C) were excluded from the activities of the Gemini Engineering Department. Propulsion systems engineering, development, and test were done by the Bell Aerosystems Corp. under subcontract to LMSC; their technical interface was through the central SSD Propulsion Department with whom they were accustomed to work on the Agena Propulsion Systems. Since the C&C Subsystem was totally peculiar to Gemini and represented a state of the art advance beyond the existing one, engineering and development were assigned to the LMSC Research and Development Division, with budgetary control by the LMSC Gemini Program Manager.

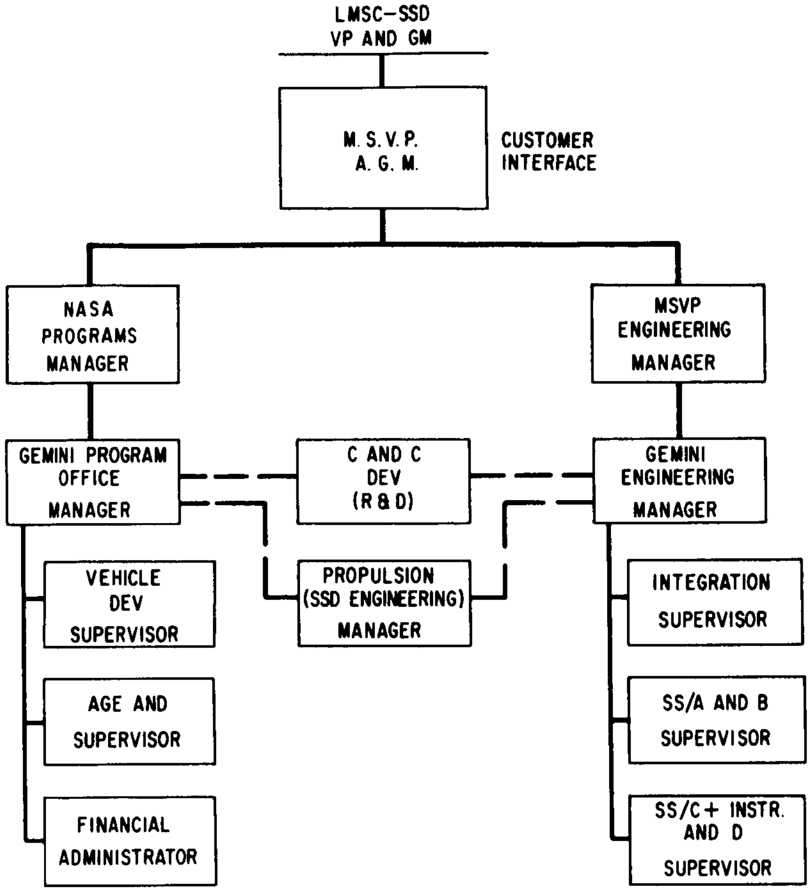


Figure III.A-4. Organization Chart Lockheed Medium Space Vehicles

The basic function of the Gemini Manager in this structure was to interpret customer requirements in terms to which supporting internal or subcontractor organizations could respond. The specific manner of meeting the technical requirements so established was the responsibility of the individual technical organizations, subject to the cost, schedule, and contract constraints placed by the Gemini Manager. This arrangement made the Program Office an interpretation and constraints agency, with the majority of detailed technical planning done and decisions made by the cognizant technical organization, which also controlled the budget assigned for its technical tasks.

(b) Intermediate Phase. The next phase in the evolution of GATV Program Management began toward the end of 1962 and was fully implemented early in 1963. The Manager of the Gemini Program was raised by one management level, now reporting directly to the Director in charge of Medium Space Vehicle Programs (Figure III. A-5). The Gemini Engineering Department was dissolved as an entity, and MSVP Engineering was restructured with departments aligned to functional subsystems. In view of the previous change, the position of Gemini Project Engineer was created in the MSVP engineering organization, on the Staff of the Engineering Manager, to serve as the direct interface between all vehicle engineering activities and the Program Office. Because of the state of its development and the realignment of MSVP Engineering, C&C subsystem responsibility was transferred from the Research and Development Division to MSVP Engineering

Several distinct objectives had led to the changes delineated. The reporting level increase for the Gemini Program Manager was intended to, and did, result in improved customer interface communications, and a formal acknowledgement by LMSC of the increasing importance of and management attention to, the program. This further enabled the Program Manager to establish working relationships and technical and budgetary controls with supporting organizations and subcontractors. The functional realignment of the MSVP Engineering organization established a more efficient utilization of available technical skills. The creation of the Vehicle Project Engineer in that organization further simplified internal communciations and provided, in effect although not in title, a strong technical assistant to the Program Manager.

During this period, the Air Force Space Systems Division correspondingly raised the rank level of its Program Manager. Meanwhile during 1963 when the Manned Spacecraft Center assumed the role of integrating agency for all elements of the Gemini Program, LMSC was relieved of its contract responsibilities for Atlas/Agena integration.

LMSC retained and successfully operated within this basic structure through 1963 and well into 1964. This was also the period of major changes in the concept, magnitude, timing, and funding levels of the program.

(c) Final Phase. First indications of the need for a transition to a new organizational phase were evident early in 1964 with the partial termination of the RIFT contract. Restructuring took place in the RIFT portion of the Space Programs Division without corresponding changes in the Medium Space Vehicles Programs organization. By mid-1964, however, it became apparent that the Space Programs Division could not be retained as a separate product division, so it was dissolved. The Medium Space

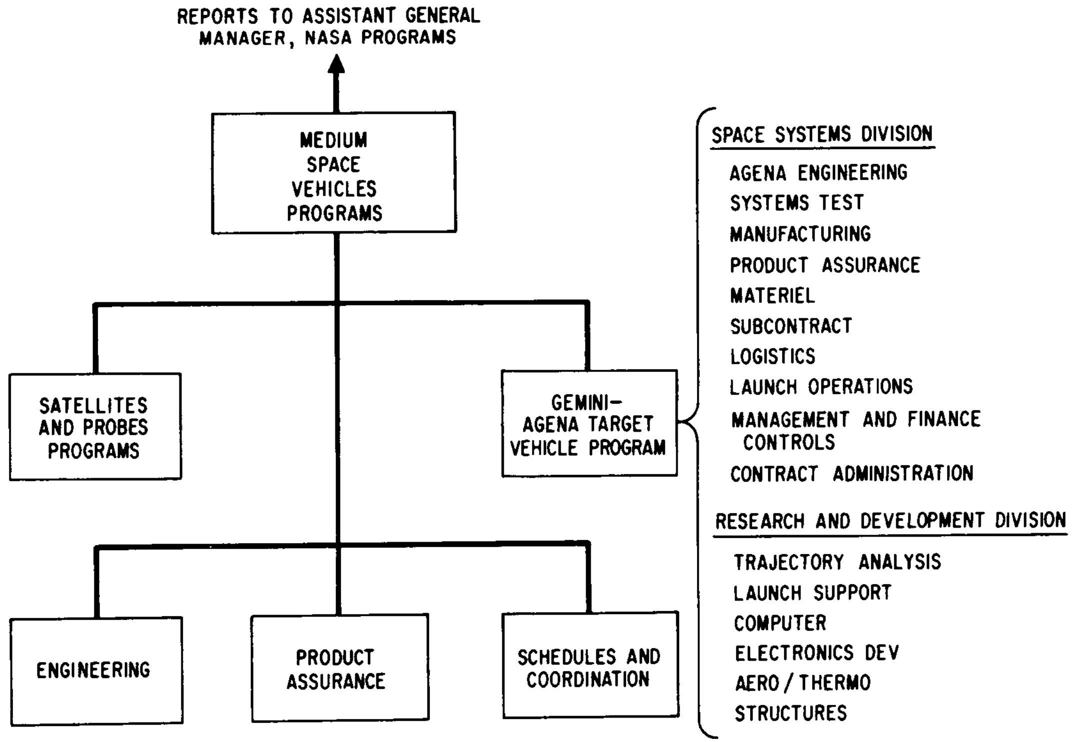


Figure III. A-5. Organization Chart - Intermediate Phase, Lockheed

Vehicles portion of the organization was renamed NASA Programs and moved, without structural change, back into the Space Systems Division, retaining the same management level of reporting it had had in Space Programs (Figures III. A-6 and 7).

During 1964, however, Air Force programs using the Agena had been testing various organizational realignments and program office organizations. The intent was to clarify and strengthen the role of the program office and to further improve the lines of communication between LMSC and its customer agencies.

In January 1965, a new Program Manager was assigned. The man selected had an extensive background in Vehicle Systems Test, Product Assurance, and Test Base operations, all areas critical in the successful completion of the balance of the program. As part of this change, and utilizing the experience gained in other Air Force programs, the entire Gemini Program office was restructured. Three significant posts were established, namely, the Chief Systems Engineer, the Assistant Program Manager for Program Controls, and the Configuration Management Office.

The purpose of these changes was: to permit the program office to augment its technical capability for mission success; to channel technical decision-making through the program office to its customer counter-parts; to increase the level of management concerned with the business management aspects of the program; and to formalize the implementation of configuration identification and control techniques. These changes were fully established and operating by the middle of 1965, and were retained with no significant change until the end of the program.

At about the same time, AFSSD established the GATV Directorate and separated it from the Standard Agena Directorate. Part of the LMSC implementation process included: augmenting the Program Office staff (it was approximately doubled); incorporating the technical decision-making and system integration functions into the Program Office; changing the mix of technical and management skills within the Program Office; and implementing new techniques in all support organizations for the definition, scheduling, and cost control of all program activities. In addition, since the Vehicle Project Engineering concept had proved itself with the NASA Engineering organization, similar positions were created in all other Gemini support organizations within LMSC, including: Systems Test, Product Assurance, Manufacturing, AGE Engineering, Financial Controls, Contract Administration, and Logistics.

One other characteristic apparent in this final organization phase was the task force approach. The method by which the Program Office directed and drew upon support organizations had demonstrated its ability to handle normal activities and their normal problems. Unusual situations, however, created the need for a different approach that could expedite normal activities, while fully following required policies and procedures. The GATV Program task forces were used in these circumstances.

The first major example is the C&C Task Force created in early 1965 to handle the redesign and qualification of the C&C Subsystem on an extremely compressed schedule. In this approach, carefully picked individuals were assigned full time to the Task Force, the members were grouped in close physical proximity, and a task force leader was designated to report directly to the LMSC Gemini Program Manager. Subsequent examples of this technique, which further demonstrated the tasks that could thus be successfully accomplished, were the Surefire Task Force and the later and smaller one that conducted the total refurbishment of Vehicle 5001. The additional advantage of the Task Force approach lay in the fact that management could, and on occasion did, have the task force leader and the program manager report (for the purposes of the Task Force) directly to a vice president and general manager, or to the company

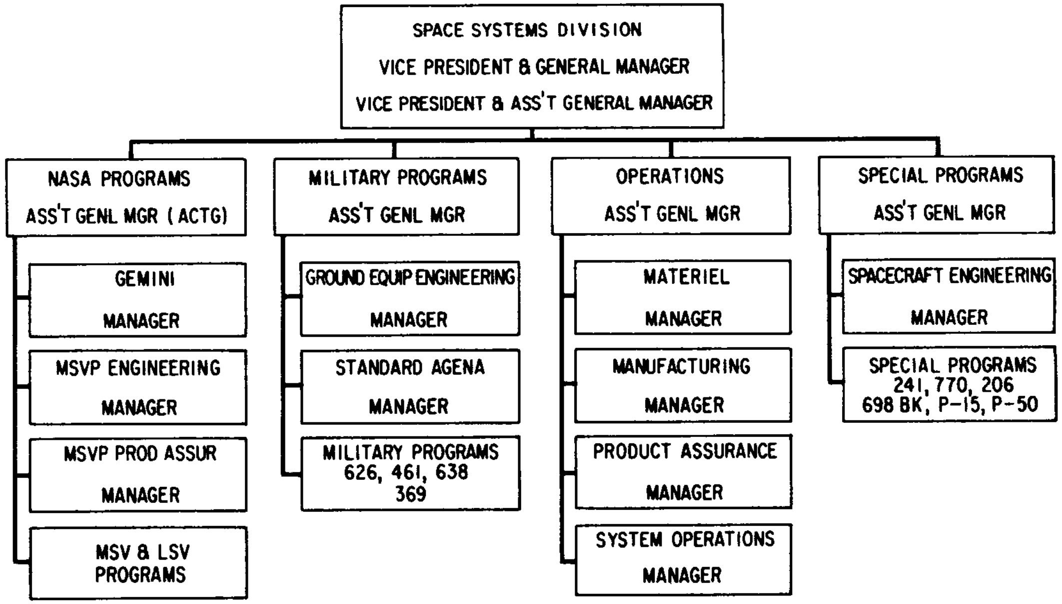


Figure III. A-6. Organization Chart - Gemini Support, Lockheed

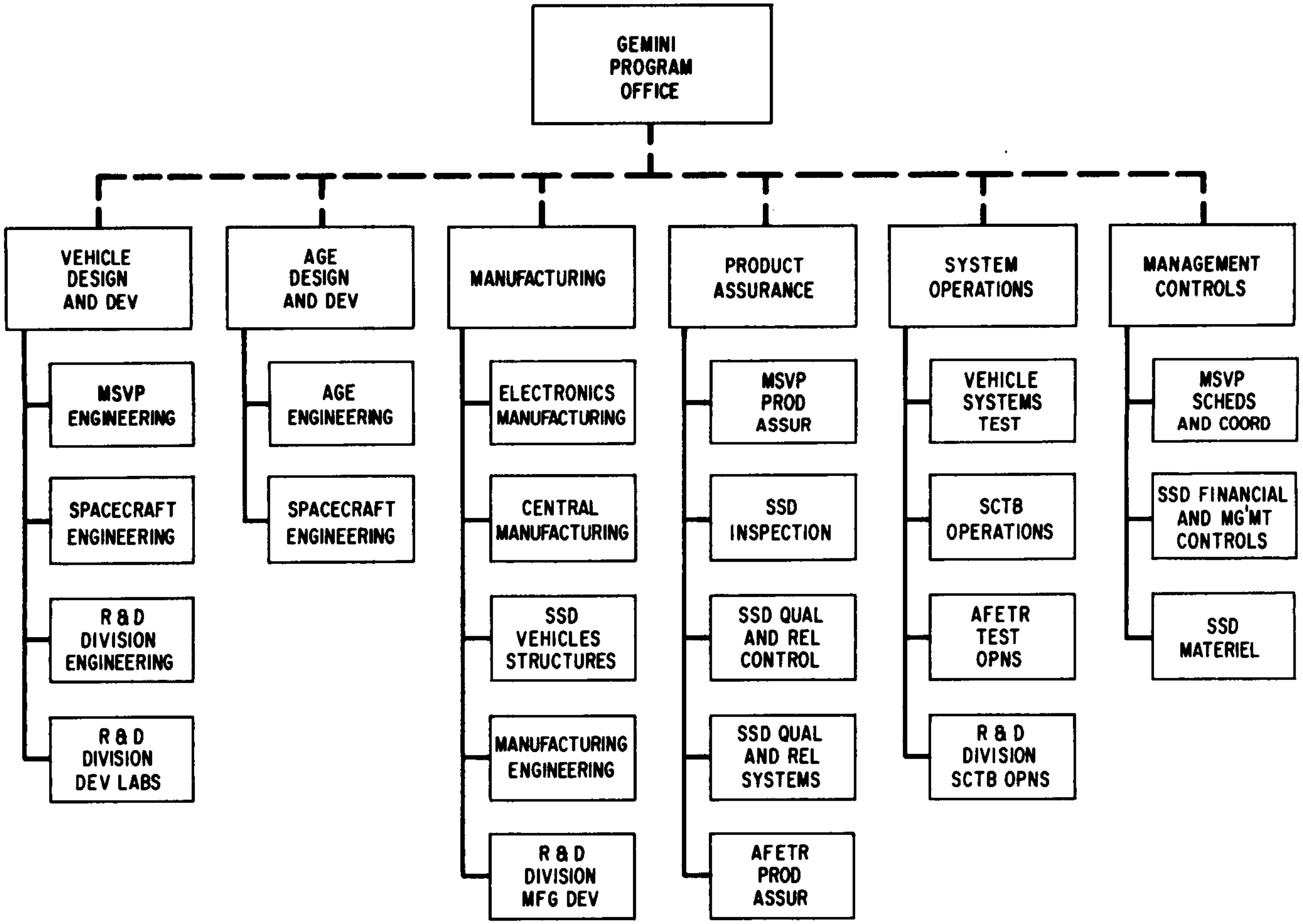


Figure III.A-7. Organization Chart, Control of Gemini Support Organization, Lockheed president himself. This made it possible (as in the case of Surefire) to involve whatever corporate resources were required to insure the successful completion of the activity. Its relation to the Program Office, however, still allowed normal program activities to be handled without interference.

If a similar program were to be undertaken today, with today's knowledge of the aerospace environment, it is believed that a Program Office structure, mode of operation, and relationship to supporting organizations, as implemented in 1965 on the GATV Program and successfully maintained through the end of the program, would be a logical manner in which to start.

3. SSD/AEROSPACE/CONTRACTOR/NASA INTERFACE

a. Atlas/Agena Coordination Committee

As in the case of the GLV, the large, diverse, and far-flung group of organizations participating in the Gemini program created two major management problems: adequate and timely communication, and proper control and coordination of the activities of the separate participants.

Because of these problems, the Atlas/Agena Coordination Committee was formed. It was headed by a chairman from the NASA Gemini Program Office and was composed of representatives of all government and industrial organizations that participated directly in the program (NASA, SSD, Aerospace, LMSC, GD/C, Rocketdyne, MAC.)

The Coordination Committee provided an instrument whereby the major program participants could review both their individual and mutual program and system problems and assess their program impact. The key results of the meetings were translated into action items, which were distributed to all participating agencies. The coordination meetings were followed by a government sessions devoted to discussions of action items and financial matters.

b. Gemini/Agena Interface Control Panel

This NASA chaired panel was established to effectively control the interface between the GATV and the spacecraft in a manner similar to that of the GLV/Spacecraft Interface Control Panel. Its charter set forth the following responsibilities and limitations. The panel:

- Consisted of representatives from NASA, LMSC, MAC, and SSD with Aerospace supporting SSD in an advisory capacity.
- Issued joint drawings and reports, which were subsequently approved by SSD/Aerospace and and NASA GPO.
- Did not make policy decisions. Policy problems were referred to SSD/Aerospace and NASA GPO for resolution.

The panel was responsible for generating the Gemini/Agena Interface Specification and Control Document (ISCD-2), using the same format as that of the ISCD-1. This formal document established and maintained Gemini Spacecraft/Agena Vehicle/Target Docking Adapters/Ascent Shroud Interface requirements and configuration. It contained requirements (both in text and drawing form) covering the mechanical, electrical, and AGE interfaces and the associated testing necessary to validate interface integrity.

The activities of the Interface Control Panel were summarized at each Atlas/Agena Coordination Meeting.

c. Trajectory and Orbits Panel

Matters relating to the Atlas/Agena launch, rendezvous, EVA, and post-rendezvous maneuvers were included in the agendas for the Trajectory and Orbits Panel, which were discussed in Section II. A. 2. e of the GLV. The GATV was represented by SSD and LMSC.

d. GAATV Ascent Guidance Working Group

This group was established and chaired by SSD to coordinate all GAATV ascent guidance activities and resolve all associated problems. The participating agencies were SSD, MSC/GPO, MSC/FOD, LMSC, GD/C, TRW, GE, and Burroughs.

e. Operational Interface

Early in the program many discussions took place between NASA, the Air Force, and LMSC concerning LMSC support to NASA FOD at Houston. As a result of these negotiations, three LMSC engineers were stationed in Houston to improve the liaison between LMSC Sunnyvale, and FOD Houston.

In Summer 1965, as the GATV program approached the operational phase, it became apparent that the Air Force and LMSC support to FOD could be greatly improved by sending a highly qualified select team of systems engineers to Houston for each GATV mission. This team, composed of LMSC engineers, SSD officers, and one Aerospace representative, provided technical consultation service to FOD, manned a console in the MCC Staff Support Room, and performed real-time evaluations of problems occuring in flight.

4. PROGRAM DOCUMENTATION

The general documentation requirements for the GATV Program were specified in the Contract Statement of Work, AF 04(695)-545, and in general followed standard program documentation. The following documents are deemed unique to the GATV Program in scope and/or application.

1) TOR Document, Gemini Agena Target Vehicle Procedure and Requirements for Acceptance established and defined the role of the Gemini Agena Target Vehicle Acceptance Team (VAT); listed the documentation required to support the VAT; and set forth the criteria, procedures, and ground rules used by the VAT.

2) The Gemini Human Engineering Program Plan described the mission and vehicle-peculiar equipment, design, aerospace ground equipment modification, and new design to provide proper allocation of functions to men and equipment. It established equipment layouts for efficient operation and ease of operation and maintenance. The plan indicated the method of accomplishing the general requirements of AFSSD Exhibit 62-44A and MIL-M-26512-B.

3) The Gemini Milestone Network Report provided a schedule of milestones and a report about them, utilizing the Program Evaluation Review Technique (PERT). The number of networks required and the scope of each were established by the PERT team. A full scale PERT program however, was never implemented.

4) The Detail Specification for the AGE at Merritt Island, NASA Radar Boresight Range covered the design, construction, performance, and test requirements necessary to permit the AGE to support NASA-directed tests of the GATV and Gemini spacecraft at Merritt Island, NASA Radar Boresight Range.

5) Gemini Agena Target Vehicle Familiarization Handbook provided a thorough familiarization of the Gemini Target Vehicle (GTV) and the function and operation of its various subsystems and components. It was revised immediately after systems test of the scheduled first flight GATV (5001). The document was also revised several times to satisfy other vehicle changes. It provided a comprehensive description of the Agena Target Vehicle hardware and its operation, including the airframe, PPS, SPS, power system, guidance system, flight control system, tracking system, command system, telemetry system, TDA, Agena Status Display Panel, flight termination system, and command functions.

6) Gemini Agena Target Operational Capability Handbook developed into a detailed drivers manual for those who were to have control of the GAT on orbit. The OCH defined the recommended operating procedures for the GAT for the orbital phase of the Gemini mission (docked and undocked) and the contents were valid for any mission plan. Since this handbook was primarily an operating procedures manual, it was necessary to emphasize operational constraints imposed by GAT hardware in orbit as determined by engineering analyses and tests. The constraints were categorized as Prohibited Operations, Mandatory Operations, Operational Restrictions, and Operational Requirements.

7) Gemini Altas Agena Target Vehicle System Ascent Guidance Contacts, Flow Diagrams, and Documentation List was published by SSVT on 4 March 1965. Section A contained the name, telephone number, and address of key persons directly involved with GAATV ascent guidance. Section B contained the GAATV ascent guidance implementation cycle as described by flow charts. Section C contained all GAATV definitive ascent guidance documentation and showed the documents needed by each agency to carry out its ascent guidance task.

SECTION III. A

REFERENCES

None

SECTION III. A

DEFINITIONS

AFSC	Air Force Systems Command
AGE	Aerospace Ground Equipment
AGM	Assistant General Manager
C and C	Command and Communication
DEV	Development
DOD	Department of Defense
EGO	Eccentric-Orbit Geophysical Observatory
EVA	Extra Vehicular Activity
FOD	Flight Operations Directorate
GAT	Gemini Agena Target
GATV	Gemini Agena Target Vehicle
GAATV	Gemini Atlas Agena Target Vehicle
GD/C	General Dynamics/Convair
GPO	Gemini Program Office
ISCD	Interface Specification and Control Document
LMSC	Lockheed Missiles and Space Company
MAC	McDonnell Aircraft Corporation
MCC	Mission Control Center
MSC	Manned Spacecraft Center
MSVP	Medium Space Vehicles Programs
NASA	National Aeronautics and Space Administration
OAL	Orbiting Astronomical Laboratory
OCH	Operational Capability Handbook
POGO	Polar Orbiting Geophysical Observatory
R and D	Research and Development
RCAF	Royal Canadian Air Force
REL	Reliability
SCTB	Santa Cruz Test Base, LMSC
SLV	Standard Launch Vehicle
SSD	Space Systems Division, USAF
TRW	Thompson Ramo Wooldridge

B. PROGRAM DESCRIPTION (METHODS OF OPERATION)

1. PHILOSOPHY AND HISTORY

The Gemini Agena Target Vehicle "Pilot Safety" requirements differed in one major way from those of the Mercury Atlas and the Gemini Titan Launch Vehicles. It is significant that the unmanned ascent phase of the GATV mission provided a severe test of the ability of the vehicle to accomplish the mission's "manned" phase, i. e., rendezvous and docking in orbit. This allowed mission flights to be used as proving flights. For example: the significantly modified but not yet flight tested 8247 engine was first flown on the target vehicle for Gemini VI. The loss of this vehicle occurred shortly before orbit insertion and well before the GLV launch and thus did not endanger the astronauts. The engine failure, however, led to a major review of hardware performance.

It was in this less demanding environment that a modified Pilot Safety program was adopted in which cost considerations influenced decisions signficantly more than in the Gemini Launch Vehicle Program. However, many of the concepts which were so successful with the Gemini Launch Vehicle were integrated into the target vehicle program in a manner consistent with the difference in mission requirements and with the fact that the target vehicles were initially accepted as Standard Agenas and then modified in the GATV program.

Close coordination between NASA, SSD, and LMSC was maintained through a series of monthly technical direction meetings to ensure that program objectives were being met. Aerospace Corporation Gemini Launch Systems Directorate was given a Technical Surveillance role on the GATV late in 1964 and, as its first responsibility, reviewed the status of the program and the adequacy of pre-flight testing. Although this was after design and fabrication of the first flight article, it was possible to include in the program some additional Mercury/Gemini Launch Vehicle Pilot Safety concepts:

1) A Critical Components History and Review Program.
2) A detailed review of all program failure analyses by an independent agency.
3) An independent detailed review of all subsystem and system data taken during pre-acceptance and acceptance testing.
4) An independent detailed review of data taken at ETR.

Detailed design reviews were conducted by NASA, SSD/Aerospace and LMSC. The review prior to the first launch resulted in such things as the decision not to fire the GATV primary propulsion engines with the manned spacecraft docked with the GATV until after there had been a successful primary propulsion system (PPS) firing after the ascent burn. The review conducted in conjunction with the "Sure-Fire Program" after the initial Agena failure resulted in locking out an overspeed shutdown circuit during ascent to prevent an unnecessary vehicle shutdown during that period.

2. CONFIGURATION CONTROL - ECP REVIEW

The configuration management and control requirements for the GATV System were in general agreement with AFSCM 375-1[1] and ANA Bulletin No. 445[2] although the GATV contract statement of work made minor modifications. Before the imposition of the aforementioned military documentation into the LMSC contract, a form of configuration control was in existence by virtue of certain LMSC procedures and practices. When the contractual requirement for configuration management was established by SSD, a Configuration Management Office was established within the program office and implementation plans for compliance with AFSCM 375-1 and ANA Bulletin No. 445 (including First Article Configuration Inspection (FACI) and Engineering Change Proposal (ECP) preparation) were issued. Serialization/effectivity programming of all changes, both Class I and Class II, was initiated by the Gemini Program for additional control of change activity and traceability.

The FACI of the GATV System was performed at Sunnyvale in conjunction with the acceptance activity of the first vehicle. The baseline configuration for the vehicle was established at the time of the first formal attempt to accept Vehicle 5001. FACI of Complex C-10 at Sunnyvale was accomplished by SSD/Aerospace after the Vehicle 5001 system acceptance test, which occurred subsequent to hot firing and the anechoic chamber radio frequency interference test. Hangar E ground equipment FACI was accomplished after Vehicle 5001 checkout and subsequent to the r-f compatibility test at Merritt Island. The FACI of the Merritt Island aerospace ground equipment (AGE) was accomplished after Vehicle 5001 checkout. Pad 14 AGE equipment FACI was accomplished after the first Gemini Agena Joint Flight Acceptance Composite Test. In each instance of FACI, an approved specification in accordance with the requirements of Exhibit II of AFSCM 375-1 as well as complete engineering drawings was required.

After completion of the FACI's and the establishment of the configuration baselines, the re-identification of hardware pursuant to change activity followed the strict interpretation of the appropriate military documentation.

The requirement for the initiation of an ECP came from one of the following agencies: NASA, SSD/Aerospace, McDonnell Aircraft, General Dynamics/Convair, Standard Agena office or within the LMSC program office. Requests for ECP preparations from other than the LMSC organization were always transmitted to SSD initially, who in turn decided upon the desirability of having LMSC prepare and submit the ECP to the SSD Configuration Control Board (CCB) for disposition.

This board was composed of the Chief of the GATV Directorate as the chairman and the following: members who represented each office in the GATV Directorate, Aerospace Gemini program office; NASA and the Air Force Plant Representative's office. In certain instances due to priority or urgent conditions, LMSC personnel were in attendance to present their ECP's.

The LMSC GATV configuration management office (CMO) established certain internal practices and policies for compliance with their contractual direction in AFSCM 375-1 and ANA Bulletin No. 445. The LMSC CCB functioned as a clearing house for all GATV change activity. After a short time, it became apparent that certain in-house functions were enhanced as a direct result of the CCB activity. Cost effectiveness and contract requirements were considerably improved. Engineering and support personnel were able to reduce the length of time spent on the reconciliation of changes to hardware and data and documentation by utilizing the configuration change accountability and traceability reports. The "Gemini Vehicle Configuration History" report[3] was especially useful in this respect. It was distributed weekly and issued separately for each vehicle. It traced the identity and change history of each item of vehicle hardware controlled by the GATV Detail Model Specifications.[4] This report was one of the basic tools used by SSD/Aerospace during vehicle acceptance and the configuration audit at ETR.

The LMSC CMO assumed responsibility for assuring that all Class II engineering changes were furnished to the AFPRO for review and concurrence prior to their incorporation through the engineering release system.

The Controlled Configuration Article system for parts accountability was a significant aid in providing traceability. Each component designated, was given an eight digit CCA number. The first four digits identified the component though not its exact configuration and the last four digits were the unit serial number. Since LMSC kept detailed records on the history of articles by CCA number, this was the basis for screening to purge the system of suspect or nonflightworthy items.

Prior to final acceptance of each GATV at the factory in Sunnyvale, representatives from SSD/Aerospace and the AFPRO conducted a configuration review of the GATV, its associated AGE and the applicable drawings, data and documentation.

A final SSD/Aerospace configuration review of the GATV and its associated AGE was performed at ETR in conjunction with the LMSC GATV Launch Audit. All ECP's scheduled for incorporation since shipment from the factory were reviewed to insure proper implementation, validation and accountability. All ETR test results were reviewed for completeness, compliance with the applicable specifications and to assure compatibility of the AGE with the particular vehicle undergoing test. The major replacement items were reviewed to assure authorized configuration and finally the critical spares list for support of the launch was reviewed to insure that the configuration of the spares was correct and compatible with the configuration of the GATV on the launch pad.

3. RELIABILITY

a. Quality Control

The failure reporting, analysis and corrective action system was a function that was common to the reliability and quality programs. Both the reliability and quality control programs were carried out by NASA Programs Product Assurance at LMSC, which could in this way assure effective coordination of all of the activities concerned. Failures and other discrepancies included in the system were those that occurred in receiving inspection, manufacturing, final assembly, systems test, and launch site tests. Flight failures were handled separately.

(1) Failed Equipment and Discrepancy Report (FEDR)

When a failure or discrepancy was observed, the proper Product Assurance (inspection) representative initiated a Failed Equipment and Discrepancy Report (FEDR). A reliability representative immediately conducted an on-the-spot failure investigation. Failure analysis and corrective action followed. Disposition of the discrepant material or equipment (i. e., rework, repair, use-as-is or scrap) was made by a Material Review Board (MRB), and was based on the judgment of cognizant engineering as well as the reliability analysis. After all action had been completed, the FEDR was signed by MRB and AFPRO and given general distribution. One copy of each FEDR was forwarded to SSD by Product Assurance. A daily listing of significant FEDR's on vehicle equipment was distributed to responsible organizations and to AFPRO within twenty-four hours of a failure. A significant FEDR was one that included a trouble or failure (but not a nonfunctional discrepancy) that occurred on a vehicle during system test or at the launch base.

(2) Corrective Action Repair and Diagnosis (CARD)

NASA Programs Product Assurance at LMSC initiated an analysis of failed equipment, as soon as the failure had been verified. On the Gemini Program the analysis was taken to as low a level as necessary to determine the actual cause of the failure or discrepancy, and to provide sufficient information to effect corrective action. On all critical equipment failures the analysis was performed down to piece part level.

In the case of nonfunctional discrepancies both analysis and ensuing corrective action was performed on the spot, insofar as was possible. However, in ordinary circumstances a CARD was initiated to document each of the one or more steps required for an analysis. A CARD was assigned to a cognizant organization, and that organization was required to provide specific information or action.

Documentation of this action was made on the CARD, and it was then returned to the reliability engineer responsible for the analysis. Wherever it was advantageous, a CARD, accompanied by a discrepant component, was assigned to the diagnostic test laboratory for analysis. Additional CARD's were assigned until a definite cause of failure had been determined. Failure analysis, exclusive of diagnostic testing, was completed expeditiously. Normally, two weeks were allowed for completion of a CARD. However, on the Gemini Program, special effort was made to expedite CARD closure, usually on an around-the-clock basis, on discrepancies occurring in the manufacturing electronics or in systems test areas when it was necessary to maintain schedules. In the course of the failure analysis of a discrepancy, a CARD was assigned to investigate a specific factor of that discrepancy. When it was found that a particular item appeared to have more than one failure mode or that a failure mode was associated with a manufacturing process and repeated itself on several items, special failure studies were initiated under the reliability program. In these studies, the reliability engineer examined all FEDR's and CARD's associated with a recurrent mode to search for a more basic cause of failure and to correlate the group of discrepancies. Each study was documented and included a listing of related failures, together with recommendations for preventing their recurrence; the recommendations were assigned as action items to the organization that could resolve the problem. Copies of reports were maintained by NASA Programs Product Assurance at LMSC for review by SSD and Aerospace.

As a result of the reliability investigation and failure analysis, the action necessary to correct the discrepancy, and to preclude its recurrence, together with the organization assigned, was indicated on the FEDR or CARD. If problems crossed organization lines, or responsibility was disclaimed by an assigned organization, the unresolved item was referred for assignment to a Corrective Action Committee, made up of individuals from the several organizations. When the corrective action was completed it was documented on the form. In the case of analysis or corrective action required by a supplier, the procurement organization coordinated the processing of the CARD. Following verification (by re-inspection, re-analysis, etc.) the CARD was closed out by NASA Programs Product Assurance at LMSC, and distribution made in the same manner as the FEDR. Although a maximum of two weeks was allowed for corrective action assigned to an LMSC organization or six weeks when assigned to a supplier, CARD's on the GATV Program were expedited. SSD/Aerospace reviewed all CARD's, along with FEDR's, to assure themselves of the adequacy of their content prior to acceptance of closeout. If additional information had to be provided to accept the conclusions on the CARD, Aerospace requested this information of LMSC, either directly or by going through SSD.

(3) Urgent Action Survey (UAS)

This survey was a method used for rapid dissemination of information related to discrepancies, failures or troubles that required immediate management action to prevent extensive damage or destruction to equipment. Although at least some of these surveys resulted in prevention of reliability degradation (e.g., from stock sweep of suspected items), they were, nevertheless, included under the quality program.

b. Surveillance - Agena Family Flights

The failure reports and subsequent analyses from all NASA Agena programs were continually monitored in search of repetitive failures that might affect Gemini Agena hardware. When these were noted, a problem report was established, and given intensive study by NASA Product Assurance at

LMSC, along with SSD/Aerospace. A report defining the common problem and indicating the steps to be taken to achieve a solution, was published. Problem area reports were published in monthly discrepancy analysis reports. The Gemini program derived a direct benefit from these reports. Gemini Agena Target Vehicle telemetered data received during the period from liftoff through mission completion was also analyzed by data groups for detection of failures. Responsible subsystem engineers, assisted by a Product Assurance reliability representative, verified the failure and recorded it on a FEDR. Task assignments for further investigation or diagnosis were made by issuing a CARD as directed by the reliability engineer. Corrective action that applied to subsequent vehicles was assigned in the same way.

AFPRO/SSD/Aerospace had the opportunity to conduct their own reviews of the reliability program at the LMSC facilities at any time. The Product Assurance group at LMSC maintained working data files for all Agena programs, in which data pertinent to the reliability programs were available for these reviews. Records of the control and audit system and the failure reporting system, together with copies of documents that had been reviewed, were included in these files. In addition, all organizations concerned delivered to NASA Product Assurance at LMSC copies of reports, analyses, and other documents prepared in the course of the Gemini Agena reliability program and required by this plan. These copies were forwarded to Product Assurance as they were generated. The customer was given full opportunity to assess the reliability program and to determine that the contractual reliability requirements were being met.

c. Continuous Review of Contractor and Vendor Problems; Failure Analysis

(1) Monthly Progress Report

The Gemini Agena Target Vehicle (GATV) Program Progress Report was a monthly report on all phases of the Gemini program, indicating significant progress accomplished during the report period. The Product Assurance section of this report contained paragraphs pertaining to reliability activities required by this program plan. Reliability Program Status Reports (including significant task accomplishments, updated milestone schedules, and summary of reliability budget status for major tasks) the Reliability Estimate, and Analysis Reports (including quantitative reliability estimates of vehicle systems and subsystems of GATV's, as well as an accompanying analysis of the reliability) were included within the Product Assurance section of the GATV Program Progress Report.

(2) Discrepancy Analysis of Preflight Equipment Failures

This report was an objective analysis of preflight equipment failures encountered during the reporting month on the Gemini Agena Program. Summaries of vehicle functional discrepancies were statused each month, pertinent vehicle nonfunctional FEDR's were presented for updating information, status of problem areas were indicated, and component functional discrepancies were listed for the month in which they occurred. A cumulative summary of rejected vehicle components was also included. In addition to this report, copies of all GATV FEDR's, CARD's and UAS's were forwarded to SSD and Aerospace on a weekly basis. A perpetual summary of all GATV FEDR's and CARD's, including their status, was maintained by LMSC/NASA Product Assurance and Aerospace.

In order to prevent the introduction of degrading effects that would reduce reliability, surveillance of manufacturing, assembly, and test activities was required. In the manufacturing electronics area, particularly, normal inspection was not sufficient; the unusual demands inherent in GATV man-in-space requirements, together with the need to utilize previously untried methods, precluded any routine operation.

In lieu of a reliability testing program, reliability engineers monitored the manufacturing and fabrication of electronic components peculiar to the Gemini vehicle in order to participate immediately in correction of inadequate practices and processes as they were detected and to modify inspection plans accordingly.

Reliability engineers also maintained surveillance of all stages of GATV production; final assembly, hot-fire, and systems test. This provided additional assurance that design reliability integrity was being maintained and that inspection and test requirements were properly interpreted. Further, there was an opportunity to present the reliability point of view during resolutions of problems encountered during systems test.

A continuous review and surveillance of contractor and vendor problems and failure analysis was maintained by Aerospace in close coordination with SSD, LMSC cognizant personnel and vendor personnel. This included FEDR, CARD and accompanying paper review, and review of those failed components, and/or subsystems in failure analysis.

d. Extra Care Program

In order to place constant emphasis on the need for quality workmanship and high reliability in the Gemini program, Lockheed Missiles & Space Company inaugurated the Gemini Extra Care Program, June 3, 1964, in cooperation with the U.S. Air Force and NASA. This program was designed to make certain that the Agena Target Vehicles were, in every way, as near perfect as they could be made. Over 2000 employees in appropriate Agena Target Vehicle departments were specially selected for Gemini team work. Selections were based on good performance, knowledge of work, seniority and attitude.

Through the Gemini Extra Care Program there was increased inspection of all work, and constant reminders of the need for Extra Care. All major components and significant parts in the work flow were marked with special Gemini stamps and decals. All related paper work was identified in a similar manner. Gemini work areas were identified with prominently displayed signs. Motion picture showings and special brochures kept Gemini teams abreast of the Manned Space Program. Regular newsletters and articles kept them well informed of Extra Care and Gemini progress.

Monthly and semi-annual performance awards were presented to those team members who exhibited outstanding performance, and who made special contributions in the Gemini program. Quarterly group awards were made in recognition of a work section's outstanding achievements during the quarter. Employees who submitted suggestions leading to higher reliability, quality, or process improvement also received special awards.

The success of the Gemini Extra Care Program is indicated by the drastic decrease in workmanship FEDR's (Figure III. B-1) recorded in the Gemini final assembly area. It is significant to note that the forward auxiliary racks for 5002, 5003, 5004 and 5006 were fabricated with no workmanship FEDR's. In fact, the 5006 rack did not have a single FEDR or discrepancy during its fabrication - over 27,000 separate operations.

Special skill training was given to Gemini employees as the need arose. Gemini managers and supervisors attended training sessions on the principles and techniques of employee motivation. Top management personnel toured Gemini areas regularly to extend support. Continuous morale assessment and follow-up corrective action, if necessary, by Extra Care coordinators, helped maintain a strong team spirit and attitude in the Gemini work areas.

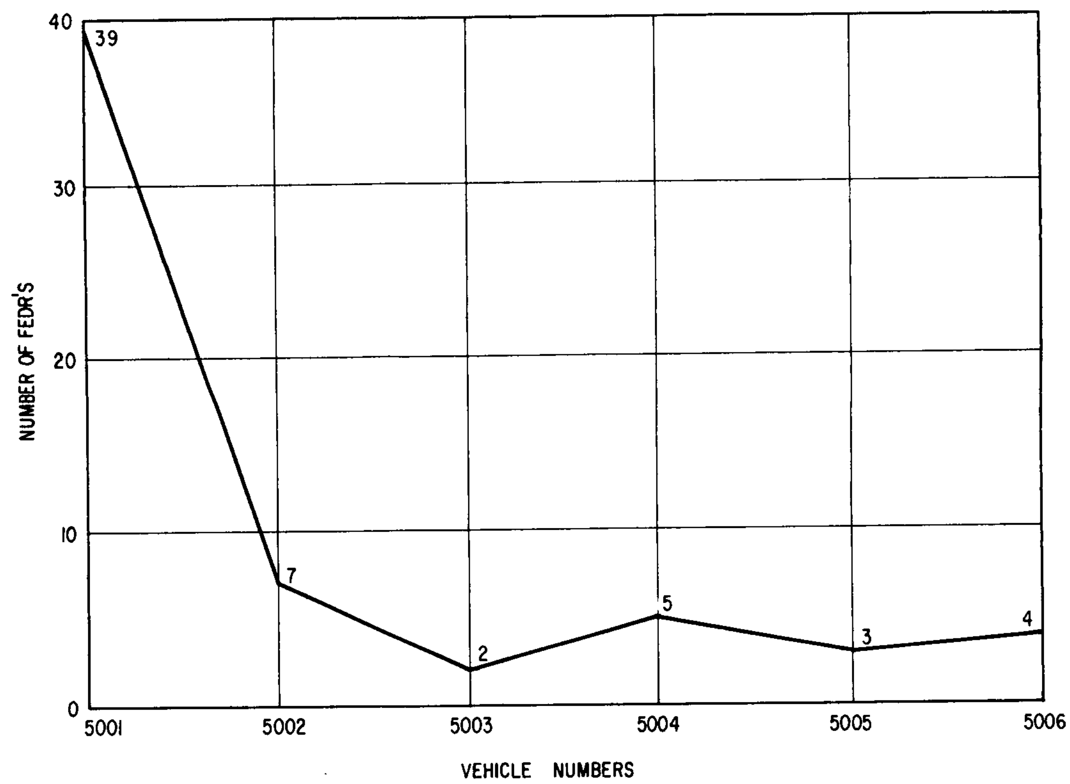


Figure III. B-1. Workmanship FEDR's

4. ACCEPTANCE

a. Engine Acceptance (PPS/SPS)

The mission plan required that the spacecraft be docked with the Agena vehicle in both the passive and active mode. It was therefore necessary to re-rate the primary and secondary propulsion systems for operation in a manned configuration. Within the limitations posed by the program, a systematic component/system review of the engine was established.

The review was patterned after the Gemini launch vehicle propulsion review and the primary purpose was to establish that:

- Hardware met the performance requirements of the model specification.
- Hardware used with discrepancies did not compromise system integrity.
- When a failure occurred a component failure analysis was performed.
- Corrective action was taken when deemed necessary.

The acceptance documentation for the engine systems was reviewed by SSD/Aerospace at the Bell Aerosystems Corporation facilities in Buffalo. This review was, when possible, made prior to the shipment of the engines from the Bell plant.

The following is a list of the major documentation reviewed.

- Acceptance Test Log. This document contains all acceptance data, anomalies, deviations, replacement summaries and configuration.
- Component History Card. This card contains the complete history of a component including a record of problems encountered in manufacturing and after component acceptance.
- Vendor Discrepancy Records. This form describes discrepancies encountered with vendor components after receipt of these components at the Bell plant.
- Failure Reports. This is primarily used when a functional component fails after it had been previously accepted.
- Failure Analysis and Corrective Action Report. Each failure report is followed by this report.
- Materials Review Report. This form is primarily used during manufacture to report non-functional discrepancies.
- Functional Test Reports. This is used by the inspection laboratory to describe all of the functional tests required for component acceptance.

After completion of the review, recommendations were made by Aerospace to SSD to either accept the system or to direct specific rework and retest.

b. Vehicle Acceptance

(1) Philosophy

The GATV vehicle acceptance plan was patterned after the one used on the Gemini Launch Vehicle but with such modifications as necessary to achieve its objectives in a manner compatible with LMSC operations and procedures. The plan consisted of a review of data, contractor failure reports, contractor documentation including critical component histories, and finally a physical inspection of the hardware. The acceptance requirements and procedures to be followed by the SSD/Aerospace vehicle acceptance team (VAT) and LMSC were detailed in an Aerospace Corporation report[5] which was formally made a part of the LMSC contract in a Contract Change Notice.

(2) Subsystem Data Review

Acceptance of a vehicle for shipment to Cape Kennedy depended upon a thorough detailed customer review of the data accumulated by subsystem and system testing in the LMSC C-10 test complex.

These tests were laid out by task number, with Tasks I through VI designated for vehicle and subsystem preparations, and Tasks VIII through X designated as final system tests used as a basis for final vehicle acceptance. Task VII was a test used early in the Agena program and subsequently discarded.

Tasks I through VI provided verification of: vehicle wiring and interconnections with the test complex; power hookup including measurement of magnitudes of voltage, current, ripple and noise; range safety command system; guidance and flight control; communications and control; and the Agena/ target docking adapter (TDA) interface.

The final systems tests were based upon a simulated countdown and flight, both ascent phase and orbital, with data recordings of the time and event functions which would occur during a mission.

Both subsystem and final tests included data for checking appropriate event sequences against time (go no-go) and quantitative recordings of gain levels, signal responses, etc. All data from both sets of tests were subjected to a detailed evaluation, and all questions and discrepancies were resolved prior to vehicle acceptance.

(3) Discrepancy Reports.

Past experience on other programs had shown that some formal method was required to expedite the resolution of problem areas and to answer questions which originated during the several VAT review activities. Since LMSC already had in operation a discrepancy reporting system within their product assurance organization, it was decided to use that basic system and adapt it to the needs of the VAT. This was accomplished through coordination with the Program Office and Product Assurance personnel.

A summary of the discrepancies written by the VAT is shown in Table III. B-1.

Table III. B-1. Summary of Discrepancies Written by the Vehicle Acceptance Team

Activity \ Vehicle	5001	5002	5003	5004	5005	5006	5001R
SYSTEMS DATA REVIEW	46	41	16	8	7	6	5
CRIT. COMPONENT DATA REVIEW	N/A	N/A	7	15	9	8	16
PHYSICAL INSPECTION	N/A	12	24	7	4	8	11
TOTALS	46	53	47	30	20	22	32

(4) FEDR Review

A preflight analysis of each GATV was conducted by LMSC during the vehicle acceptance activities. The primary purpose of these activities was to review the quality of the particular vehicle and, having corrected any discrepancies, present it for customer acceptance. During this review a reliability analysis of the vehicle under investigation was required as one of the steps of vehicle acceptance. The latest estimate of reliability, results of reliability tests, qualification status, and corrective action resulting from failure reports were all included in this analysis.

Prior to acceptance of the GATV vehicle, all vehicle logs were reviewed for any quality and reliability associated problems. This included a complete review of all FEDR's and associated paper. Where failure analysis, vendor replies, or additional work was required to satisfy the cause and corrective action of the problem, the FEDR was left open until such time as all individuals concerned with the problem, including LMSC/Aerospace/SSD design engineering and quality control, had approved the FEDR and associated action.

The working relationship on anomalous FEDR's requiring additional information was an informal one, due to the nature of the contract. However, LMSC quality did comply with Aerospace/SSD requests for additional information, when needed and requested, to complete a FEDR history and/or to close out a problem.

A FEDR summary is shown on Table III. B-2.

(5) Critical Component Review

Twenty-five (25) GATV components considered critical to the safety of the astronauts were selected by SSD/Aerospace for special attention. LMSC was contracturally required to assemble, summarize, review and maintain the detailed history of each of these components for each vehicle and the flight-approved spares.

Prior to acceptance of each vehicle, each critical component data package was reviewed by LMSC (Engineering, Program Office, Quality and Reliability), SSD and Aerospace. Each package contained, as a minimum, the following:

1) Title Page
2) Table of Contents and Check Sheet
3) Chronological History of Significant Events
4) Data - Including in part:
    a) FEDR's
    b) CARD
    c) MRB reports
    d) Calibration Data
    e) Operating Time
    f) Manufacturing Test Data
    g) Acceptance Test Data

Table III. B -2. FEDR Summary

Vehicle	Final Assy F	Final Assy NF	VST F	VST NF	ETR F	ETR NF
5001	11	128	75+	119x	13	22
5002	2	24	17	48	32	19
5003	5	28	20	27	12	15
5004	10	31	5	8	18	10
5005	3	26	4	9	6	14
5006	0	38	8	8	9	11
5001R	7	107*	7	5	7	6

\+ Includes 41 from SCTB

x Includes 48 from SCTB

* Includes 85 from refurbishment of 5001 following use of 5001 for initial testing at ETR.

F Functional

NF Non-functional

A comprehensive technical evaluation of these data was made by the contractor and customer technical teams. Components were removed and replaced on vehicles (and retested), when information found in the data packages cast doubts on the integrity of the components.

(6) Physical Inspection.

In keeping with the "Extra Care" policy instituted for the GATV program, a VAT physical inspection team was created to give an "outsiders' look" at the vehicle. Whereas the LMSC and Air Force product assurance and quality control inspection teams were bounded by contractual requirements during these inspections, the VAT inspection team was authorized to squawk anything which looked unsound - even though it may have been per blueprint. This approach proved to be beneficial in that in several cases where an item was per print but a borderline case, a reanalysis of the situation resulted in equipment change-out, a minor redesign or other remedial action.

5. PRE-LAUNCH ACTIVITIES

Launch preparations for each vehicle were of a continuing nature but there were several regularly scheduled pre-launch activities which in themselves were of value because they were "forcing function" status reviews. These in chronological order were:

1) An SSD/Aerospace "Scrubdown" at LMSC conducted approximately a month prior to launch.
2) An "Technical Review" conducted for NASA MSC about 3 weeks before launch.
3) An LMSC "Launch Audit" conducted during the final two weeks before launch.
4) Technical Review Update, conducted about a week before launch.
5) The Flight Safety Review conducted two days before launch.

A short description of some of these reviews follows.

a. Scrubdowns

The "Scrubdown" was a thorough review of hardware and procedure readiness for launch conducted by SSD/Aerospace with LMSC. All flight problems from other Agena programs were reviewed for their impact on the upcoming GATV flight and those which could influence GATV performance were studied in depth along with the corrective action being taken. Although day-to-day contact was maintained between SSD/Aerospace and LMSC engineers, the scrubdowns did generate new action items which when completed either increased the mission probability of success or at the minimum provided program management with a detailed understanding of risks involved.

b. Technical Reviews

In addition to the normal technical and management program monitoring, a significant design review[6] was conducted prior to the first launch of the Gemini Atlas Agena Target Vehicle. This effort culminated in a formal presentation to an ad hoc group of NASA Directors. Although the data presented to this committee was considerably summarized, all facets of the program were investigated in depth.

A second formal design review[7] was presented to the NASA ad hoc committee after the redesign of the GATV following the engine "hard start" failure of GATV 5002. Although all aspects of the target vehicle were open to reconsideration the emphasis of this second design review was centered around the start system and start characteristics of the 8247 engine.

The Technical Review conducted for MSC prior to each launch consisted of a thorough review of all technical matters which could influence the upcoming flight. It provided MSC with an understanding of open problems and presented to them sufficient facts so that they could evaluate all known flight risks.

The LMSC "Launch Audit" was a Lockheed in-house review of hardware status to insure that all directed action had taken place. Since it was conducted by the LMSC program Chief Systems Engineer it represented an active meaningful review.

c. ETR Surveillance

The difference between the GLV and the GATV effort at ETR was related to the difference in program philosophy and the extent of surveillance required to achieve the basic mission. Accordingly, Lockheed did not make any significant changes in their methods of normal Agena ETR surveillance except for program-peculiar tests.

The Aerospace technical surveillance role permitted their participation in the Flight Test Working Group and all significant activity at ETR

d. Flight Safety Review

The Flight Safety Review Board activities associated with the GATV were the same as those of the GLV and are outlined in the previous section II. B. 8. Flight Safety Review Board.

6. POST-FLIGHT DATA ANALYSIS

Immediately after each GATV launch, and prior to the GLV launch, all data was given a rapid, but thorough analysis in the LMSC hanger at ETR. The data reviewed was from Telemetry Station 2 (TEL 2) and in analog form. This "quick look" allowed Gemini Launch Control to launch the manned vehicle, knowing that the target vehicle had met its objectives.

After the mission, LMSC, SSD, NASA and Aerospace conducted separate ascent data analyses using data available from NASA-Houston, ETR (TEL 2) and LMSC. Particularly after those flights where troubles were encountered, the unfiltered analog data from ETR made possible a more rapid and accurate identification of the problems, than data which had been filtered and computerized. The PCM data received from the Agena was printed point for point to produce analog-type data. The computer gated all data received and discarded all data during temporary synch losses. Much of this discarded data was recovered by careful review of the analog type data rolls.

When the individual studies were completed, joint meetings compared results and the participating organizations arrived at the final results.

While excellent coverage of the ascent of the Agena was possible, the "on orbit" data available ranged from scanty to almost non-existent. Data was normally available only from maneuvers which took place over a ground station, (although up to 20 minutes of taped data could be recovered) and after a few orbits, the number of ground stations was reduced to only one or two.

SECTION III. B

REFERENCES

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   Engineering Changes to Weapons, Systems, Equipment & Facilities
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   Specification No. 1417169, Rev. A, dated 1 March 1966
   Superseded No. 1417169, dated 7 May 1965
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6. GATV Design Certification Report for GTA-6 Mission, dated 5 October 1965
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7. GATV Design Certification Report for Gemini VIII Mission, dated 26 February 1966, Lockheed Missiles & Space Company A794903

SECTION III. B

DEFINITIONS

AGE	Aerospace Ground Equipment.
AFPRO	Air Force Plant Representative's Office.
ATV	Agena Target Vehicle.
CARD	Corrective Action Repair & Diagnosis.
CCA	Controlled Configuration Article.
CCB	Configuration Control Board.
CMO	Configuration Management Office.
ECP	Engineering Change Proposal.
ETR	Eastern Test Range.
FACI	First Article Configuration Inspection.
FEDR	Failed Equipment Discrepancy Report.
GATV	Gemini Agena Target Vehicle.
J-FACT	Joint Flight Acceptance Composite Test.
LSMC	Lockheed Missiles and Space Company
MRB	Material Review Board.
MSC	Manned Spacecraft Center
NASA	National Aeronautics and Space Administration.
PCM	Pulse Code Modulation.
Plan X	GATV Spacecraft Compatibility Test conducted at ETR.
PPS	Primary Propulsion System.
SCTB	Santa Cruz Test Base, LMSC
SLD	Simultaneous Launch Demonstration.
SPS	Secondary Propulsion System.
SSD	Space Systems Division, USAF
TDA	Target Docking Adapter
TEL 2	Telemetry Receiving Station at ETR.
UAS	Urgent Action Survey.
VAT	Vehicle Acceptance Team.
VST	Vehicle Systems Test.

C. GEMINI AGENA TARGET VEHICLE DEVELOPMENT AND CONFIGURATION

1. GENERAL

The Gemini Agena Target Vehicle (GATV) was developed based on mission requirements from NASA/MSC:

1) Establish a circular orbit within specified limits.
2) Provide a stable target with which the spacecraft can rendezvous and dock.
3) Respond to commands from either ground stations or the spacecraft.
4) Perform a complex sequence of orbital maneuvers by means of either real time or stored commands if less than optimum launch of Agena or spacecraft occurs.
5) Provide an active orbit life of 5 days.

These requirements were analyzed by Lockheed Missiles and Space Company (LMSC) to provide design criteria for the unique hardware needed to adapt the Standard Agena D to the Gemini mission. This effort was begun with the award of a letter contract on 1 May 1962, and evolved into the following major hardware items:

1) Modification of the primary propulsion system to provide a multiple restart capability (one ascent burn and four restarts on orbit).
2) Addition of a secondary propulsion system (two 16-pound and two 200-pound thrusters) to provide ullage orientation and minor orbit adjustments.
3) Design of a digital command and communications subsystem including a programmer, controller, PCM telemetry system and on-board tape recorder.
4) Design of changes to provide the guidance and control functions peculiar to the GATV. These included modification of the Standard Agena guidance J-Box, flight control J-Box, and flight control electronics package, and the addition of a new unit, the flight command logic package, to provide the appropriate interface between the guidance and control functions and the command equipment.
5) Addition of an auxiliary forward equipment rack with an interface capable of supporting the target docking adapter.

The sections following will provide more detail regarding these Gemini peculiars. General vehicle configuration is shown in Figure III. C-1.

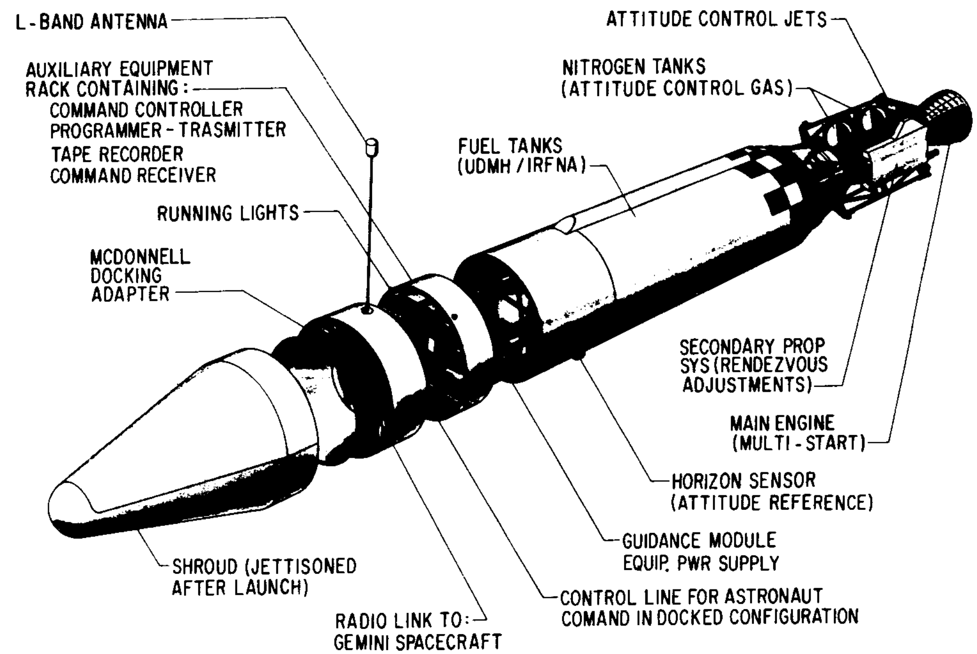

Figure III. C-1. General Configuration - Gemini Agena Target Vehicle

2. AIRFRAME

a. Subsystem Description

The GATV configuration is illustrated in Figure III. C-2.

The target vehicle consisted of seven major structural sections designated as follows:

- Aerodynamic Shroud
- Target Docking Adapter (Government Furnished Equipment (GFE) manufactured by McDonnell Aircraft Corporation)
- Auxiliary Forward Rack
- Forward Section
- Integral Skin/Propellant Tank
- Aft Section
- Booster Adapter

The shroud and the booster adapter were not a part of the orbital vehicle; the shroud was ejected during the ascent phase and the booster adapter remained with the Atlas launch vehicle at separation.

(1) Aerodynamic Shroud

The aerodynamic shroud was a weather-tight, RF-transparent, jettisonable fairing constructed in two segments with a longitudinal parting plane. It was 117 inches long and consisted of a cylinder 23.5 inches long by 65 inches in diameter, blended to a 15-degree half-angle cone topped by a 12-inch-diameter hemisphere. The phenolic fiberglass skin was the main structural member. A redundant nose-cone latch assembly provided positive nose cone-shroud closure. A change that was introduced after the flight of Vehicle 5002 was the modification of the aft actuator bracket to preclude spring hangup which occurred on separation tests for another program.

These failures were on one "cold" ambient test, when one of the two brackets hung up, and on one "hot" simulated-ascent-condition test when both brackets hung up. The hangup was caused by excessive vertical and horizontal translation of the forward-pivot fitting on the shroud about the aft-pivot fitting on the Agena-forward rack, which resulted in the shroud-pivot fitting locking into coils of the actuator spring between the two fittings. The modification applied to the GATV 5003 shroud (and subsequent shrouds) was to widen the lip of the pivot fitting and preclude future hangup.

(2) Target Docking Adapter

The TDA was bolted on the forward face of the auxiliary rack and the shroud mounts on the TDA adapter section, shielding the docking cone during ascent.

(3) Auxiliary Forward Rack

The auxiliary forward rack was a 17.5-inch extension of the forward midbody, peculiar to the Gemini Agena Target Vehicle design. Its purpose was to house equipment peculiar to the GATV, and was in essence part of the payload. It was bolted to the forward midbody and structurally supported the TDA and the shroud. A diaphragm at the forward end of the auxiliary rack controlled the thermal environment of the TDA. The auxiliary rack housed the equipment indicated in Figure III. C-3; access doors were provided for all components and interface electrical connectors.

(4) Forward Section

The GATV forward section consisted of external skin and access doors held and reinforced by three stiffener rings. Within the forward section, a truss-type tubular aluminum frame provided additional strength to the shell and mounting locations for equipment. The forward section housed the equipment shown in Figure III. C-4.

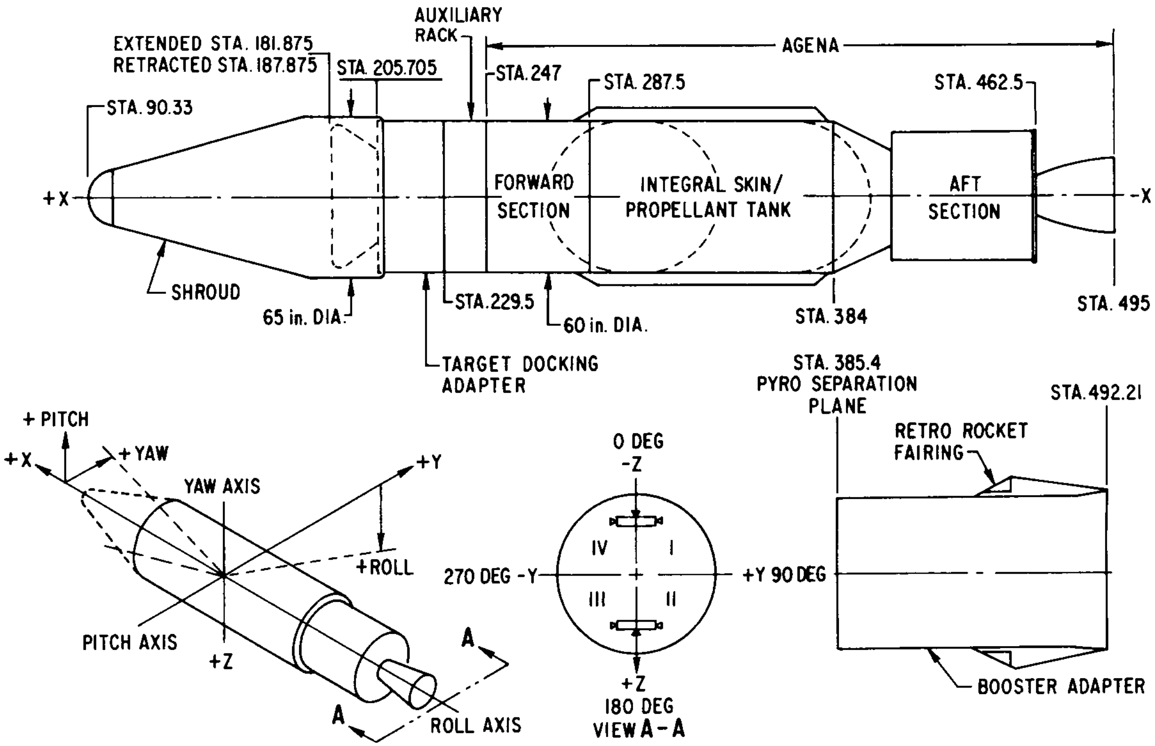

Figure III. C-2 GATV Configuration

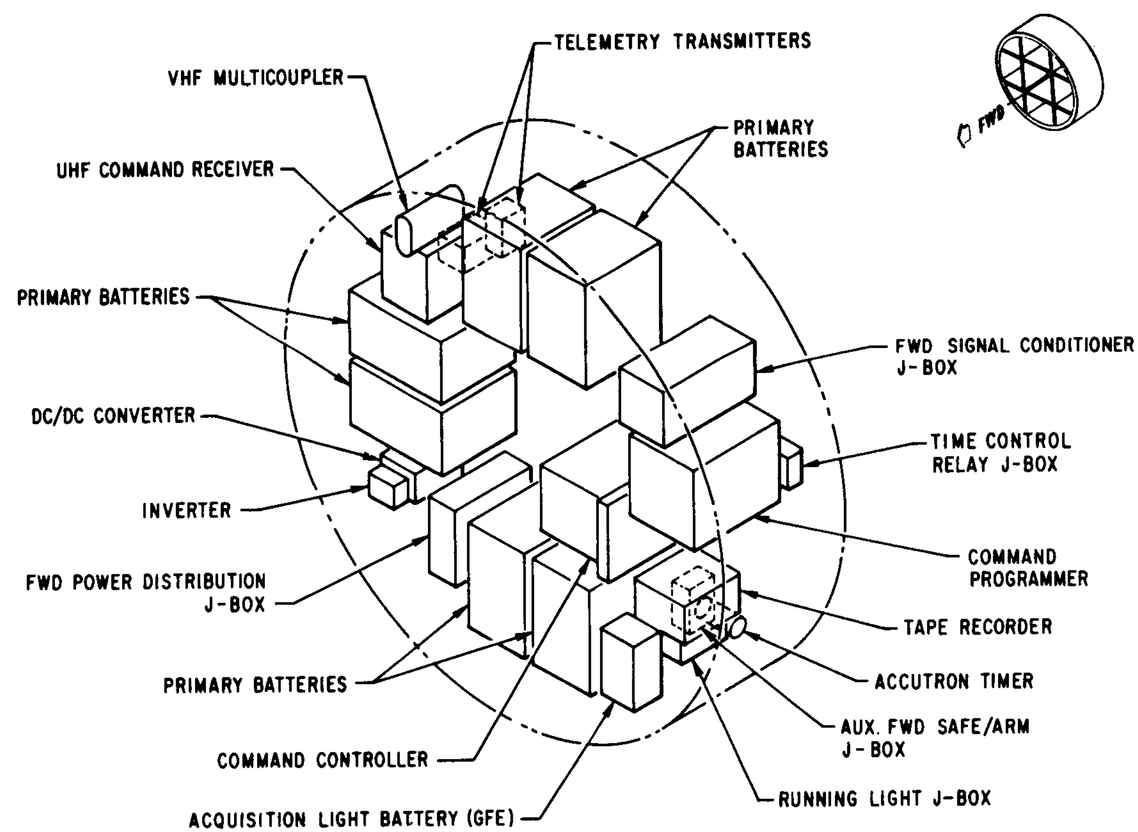

Figure III. C-3 Auxiliary Forward Rack Equipment

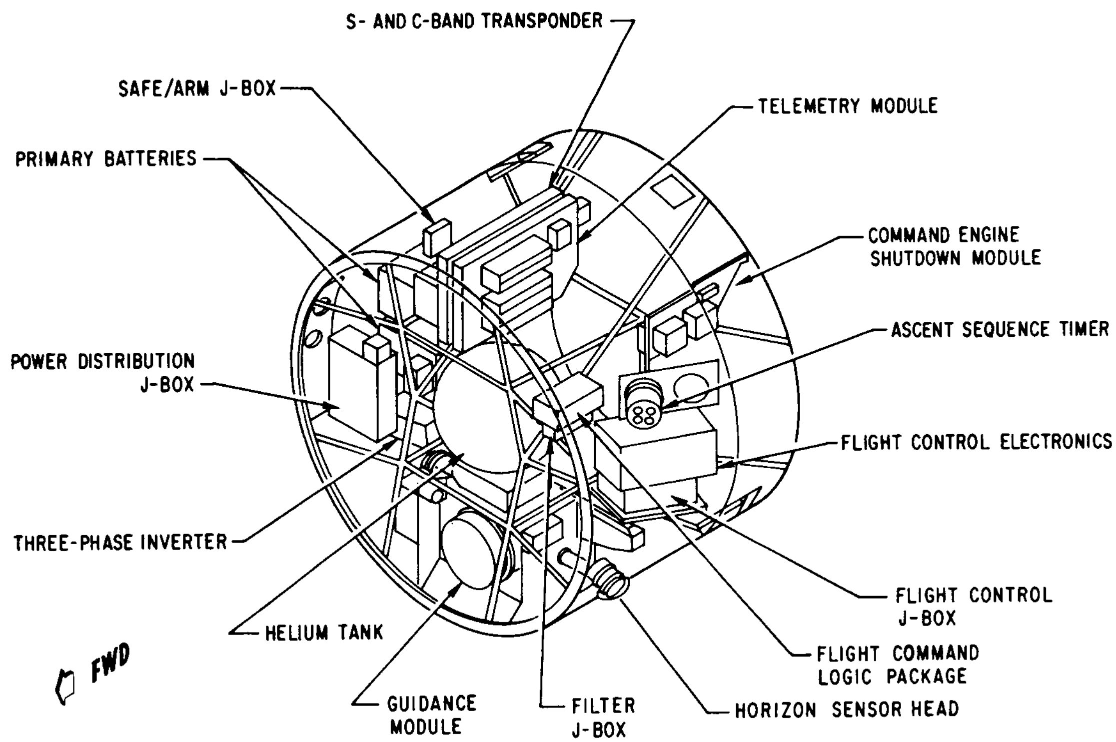

Figure III. C-4 Forward Section Equipment

(5) Integral Skin/Propellant Tank Assembly

The GATV propellant tank was both the vehicle cylindrical skin and a dual-chamber tank assembly. The overall tank length, including the hemispherical ends, was 129.08 inches. The volume of the forward tank was 75.3 cubic feet, and with baffles installed, the nominal capacity was 553.31 gallons (or 3,818 pounds) of unsymmetrical dimethylhydrazine (UDMH). The volume of the aft cell was 98.4 cubic feet, and with baffles installed, the nominal capacity was 738.0 gallons (or 9,700 pounds) of inhibited red fuming nitric acid (IRFNA). Two fairings on the outside of the tank section accommodated electrical wiring, plumbing, and cooling air flow from the forward to the aft section of the vehicle.

(6) Aft Section

The aft section provided the mounting structure for the primary and secondary propulsion systems, the attitude control gas tanks, the thrust valve clusters, and the hydraulic power package. The aft section consisted primarily of the engine-mounting cone and the equipment rack (Figure III. C-5). The aft section was fabricated from the basic Agena thrust cone and equipment rack by adding the following: a modified optional engine-cone-shear-panel kit to support additional weight; program-peculiar shear panels, Secondary Propulsion System (SPS) mounting structure on the aft equipment rack structure (Figure III. C-6); and thermal protection shields over the thrust cone and the SPS units.

A change made after the GATV 5002 flight was the shock-mounting of the following equipment in the aft section:

a) Safe/Arm J-box
b) Aft signal conditioner J-box
c) Aft power distribution J-box
d) Two accelerometer amplifiers (-Z axis forward of Sta 462.5)
e) Turbine overspeed signal detection J-box
f) Turbine overspeed gate and Pilot operated solenoid valve (POSV) (and the relocation of the accelerometers and amplifiers to the aft approximately five feet)

(7) Booster Adapter

The booster adapter connected the GATV to the Atlas booster. The adapter consisted of a 60-inch diameter cylinder that flared to a diameter of 71 inches at the Atlas attachment plane. The adapter was permanently attached to the Atlas booster.

b. Component History

(1) Development Testing

(a) Static Tests. The calculated loads for the Gemini Agena Target Vehicle were less than the design loads for the Standard Agena. Structural tests performed on the Standard Agena program satisfied the test requirements of the Gemini program for the structural elements that were identical. A comparison of expected and design loads is shown in Figure III. C-7 and III. C-8. After a detailed stress analysis was conducted, the Gemini peculiar structure was successfully tested for the GATV design loads. The GATV-peculiar structural tests were as follows:

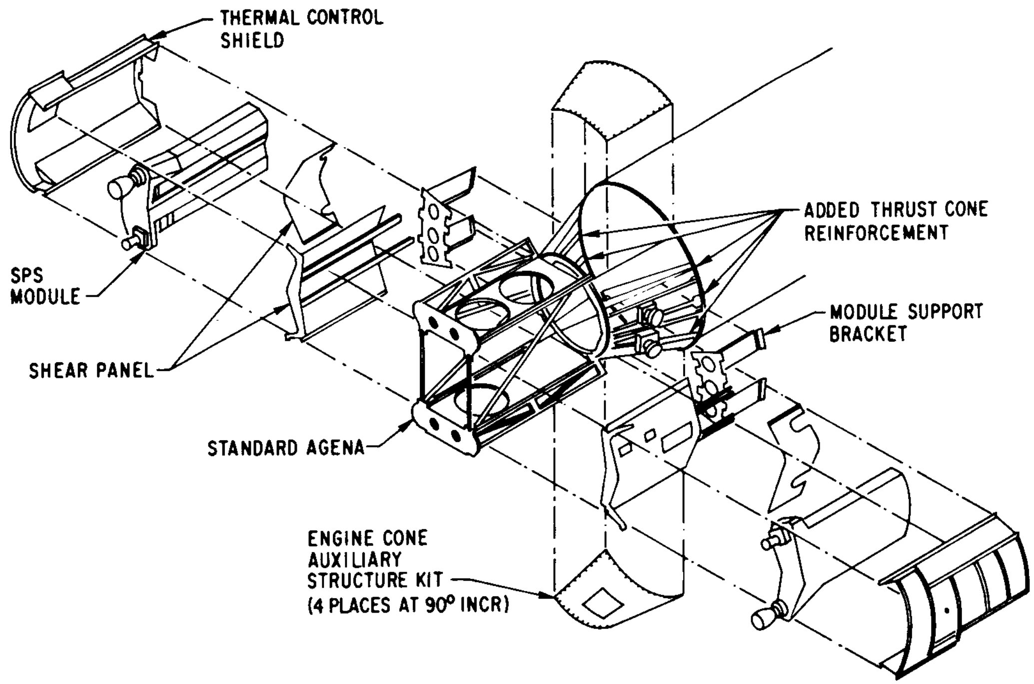

Figure III. C-5 GATV Aft Section Structure

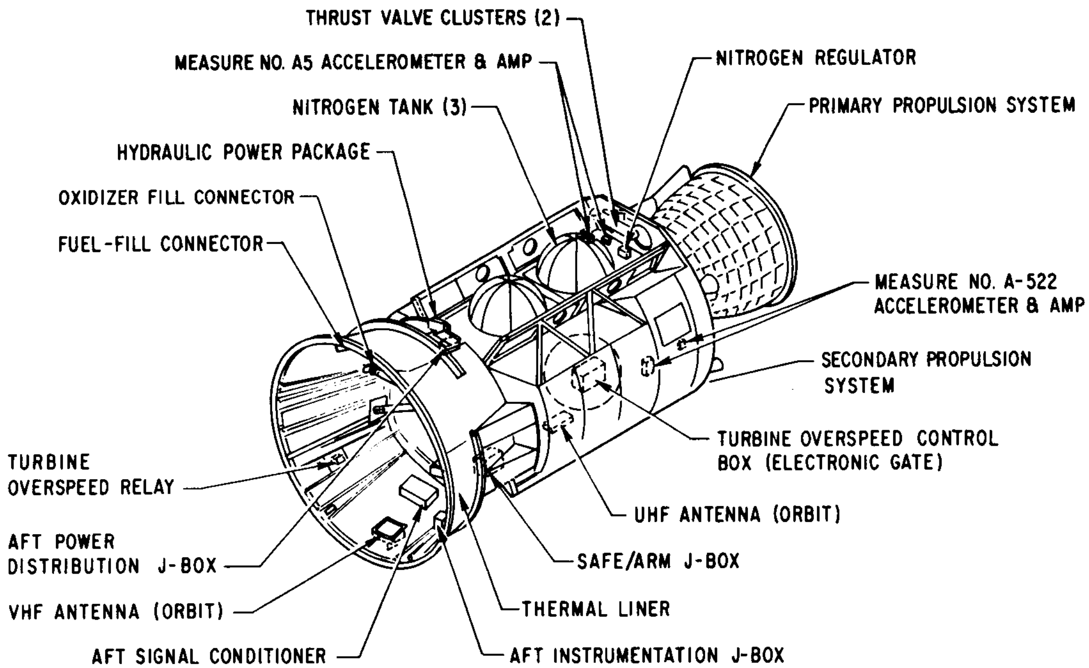

Figure III. C-6 GATV Aft Section

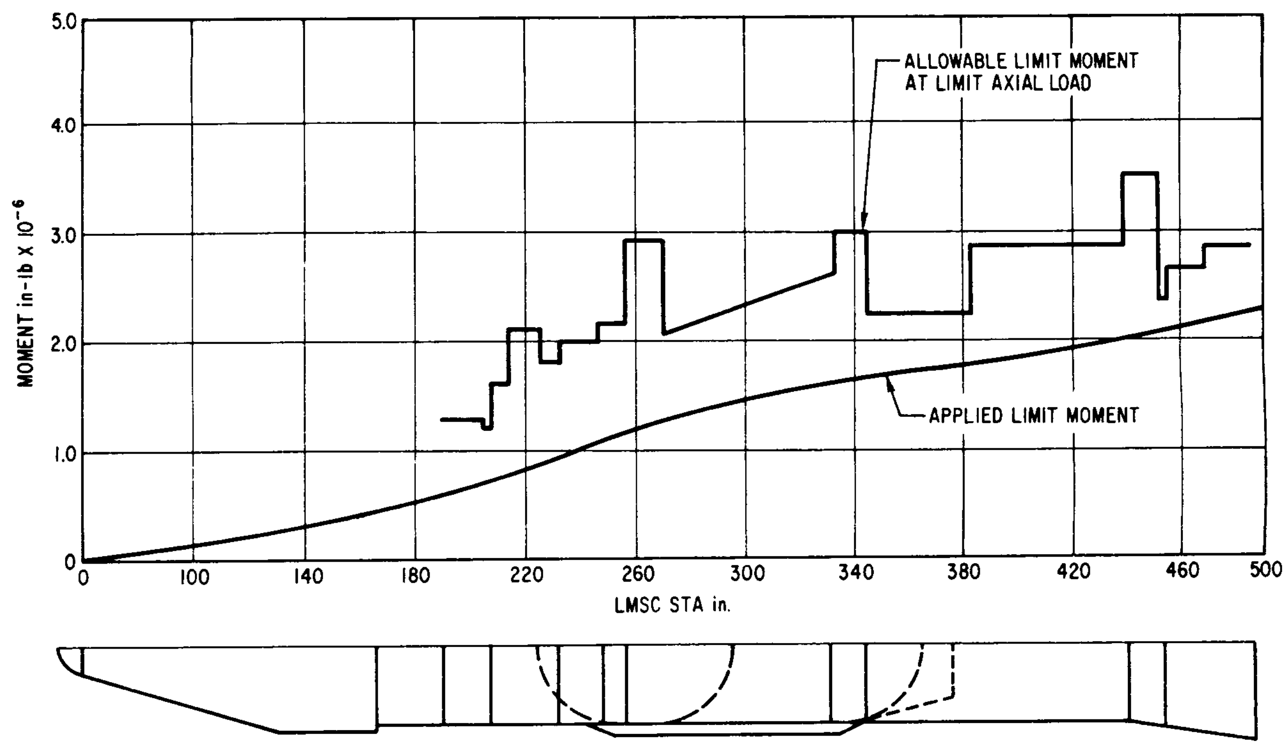

Figure III. C-7 Vehicle Axial Load Comparison, BECO Condition

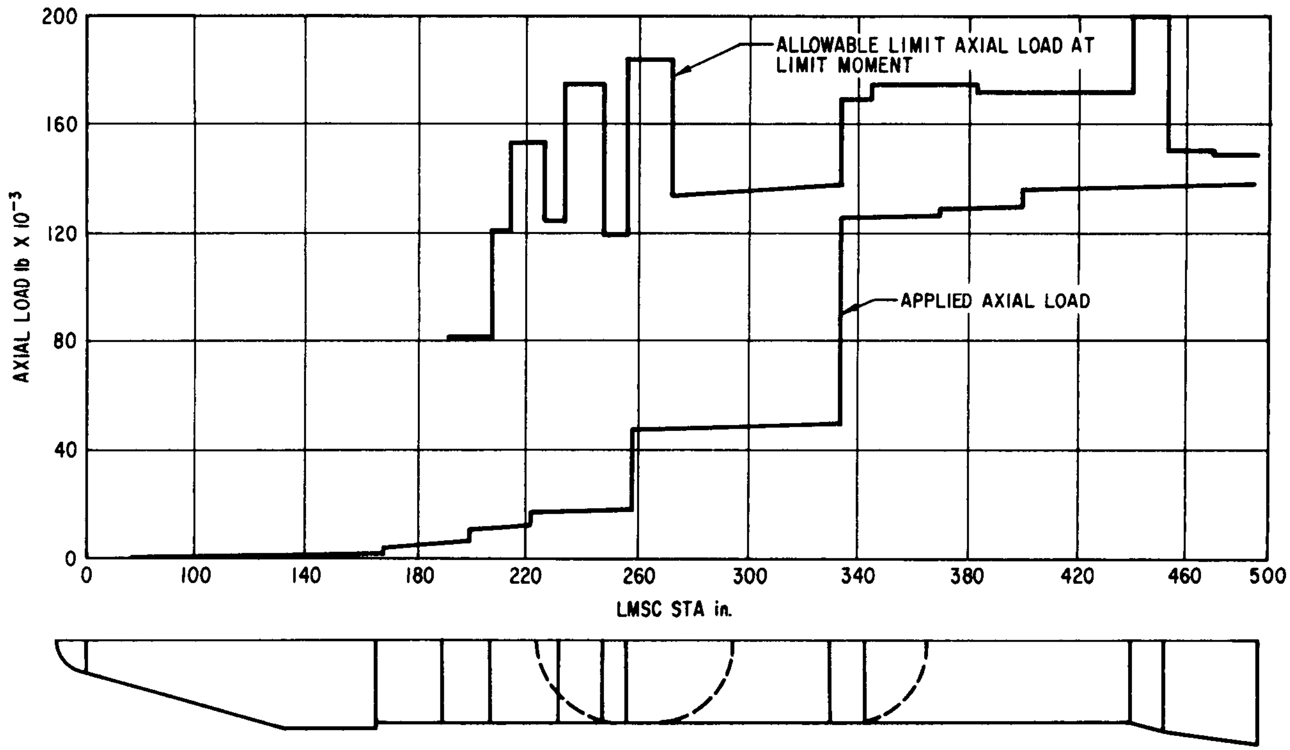

Figure III. C-8. Vehicle Axial Load Comparison, BECO Condition

1) The forward section, consisting of a simulated shroud, target docking adapter, forward auxiliary rack, forward midbody, and simulated tank "Y" ring, was tested to the following loading conditions:

   a) Bending moment and shear loads caused by maximum airload conditions.

   b) Compressive axial loading corresponding to the BECO condition.

All loads were applied to 200 percent of limit without failure.

2) The aft section, consisting of the rack and thrust cone, was tested to the following loading conditions:

   a) Tensile axial loading corresponding to the SECO condition.

   b) Side loads corresponding to the maximum ground handling conditions.

All loads were applied to 200 percent of limit without failure

GATV design air loads were based on launch trajectories defined in LMSC-A604934, Design Trajectory for Gemini Mission, 15 August 1964,[1] and LMSC document SS-570-5351, "Computer Program for MSFC Wind Criteria for AMR," 8 May 1963.[2] Another source is, Loads Analysis Report for Gemini 37205, LMSC-A633170, 1 September 1964.[3]

(b) Dynamic Tests. Structural qualification for dynamic flight environments for the forward and aft sections was conducted in two steps. These were (1) sinusoidal sweep testing with an electrodynamic shaker in the three mutually perpendicular axes, for frequencies noted below, and (2) acoustic testing at Santa Cruz Test Base, using an acoustic environment produced by an 8247 engine firing, to account for high frequency random excitation. The maximum overall acoustic levels reached during these tests were 158.3 db for the forward rack and 147.9 db for the aft rack.

The sinusoidal test levels for the forward rack were as follows:

Axis	Range (cps)	Level (g's)
X	10-140	1.5
	140-250	3.0
Y - Z	10-100	1.0
	100-250	2.0

Sinusoidal test levels for the aft rack were:

Axis	Range (cps)	Level (g's)
X	10-100	2.0
	100-400	3.0
Y - Z	10-100	1.0
	100-400	2.0

The basic structure and equipment support bracketry in the forward area of the Gemini Target Vehicle were qualified for the dynamic flight environment. The term "forward area" includes the Gemini forward auxiliary rack and Gemini-peculiar installations in the Agena D forward rack.

The aft rack basic structure and equipment support hardware also were qualified for the dynamic flight environment. The aft rack included the aft rack basic structure, engine cone, program peculiar support structure, dummy engine and two secondary propulsion units.

3. PROPULSION SYSTEMS

(a) Primary Propulsion System. The primary propulsion system was composed of a helium pressurization system, propellant tanks and a turbopump-fed 16,000-pound thrust engine as shown in Figure III. C-9. The primary purpose of this system was to inject the target vehicle into the prescribed orbit. In orbit it had a minimum capability of four engine restarts to make large plane or phase change maneuvers.

The pressurization system maintained the required pump inlet pressure by supplying helium gas to both propellant tanks. The helium was stored in a high pressure sphere and was separated from the propellant tanks by the helium control valve. The propellant tanks were pre-pressurized in order to provide adequate pump inlet pressures during engine start. Tank pressure was maintained during engine operation by flowing the helium from the helium tank through orifices located in the helium control valve to the propellant tanks. Helium flow was started 1.5 seconds after engine start by an electrical signal that initiated a squib. Helium flow continued until all of the helium was expelled into the propellant tanks. At 318 seconds after opening the helium valve, a second squib was fired isolating the oxidizer tank from the helium supply. This prevented a hoop compression load on the common tank dome by ensuring that the oxidizer tank pressure was lower than the fuel tank pressure.

The main engine used IRFNA (inhibited red fuming nitric acid) and UDMH (unsymmetrical dimethylhydrazine) as propellants. The engine was capable of multiple restarts, using a liquid start system with tanks that were recharged during engine operation.

Starting was accomplished by electrically signalling the fuel and oxidizer gas generator solenoid valves open. The combustion gases generated were utilized to operate a single stage impulse turbine which was geared to a centrifugal fuel and oxidizer pump. The pressure generated by the oxidizer pump opened the main oxidizer valve, permitting the oxidizer to flow first through the thrust chamber cooling passages and then to the injector. A pressure buildup in the oxidizer manifold of the injector tripped a pressure switch that signaled the fuel valve to open. This method of starting ensured that the oxidizer entered the thrust chamber before the fuel.

This starting sequence was a change from the original sequence which did not contain any pressure switches. The original start signal, in addition to opening the fuel and oxidizer gas generator valves, also opened the main fuel valve. This method of starting permitted the fuel to enter the thrust chamber before the oxidizer and was considered the cause of the flight failure of Vehicle 5002. A further discussion of this redesign is in Section III-E where Project Surefire is described. This change was effective on all vehicles subsequent to Vehicle 5002.

Thrust vector control was obtained by two mutually perpendicular hydraulic actuators. The forward end of the engine was mounted in a gimbal ring. A portion of the fuel from the fuel pump was used to drive a hydraulic pump.

Propellant isolation valves were installed between the propellant tanks and the pump inlets. These valves were closed when the engine was shut down and the residual propellants in the engine were vented overboard through these valves.

An electronic overspeed trip device connected to the gearbox was utilized to shut down the engine when the turbine speed was 20 percent above normal. This device was only in operation during orbital operation of the engine. The original configuration had the overspeed trip enabled during the ascent burn in addition to the on-orbit operations. After a slow opening fuel valve on a standard Agena flight caused an overspeed condition, without the loss of the vehicle/mission, a new study of the overspeed trip was made. The study showed that there were conditions that could cause a temporary overspeed from which the engine could recover. Since it could not recover during the ascent burn from an overspeed shutdown and, since crew safety was not a factor, it was decided to disable the overspeed trip during the ascent burn.

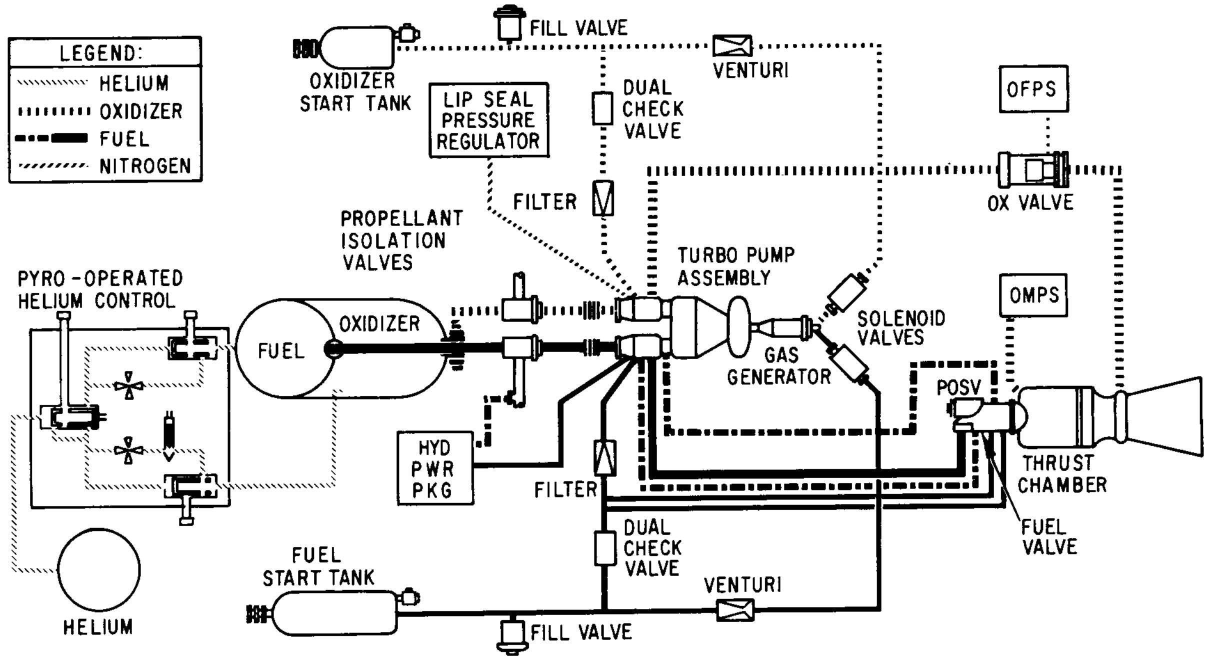

Figure III. C-9. GATV Primary Propulsion System

(b) Secondary Propulsion System. The secondary propulsion system (SPS) consisted of two independent modules mounted on opposite sides of the target vehicle aft section. Each module was capable of producing a total impulse of 40,000 pound/seconds and consisted of a pressurization system, propellant tanks and a large and a small thrust chamber capable, respectively, of producing 200 pounds of thrust and 16 pounds of thrust. The 200-pound thrust chambers were used for producing velocity increments that were below the minimum capability of the main engine. The 16-pound thrust chambers were used to orient the propellants prior to each main engine firing. This system was unique to the target vehicle and was not a part of the standard Agena configuration. A schematic is shown in Figure III. C-10.

The pressurization system maintained the required pressure to the thrust chamber by supplying high pressure nitrogen gas through a pressure regulator that reduced and maintained the pressure at approximately 205 psi. This pressure was supplied to the propellant tanks, each of which contained a metal bellows used to provide orientation of the propellants during zero g coast periods. A solenoid valve at the nitrogen tank outlet was closed at the end of each burn period, and opened 16 seconds prior to each burn period to ensure that adequate tank pressures are available.

The thrust chambers utilized UDMH (unsymmetrical dimethylhydrazene) and MON (mixed oxides of nitrogen) as propellants. The system was capable of multiple restarts. Starting was accomplished by sending an electric signal to either the 16-pound or the 200-pound 3-way solenoid valve. When the solenoid opened, nitrogen gas pressure forced both the oxidizer and fuel poppets open, thereby allowing propellant flow to the thrust chamber; ignition was hypergolic. Shutdown was accomplished by removal of the electrical signal which closed the 3-way solenoid and vented the nitrogen gas causing the poppets to close.

This configuration was utilized for the entire flight program.

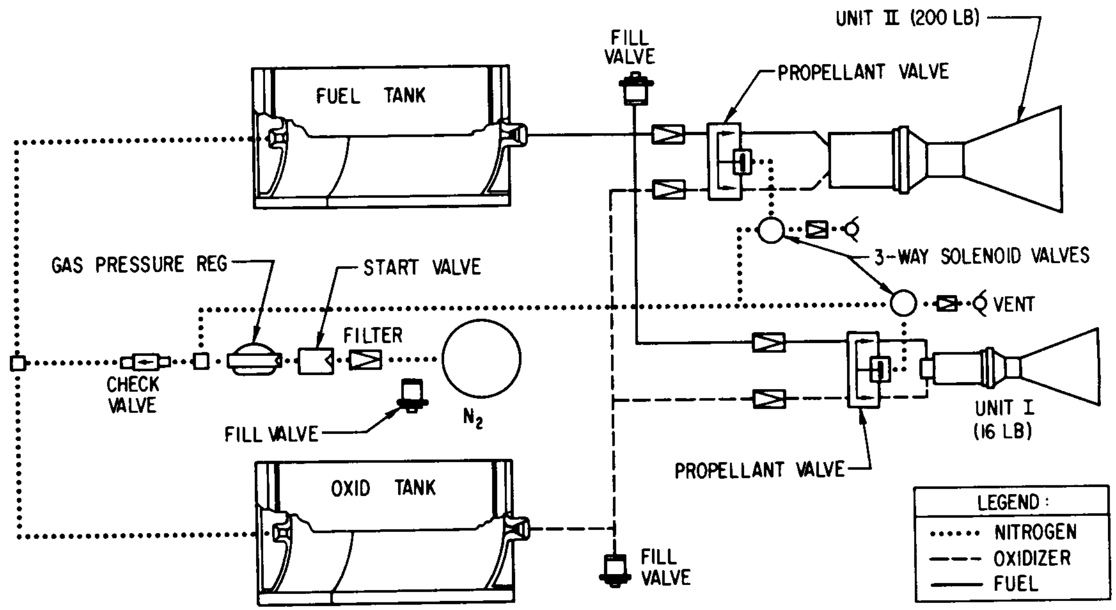

Figure III. C-10. Secondary Propulsion System Schematic

4. ELECTRICAL POWER SUBSYSTEM

a. Subsystem Description

The electrical power subsystem of the GATV supplied the operating voltages and power requirements for the propulsion, pyrotechnic, telemetry, command and communications, guidance and control, and flight-termination system, as well as for the TDA.

Internally-mounted primary batteries began to furnish power shortly before launch when the primary battery circuit was energized and the ground-supplied electrical power was disconnected. Unregulated 28 vdc power was supplied for most of the using components. Electrical power subsystem components also modified the battery-supplied power to provide regulated 28 vdc and 115 v three-phase and single phase a-c power. Power was distributed through a network of harnesses and junction boxes, with power for pyrotechnic devices routed separately. Power for pyrotechnics was furnished by a diode-isolated primary battery. Figure III. C-11 is a block diagram of the electrical subsystem.

b. Component Description

Components of the electrical power system may be grouped into three functional categories: power source components, power conversion components, and power control and distribution components. The power source components included the primary battery units that supplied the initial source of energy to the power conversion components and to the using system components. Also included were the self-destruct system batteries, range safety shutdown system batteries, running lights system batteries, and a GFE battery for acquisition lights in the TDA. Power conversion components, which modified the initial power source to regulate tolerance limits or convert the power into various voltage forms, consisted of a Type XIIA inverter and two Type IX dc-dc converters. Power control and distribution components switched power from one unit to another and distributed power from the source to load points. The power transfer switch, wiring harnesses and junction boxes were the primary components in this category. The majority of the harnesses used on the GATV were supplied with the Standard Agena. Other components of this system that were received with the Agena were as follows:

2	Type 1-C primary batteries
1	Type IX dc/dc converter
1	Current sensor with differential amplifier
1	Ampere hour meter with current sensor
1	Power distribution J-Box
1	Aft instrumentation J-Box
1	Booster discrete J-Box
1	Aft safe/arm J-Box
1	Forward safe/arm J-Box.

Additional standard Agena components added to support the requirements of the GATV are as follows:

4	Type 1-C primary batteries*
1	Type IX dc/dc converter
1	Current sensor with differential amplifier
2	Premature separation switches

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* Reduced to 3 (total of 5) to correct center of gravity offset on Vehicles 5004, 5, 6, and 1.

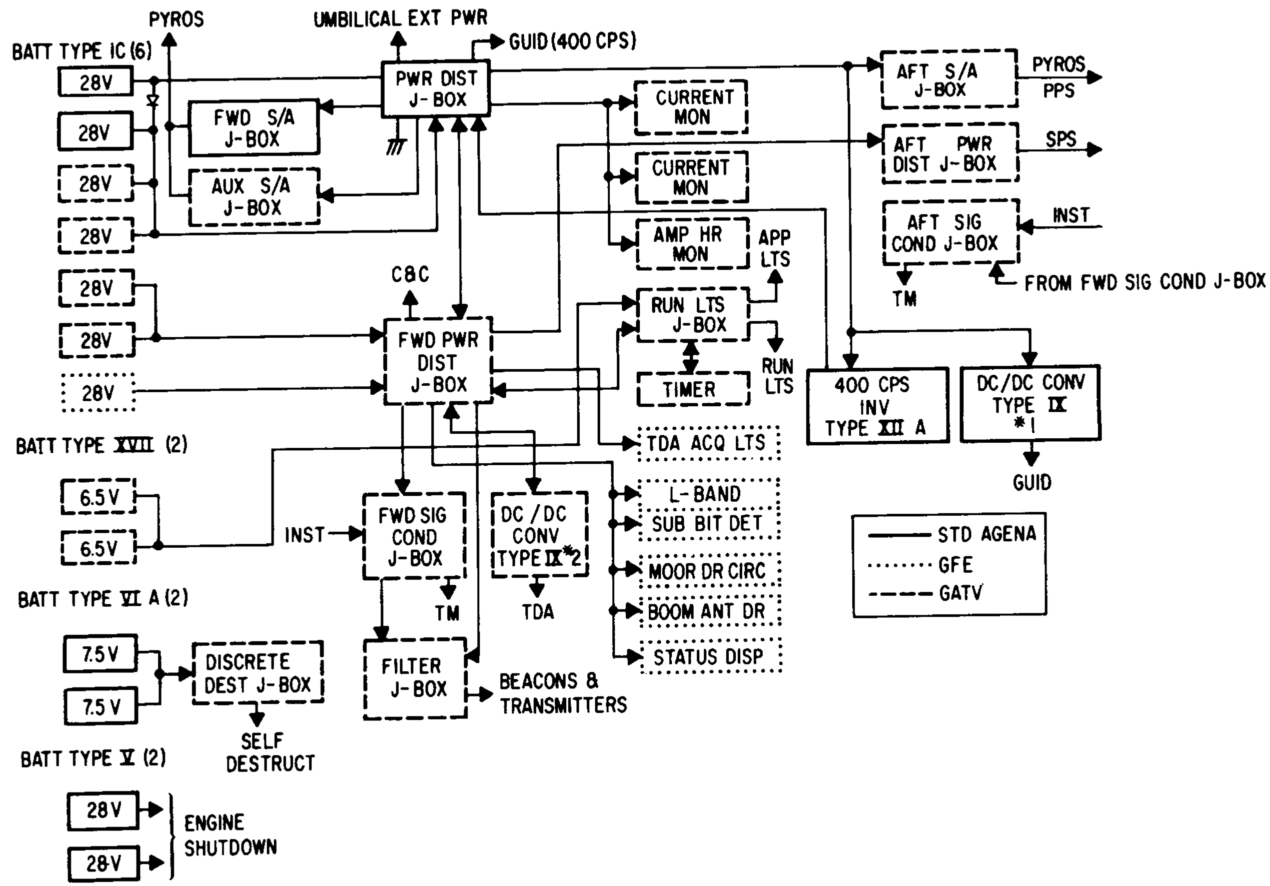


Figure III. C-11. GATV Electrical Subsystem Block Diagram

To complete the GATV electrical subsystem the following equipment was added:

1	Forward power distribution J-Box
1	Aft power distribution J-Box
1	Forward signal conditioner J-Box
1	Aft signal conditioner J-Box
1	Auxiliary forward safe/arm J-Box
1	Running lights J-Box
1	Filter J-Box
1	Time Control J-Box
2	Type XVII primary batteries
2	Type V secondary batteries
2	Type VIA secondary batteries
1	Long delay timer
1	GFE battery

c. System Operation

The electrical power system supplied the necessary power in various forms to the using components systems. Power distribution and on-off control was handled in several ways. Prior to launch, power distribution and control were regulated by ground commands routed through the umbilical connection. During ascent, power was controlled principally by the GATV sequence timer and by the Atlas booster discrete signals. Following the Atlas-GATV separation, the command controller and the flight command logic package were the principal power control devices. A typical power profile is presented in Figure III. C-12.

(1) Acquistion Lights. The acquisition lights were illuminated from a GFE 28 v battery. Control of power application was provided by a relay in the forward power distribution J-box. This entire circuit was shielded, and the shield multigrounded, to prevent EMI problems from the pulsing strobe lights. The system was referenced to the vehicle equipotential plane at the battery connector.

(2) Approach Lights. The approach lights were illuminated upon command through dropping resistors in the running lights J-box. The dropping resistors reduced the 28 v main battery bus voltage to 12 v to eliminate excessive brilliance.

(3) Running Lights. The running lights were powered by a separate 6.5 v power source; two Type XVII primary batteries were used. The running lights could be commanded on or off in conjunction with the approach lights. They could also be reactivated by the long delay timer after a predetermined period.

(4) Major Problems

- Several junction boxes originally contained relays with getters. These indicated a tendency to shatter and contaminate the relays and were replaced throughout the vehicle with getterless relays. No flight failures were attributed to this problem.
- Older Type IX dc/dc converters contained tantalum capacitors which exhibited a tendency to leak electrolyte and short out. A newer model of the converter was substituted containing fewer of these capacitors. No flight failures occurred.
- When the type 1-C main batteries were activated for 5001 (GT-12), they were found to have lower than specification short-circuit voltage. This was due to normal aging (batteries were 10 months old). Tests showed that the batteries still had sufficient capacity to perform the mission and were used with no problem.
- Several problems concerned with the single point ground system were solved by revision of grounding methods (including occasional violations of the single point ground principle).

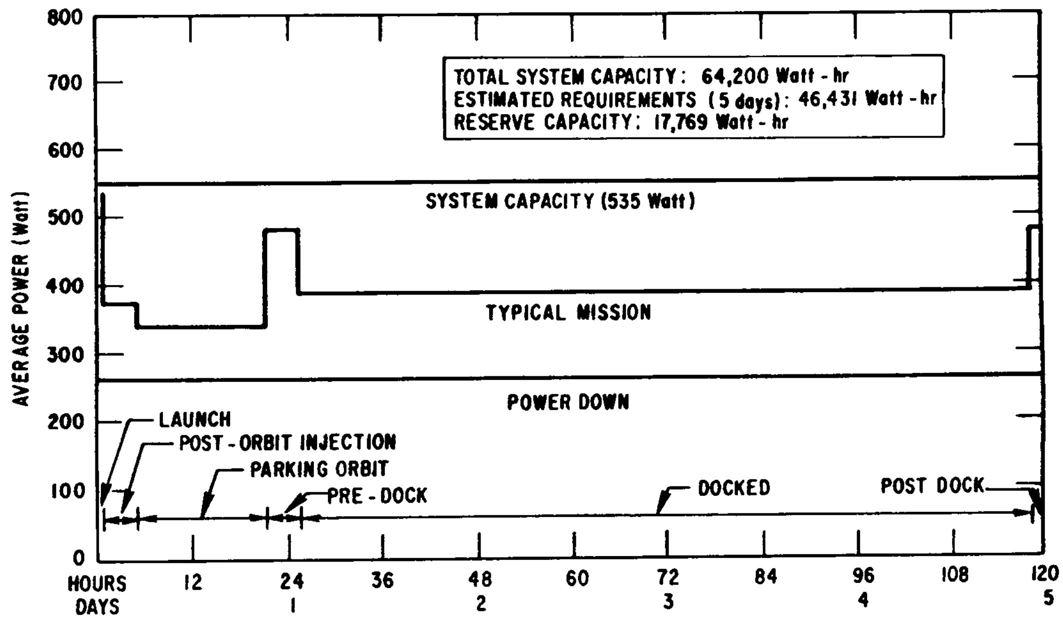


Figure III. C-12. GATV Power Profile

5. GUIDANCE AND CONTROL

a. Subsystem Description

The primary purpose of the Guidance and Control Subsystem was to provide the GATV with adequate guidance and control capabilities to satisfy the overall mission requirements. The primary functions were to:

1) Provide an earth-oriented inertial attitude reference.
2) Sense and control velocity increases along the plus vehicle longitudinal axis.
3) Control the vehicle attitude in a stable manner.
4) Accept commands from both spacecraft and ground stations for vehicle functional conditioning and events.
5) Initiate timing signals for prescribed events (ascent phase only).

This subsystem was composed of two major sections: (1) the guidance system and, (2) the flight control system. These are shown in Figure III. C-13.

The major part of this subsystem was unchanged from the Standard Agena components. However, there were numerous changes made to the Standard Agena equipment during the course of the Gemini program. Several of these changes were of major importance. New and modified major components are indicated in Figure III. C-14.

(1) Guidance System

The requirement for an earth-oriented inertial attitude reference was fulfilled by the inertial reference package (IRP) and the horizon sensor (H/S). The IRP provided an inertial attitude reference and the horizon sensor provided the earth-orientation. The velocity meter (V/M) satisfied the requirement for sensing and controlling velocity increases along the vehicle longitudinal axis (plus polarity only). The ascent sequence timer provided discreet signals for timed functional events, during the ascent phase. The flight command logic package (FCLP) was the interface with the Command Communication Subsystem which enabled various functional configuration modes to be selected and commanded events to occur. These modes and events could be selected by either the spacecraft or ground stations. Vehicle attitude could be changed by applying the appropriate signals to the various components. Vehicle velocity could be changed by firing the appropriate propulsion system and utilizing the velocity meter to terminate thrust after the desired delta velocity had been achieved. Both of these events could be initiated by real-time commands (RTC's) or by stored program commands (SPC's). Either of these provided command inputs to the flight command logic package. Although it represented an interface between the command system and the remaining systems in the vehicle, the flight command logic package was considered a part of the guidance system, since many of its outputs were routed to this system. The guidance J-box and the flight control J-box provided the interconnects between the major components of the system. In addition, portions of the component circuitry were located in these boxes to facilitate program-peculiar modifications to these components.

(2) Flight Control System

The principal function of the flight control system was to provide stabilization and control of the vehicle attitude in response to signals from the guidance system. During primary propulsion system

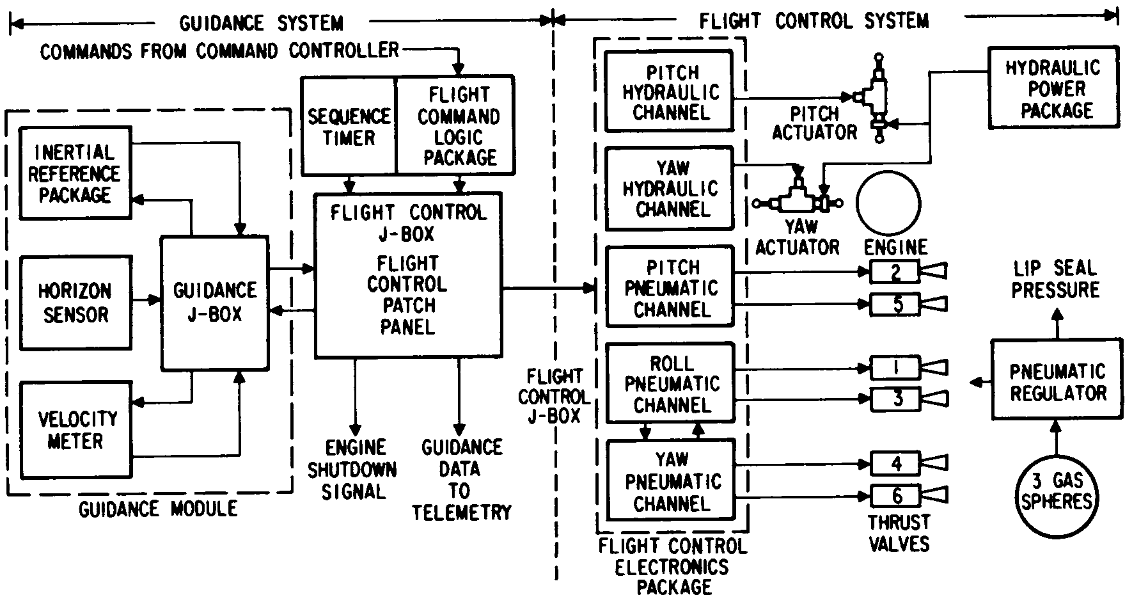


Figure III. C-13. Guidance and Control Subsystem Simplified Functional Block Diagram

Guidance System

Inertial Reference Package (Mod III)	Standard Agena
Horizon Sensor (Mod IIC)	
Mixer Box	Standard Agena
Right Sensor Head	Standard Agena
Left Sensor Head	Standard Agena
Velocity Meter	
Accelerometer Unit	Standard Agena
Accelerometer Electronics	Standard Agena
Counter (Mod IIA)	Standard Agena
Guidance J-Box	Modified
Ascent Sequence Timer	Standard Agena
Flight Command Logic Package	New

Flight Control System

Flight Control J-Box	Modified
Flight Control Electronics Package	Modified
Pneumatic System	
Pneumatic Storage Spheres	Standard Agena
Pneumatic Pressure Regulator	Standard Agena
Pneumatic Thrust Controllers	Standard Agena
Hydraulic System	
Pitch and Yaw Hydraulic Actuators	Standard Agena
Hydraulic Power Package	Standard Agena

Figure III. C-14. Guidance and Control Subsystem Component Modifications – Standard Agena vs GATV

(PPS) engine operation, the hydraulic control system controlled direction of the thrust vector about the pitch and yaw axes by the use of hydraulic actuators while the roll attitude was maintained by the pneumatic control system. During the coast phase, the pneumatic control system controlled vehicle pitch, yaw, and roll motion by means of thrust controllers. Both the pneumatic and the hydraulic control systems were controlled by the IRP through the flight control electronics package (FCEP).

When the PPS engine is not operating, the pneumatic system controlled vehicle attitude by applying corrective force about three axes by means of pneumatic thrust valves. Signals from the IRP were fed to the flight control electronics package, and trigger circuits applied pulses to the proper thrust valve to produce the desired corrective force.

The hydraulic control system guided the vehicle by providing thrust-vector control during each period of PPS engine operation. The pitch and yaw attitude of the vehicle were controlled by angular positioning of the thrust vector so that control movements were imparted to the vehicle. This was accomplished by two hydraulic actuators that deflected the engine about each gimbal axis.

b. Component History

As has been pointed out, standard Agena Guidance and Control components were adapted for use in the GATV with minimum changes. Components that required extensive modifications and could, therefore, be labeled Gemini-peculiar items, were the guidance J-Box, flight control J-Box (Patch Panel) and the flight control electronics package. The flight command logic package was a new component and unique to the GATV. This component was required to interface with the command and control subsystem. The modified and new components were tested as required both in the breadboard level and production configuration in Electromagnetic Interference (EMI) tests; elevated stress and extended life tests were also added to demonstrate suitability for the Gemini application. Qualification tests were performed on all the Gemini peculiar components. In addition, the Standard Agena component, Mod II-A counter was requalified by the Standard Agena Office during the program because a new design was incorporated in the velocity meter.

Qualification test reports for the unmodified Standard Agena components were reviewed early in the program. An evaluation was made to ascertain whether limited requalification tests or "qualified by similarity" gave adequate assurance for changes that had been made to the equipment. Random sampling test results were also monitored where possible for additional confidence.

During the Gemini program, the reliability of LMSC manufactured electronic components was increased by piece part selection. This equipment upgrading was not unique to the GATV, but was phased in on the Standard Agena Vehicle.

It was established that guidance and control system performance trend analyses on the vehicle level testing were not being accomplished. Furthermore, it was determined that there were deficiencies in the test procedures, both at Sunnyvale and ETR. This situation was corrected by:

- Standardizing methods of testing.
- Recording and tabulating necessary data and calculations.
- Increasing scope of specific tests.

Performance trend analyses were performed and data correlations were made. This proved to be a successful operation and a valuable indicator for the guidance and control equipment.

c. Major Problems

During the program, several problems or piece-part deficiencies required intense developmental testing activities but of limited scope. Secondary GATV 5004 failures which occurred after the primary Atlas failure resulted in a test program evaluating effects of umbilical arcing on the inertial reference package, ascent sequence timer, static inverter and associated power J-boxes. Results gave additional confidence that the components were adequate for the Gemini program. Prior to GATV 5001 flight, several transistors in the gyro heat control circuitry were suspected of contamination. The entire guidance module was changed out and replaced with a spare unit, which was used for flight. The suspect piece parts were found to be faulty and were replaced.

The horizon sensor, which was used without modification from the Standard Agena configuration, exhibited erratic outputs during the flight of GATV 5006. An extensive test program to analyze the problem resulted in the implementation of special test monitoring on vehicle 5001. This problem is still under investigation by LMSC to determine its effect on other programs.

The original practice of modifying guidance junction boxes with limited disassembly was changed to a more complete disassembling and rebuilding practice. This improved the confidence factor in modifying this high density component. During the program, a temperature cycling test was initiated on these units to discover workmanship type errors, particularly bad solder joints. Variations of the same cycling test were applied to other guidance and control components for the same reasons and served to increase confidence in their successful operation.

During the course of the program, the velocity meter counters were subjected to rework cycles to eliminate transistors suspected of contamination, leaky tantalum capacitors, and transistors afflicted with a dew point moisture problem. The latter problem also affected the guidance junction box, the horizon sensor mixer, and the flight control electronics package, and was satisfactorily resolved in each case.

Prior to the flight of GATV 5003, a major modification was made on the hydraulic channels in the flight control electronics package. This was required because of the lower bending frequencies present when the Agena is docked with the Gemini spacecraft. The dynamic characteristics of the Standard Agena hydraulic channels would not permit stable operation in this mode. To obtain a stable system, the passive lead-lag shaping network was replaced by an active lead-lag shaping network with a longer time constant. This redesign is discussed in detail in the special studies section (III-E) of this report.

Progressive current leakage of capacitors caused a problem in the capacitor block in the flight control electronics package. Failure modes were either a loss of guidance power or a reduction of stability because of attenuation of the derived rate signals. Special test data correlation was initiated on GATV 5003 where all factory and ETR vehicle level test results of this subsystem were compared. This trend analysis of data proved invaluable in giving assurance that a critical failure mode had not occurred and that consecutive data examination would reveal any deterioration that might be developing.

6. COMMAND AND COMMUNICATION SUBSYSTEM

The Command and Communications Subsystem (C & C) performed three functions: commanding, telemetry, and tracking. These three functions are described separately in this section. Figure III. C-15 is a block diagram of the overall system.

a. Control Commanding

(1) System Description

The control commanding system received, decoded and processed command signals for operationally conditioning the Agena. These commands were generated by the ground tracking stations and/or by the Gemini spacecraft. UHF commands from the ground stations were sent either as real time commands which were executed immediately or sent as stored program commands and executed at some later, preselected time. The spacecraft sent r-f commands by means of its L-Band command subsystem or, when docked, by means of the TDA hardline command capability. The spacecraft commands were all sent as real time commands and executed immediately.

Figure III. C-16 shows the format of the real-time and stored-program commands. The vehicle address identified the spacecraft to which the command was directed. The system address identified the message as a real-time or a stored-program command. The other words in the command message were self-explanatory. The command system continuously monitored the sub-bit code to validate the commands that had been sent.

The programmer checked the command message for proper command address and length, decoded the command function and, for real-time commands, sent a command to the controller which conditioned the execution signal and forwarded it to the required vehicle area. Upon accepting either a real-time or stored-program command, the programmer sent a message accepting pulse to the telemetry subsystem which transmitted it back to the ground station.

For stored-program commands, the programmer located the command in its memory according to the address designated in the command message. The vehicle clock in the programmer continuously counted. The clock's count and the stored-command times were continuously circulated through a comparator. When the clock and command message times were coincident, the command function was decoded and a proper execution signal was sent to the command controller.

The command controller also contained an engine sequencer which, upon initiation by means of a command, properly sequenced the events for starting and stopping the primary propulsion system. The controller also contained the emergency reset timer. If, during a preselected mission time, the emergency timer were not reset because of loss of command contact, the timer would time-out, turning on the telemetry links, tracking beacons, the tape recorder, L-Band System and enabling the UHF commanding capability (if previously disabled). This assisted the ground stations in reacquiring the vehicle.

(2) Major Problems

During the early months of the program, there were a large number of failures of the integrated circuits used in the programmer and controller. These were traced to failure of the gold-to-aluminum bond between the substrate and the leads of the circuit chips. Replacement with integrated circuits manufactured with gold-to-gold bonded junctions eliminated the problem.

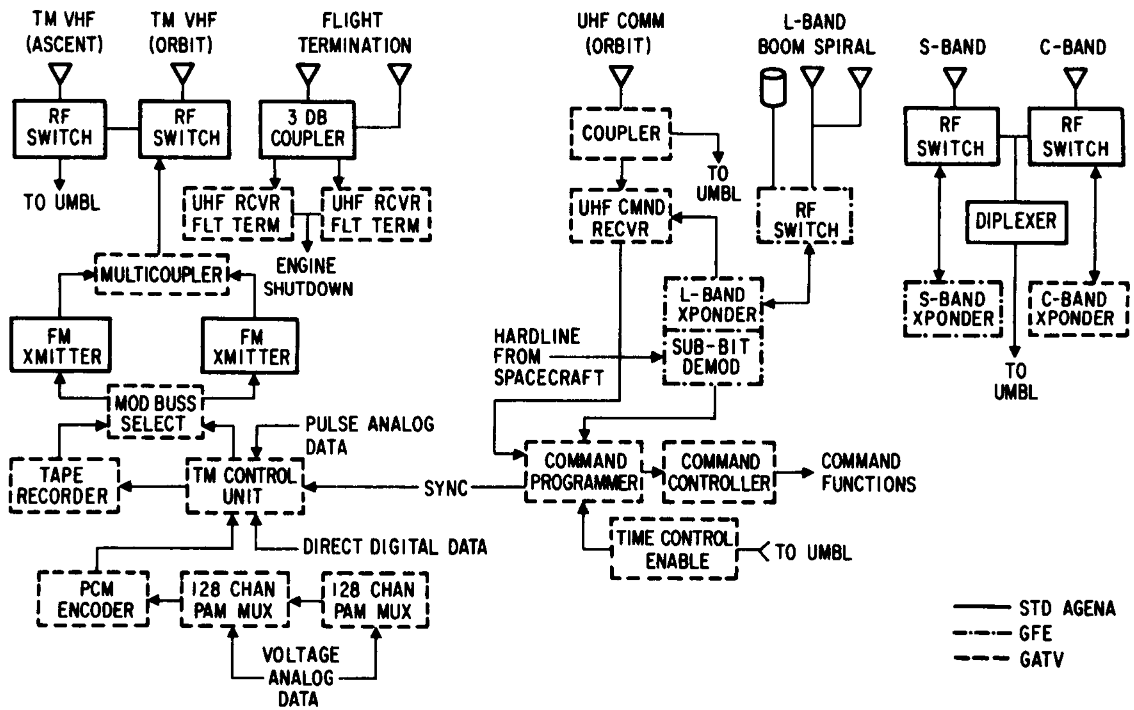


Figure III. C-15. GATV Communications and Command Subsystem

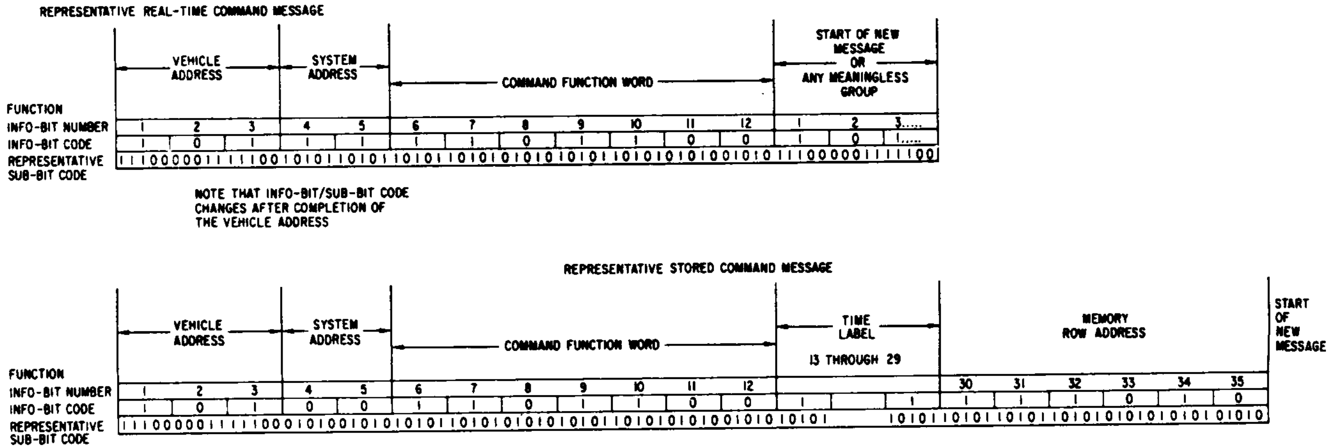


Figure III. C-16. Representative Command Messages

Undesirable time-outs of the emergency reset timer occurred during pre-launch checkouts of Vehicle 5004 at Cape Kennedy. Analysis showed the problem to be a function of the design of the timer circuit which could set itself to the time-out position when power was removed from the vehicle. Rather than redesign the circuitry, operational procedures were changed to automatically turn off all equipments turned on by ERT timeout and automatically reset the timer each time power was reapplied to the vehicle.

During pre-launch checkout of Vehicle 5006, spurious commands were processed by the command system. These commands were generated only when the vehicle clock was started by umbilical disconnect. Circuit analysis, which was verified by test, showed that the basic programmer design allowed the generation of spurious commands at clock start, but only if three conditions were met:

1) The memory must contain a command load.
2) The vehicle clock must be stopped by AGE at a specific time during a memory row readout.
3) The vehicle time accumulator must be reset to zero by AGE before restarting the vehicle clock.

The problem could not occur during orbital operations because the time accumulator was reset by stored program command and the clock was never stopped. Again, the problem during ground test was circumvented operationally by loading the time accumulator with a number other than zero prior to any clock restarts. For the special case of liftoff, the clock was started several minutes prior thereto rather than via umbilical pull. These procedural changes were utilized successfully for the launches of 5006 and 5001.

b. Telemetry System

(1) System Description

The telemetry system was designed to sense, encode, record, and transmit vehicle information to telemetry ground systems. This PCM-FM telemetry system (designed in accordance with IRIG 106-60), consisted of the following:

- 128-channel PAM Main Multiplexer
- 128-channel PAM Submultiplexer
- PAM-PCM Encoder
- Telemeter Control Unit
- Tape Recorder
- VHF Telemetry Transmitters (2)
- Telemeter RF System, including antennas and multicoupler.

A simplified block diagram of the system is presented in Figure III. C-17.

The system accommodated three broad categories of data as inputs to the telemeter; analog, direct digital, and pulse analog.

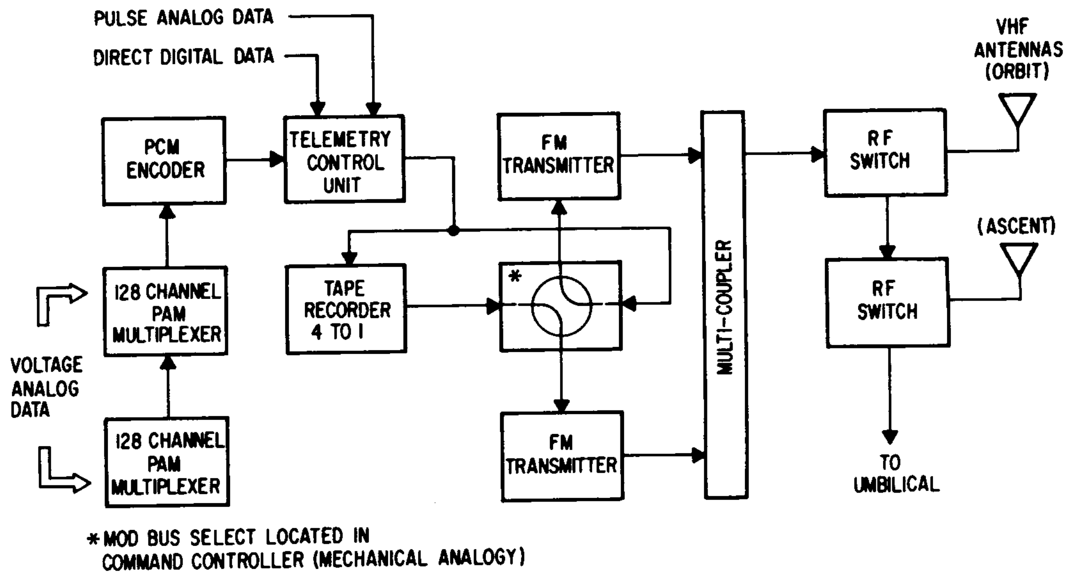


Figure III. C-17 Telemetry System Simplified Block Diagram

(2) System Operation

Analog data requiring low frequency response was sampled by the 128-channel PAM submultiplexer. The output of the submultiplexer was in turn sampled (in a synchronous manner) by the 128-channel PAM main multiplexer. Analog data requiring a somewhat higher response was sampled directly by the main multiplexer. The pulse amplitude samples of the analog data were then encoded to a seven-bit word. An eighth-bit that is the complement of the least significant bit was added to provide word synchronization and separation. The output of the encoder was then fed into the telemeter control unit where it was combined with the content of the direct digital input registers and pulse analog counters in a sequential manner to provide an orderly format that became a continuous serial bit stream as an output. The telemeter control unit contained the circuitry for maintaining synchronism with the programmer so that the direct digital inputs from the programmer could be transmitted.

The telemeter control unit output frequency modulated a VHF transmitter (240-250 mc region) to form the PCM/FM transmitted signal at a rate of 16, 384 bits per second.

The output of the telemeter control unit could, upon command, be recorded on the PCM Tape Recorder, Type IX, for playback later. The tape recorder had a capacity for 20 minutes of stored data and played it back at four times the read-in rate, which allowed for complete readout in five minutes at a rate of 65, 536 bits per second. The output of the tape recorder frequency modulated a second VHF transmitter. It was possible to reverse the two VHF transmitters inputs by use of a pair of commands to operate the Mod Bus Select. The VHF Multicoupler, Type XI, allowed the two transmitter outputs to feed a common antenna system. Separate antennas were used for the ascent and orbit phases.

c. Tracking System

(1) System Description

The tracking system for the GATV included the following on-board equipment:

- S-band Transponder
- C-band Transponder
- L-band Transponder
- Antennas

Tracking of the GATV by ground stations was by the interrogation of independently operated S-band and C-band transponders. The telemetry signals also could be used as an acquisition aid. Tracking of the GATV by the spacecraft was by interrogation of the L-band transponder in the TDA.

(a) S-Band Transponder. In response to ground-generated signals, the S-band transponder accepted a pulse code consisting of two one-microsecond pulses spaced 15. 25 microseconds apart, at an interrogation rate of 150 to 1600 pulse pairs per second. Signal strength at the receiving antenna terminals had to be minus 60 dbm or greater.

Nominally, one microsecond after receiving the second pulse of a pair, the transponder transmitted a single one-microsecond pulse on a slightly higher frequency. This was with a nominal peak power output of 630 watts at the antenna terminals, which included line loss to the antenna.

(b) C-Band Transponder. In response to ground-generated signals, the C-band transponder accepted a pulse code consisting of two one-microsecond pulses spaced 3. 5 microseconds apart at an interrogation rate of 100 to 1600 pulse pairs per second. Signal strength at the receiving antenna terminals had to be minus 60 dbm or greater.

Nominally, 2.75 microseconds after receiving the second pulse of a pair, the transponder transmitted a single one-microsecond pulse on a slightly higher frequency. This was with a nominal peak power output of 220 watts at the antenna terminals, including line loss to the antenna.

(c) L-Band Transponder. The L-band transponder accepted from the spacecraft radar a single one-microsecond pulse at an interrogation rate of 256 pulses per second.

Binary-coded commands were transmitted via pulse position modulation of the interrogating pulse train. This transponder was included in the TDA.

(d) Antennas. Transmitting and receiving antenna systems were provided for tracking functions during the ascent, orbit, and docking phases of flight operations. During ascent and orbit, single linearly polarized C-band and S-band antennas were used; and during the docking phase, an L-band antenna system consisting of three antennas was used.

(2) Major Problems

Several EMI problems resulted in momentary synchronization losses of the telemetry system. Grounding revisions reduced these losses to a minimum.

Two problems affected the tape recorder. The first was due to a breakdown of adhesive used to apply the metallic tape end sensing strip to the tape. This was cured by splicing in the metallic strip rather than laminating it to the tape. The second was failure of the bearings in the idler assembly. New bearings were installed after inspection and test to assure proper lubrication and an audible noise test was used to monitor bearing operation.

Individual transducers and sensors failed during the program. Only two continuous problems were monitored. One was with stick-on temperature sensors which showed their characteristic high mechanical failure rate, and the other was a problem with welding temperature sensors to the engine in high temperature areas. No completely satisfactory solution was found to either problem.

(3) Flight Termination System

Range-safety requirements were satisfied by two independent systems: the premature separation self-destruct system and the range safety engine-shutdown system. The premature separation self-destruct system provided two capabilities:

a) During the period between liftoff and Atlas/GATV separation, a destruct signal from range-safety transmitters will ignite destruct charges in the Atlas and in the GATV booster adapter that will destroy both vehicles.

b) During the period between liftoff and Atlas sustainer engine cutoff, the GATV will be automatically destroyed by ignition of a charge in the booster adapter if premature separation from the Atlas booster occurs.

The range safety engine-shutdown system provides two capabilities:

a) During the period between liftoff and GATV engine ignition, a thrust-termination signal from range-safety transmitters will prevent GATV primary propulsion system start.

b) During the period of GATV-powered flight prior to injection into the parking orbit, a thrust termination signal from the range-safety transmitters will cause PPS thrust cutoff.

There were no significant problems associated with this system. A thrust termination signal was sent following the failure of Atlas 5303 (Gemini mission IX). This signal inhibited Agena PPS first burn representing satisfactory operation of the system.

7. TARGET DOCKING ADAPTER AND DISPLAY[4,5]

a. General

The Target Docking Adapter (TDA), manufactured by the McDonnell Aircraft Corporation, was attached to the forward end of the Gemini Agena Target (GAT) to permit mating with the Gemini spacecraft for docked operation of the combined vehicles. The TDA consisted of two principal structural parts: the adapter which was mounted rigidly on the GAT as an integral assembly and contained the equipment necessary for rendezvous and docked operation as well as for spacecraft - commanded undocked operation, and the docking cone which enabled the mating and demating of the two vehicles and their joint operation.

b. Mechanical Operation

In the unrigidized (extended) position, the docking cone:

- Received the rendezvous and recovery (R & R) section of the Gemini spacecraft and absorbed the associated contact shock loads,
- Locked the spacecraft's R & R section into the cone with spring-loaded latches, and
- Automatically rigidized (retracted) against structural pads, pulling the spacecraft in to form a rigid structural connection between the two vehicles and a hardline electrical connection through the mating of a nine-pin umbilical connection.

In the rigidized position, the docking cone could be commanded to unrigidize and to release (unlatch) the spacecraft.

During the ascent phase, the adapter supported a nose shroud which served as a protective shield for the TDA against max q and heating effects. This shroud was separated from the GAT shortly after PPS thrust initiation during the orbital insertion phase.

c. Electrical Control and Spacecraft - GAT Interface

For the purpose of effecting a rendezvous between the Gemini Spacecraft (S/C) and the GAT by radar tracking and providing a capability of sending S/C commands to the GAT in docked or undocked configurations, the TDA was equipped with an L-Band transponder and associated antennas, and with C- and S-Band transponder beacons for ground tracking.

In the undocked configuration, the S/C L-Band radar operated with the TDA's transponder in the tracking mode from approximately 200 n. mi. out, and in the command mode to prepare the GAT for the final rendezvous phase and docking. To complement the radar, two high intensity flashing acquisition lights were mounted on the TDA, which became visible on the night side at an approximate range of 50 n. mi. Visual acquisition under reflected sunlight conditions occurred at approximate ranges of 70 to 80 n. mi. For the docking phase, 2 flood (approach) lights were installed to illuminate the area of the docking cone and permit visual inspection of the cone prior to docking. In addition, colored running lights were provided on the GAT to permit determination of its orientation and attitude on the night side.

In the docked configuration, S/C commands were sent through the umbilical to the L-Band transponder's sub-bit detector and on to the GAT's programmer for execution. Since the sub-bit detector was powered directly off the GAT's regulated bus, the L-Band transponder need not be on during docked flight. Acceptance of S/C commands by the GAT was indicated by the Message Acceptance Pulse (MAP) illuminating the MAP light in the S/C.

Additional hardline circuits crossing the S/C - GAT interface were the Engine Arm/Stop switch circuit, which enabled or disabled the PPS and SPS control circuits in the docked configuration, and 3 hardline TDA cone command circuits controlled by a two-way Rigidize/Rigidize Stop and a one-way Undock momentary toggle switch. The rigidize and undock circuits ran through special single point umbilicals which mated when the docking latches were seated and permitted control in the event of a failure of the automatic rigidizing sequence of the GAT command and control system.

Ground commands to the GAT were received by the Agena's UHF command receiver and transmitted to the programmer for execution. These commands were locked out by an L-Band command presence signal which in the undocked mode was generated by Agena phase lock, with the S/C radar power and encoder power switches on. In the docked mode, this same ground command lockout was achieved when the S/C encoder was on. Ground transmittal of stored program commands (SPC's) was inhibited in the same way, but not the execution of the commands unless the SPC disable command was sent.[6]

d. Agena Status Display[7]

The Agena status display (Figure III. C-18) was located in the upper forward section of the adapter. A cutout was provided in the docking cone to make it visible to the S/C crew in the docked configuration. It reflected the status of the propulsion, flight control and electrical subsystems of the Agena in addition to the position of the docking cone. Figure III. C-19 shows the logic driving the display in block diagram form.

The configuration and location of the display were largely determined by the limitations of space in the spacecraft and TDA, and the limited capacity of the electrical interfaces between the TDA on the one hand and the spacecraft and Agena on the other.

In the light of the above mentioned restrictions and taking into account the backup role of associated telemetry measurements, the logic behind the legend indicators was acceptable on the theory that no single failure which was not displayed could give rise to an unrecoverable condition even if associated with PPS operation. The unavailability of a status display for the PPS, particularly covering conditions of the OMPS, OFPS and PIV's, made necessary the imposition of an operational restriction for docked PPS burns during initial flights to be performed within range of a ground station. The PPS control circuit configuration for the GAT did require verification of a proper shutdown on the previous burn to ensure a safe start.

Ground station monitoring of all docked maneuvers was also desirable due to the limited accuracy of the GAT propellant-remaining displays for the PPS, SPS, and ACS. The first two (PPS, SPS) required real time or delayed tape playback monitoring of the maneuvers for the reasons mentioned in this and preceding paragraphs.

Operationally, the location and size of the display imposed restrictions on its utility, and on mission flexibility. Inability to read the panel from a distance greater than 50 feet, and thus ascertain safe status of the GAT, imposed the requirement that the final stages of rendezvous from 1500 feet out be conducted within range of ground stations. In docked configuration, legibility of the display was dependent on the sunlight incidence angle and the particular flight condition. Furthermore, the need for monitoring S/C displays during PPS burns made simultaneous monitoring of the Agena display difficult.

Optimum reliability of the Agena Status Display was limited by the non-redundancy of transducers and logic circuits, and by the lack of fail-safe design. However, within its functional and operational limitations, it performed reasonably well and experienced no failures during any of the four missions flown.

In summary, if operational flexibility is to be attained in the performance of rendezvous, docking and joint maneuvers between a manned spacecraft and an unmanned space vehicle, it is imperative that subsystem status of the space vehicle be displayed accurately in the spacecraft in both the docked and undocked configuration, preferably from maximum ranges required for rendezvous radar lock-on. Furthermore, the design should provide maximum redundancy and fail-safety to reduce dependence on ground stations to a minimum.

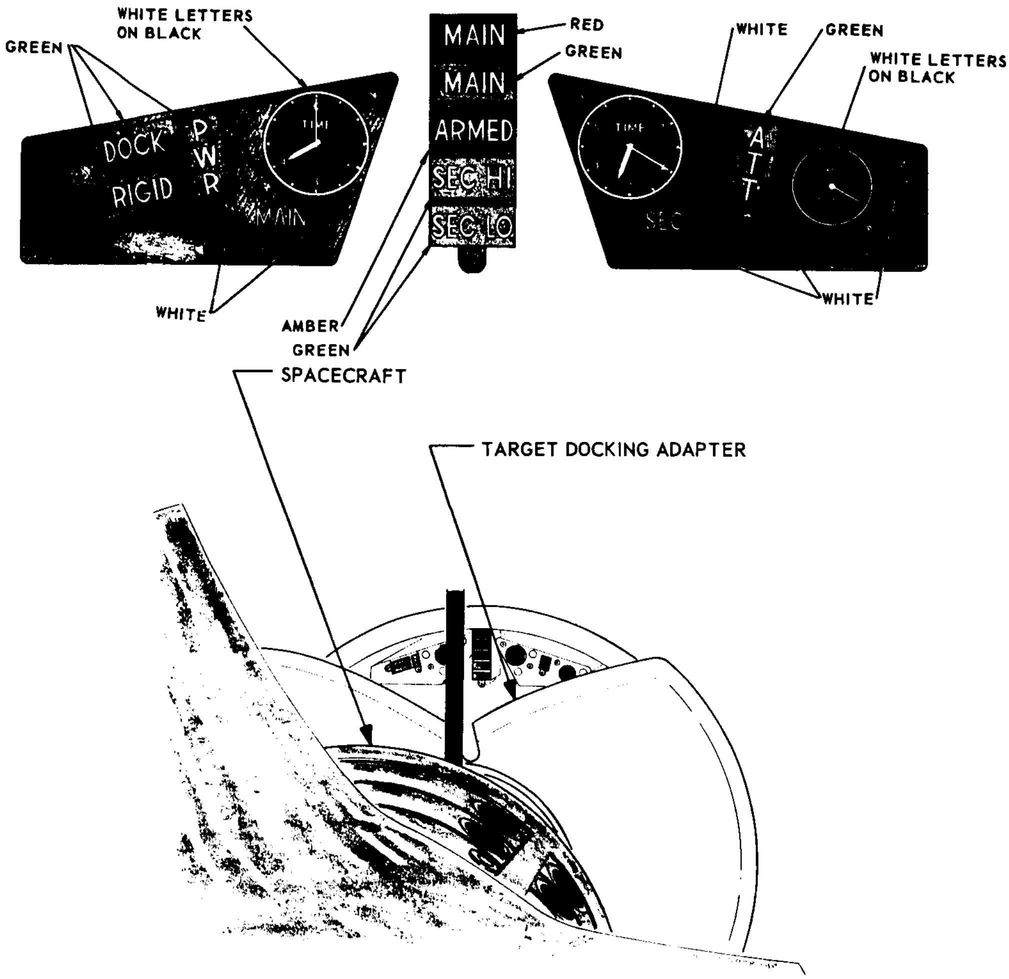
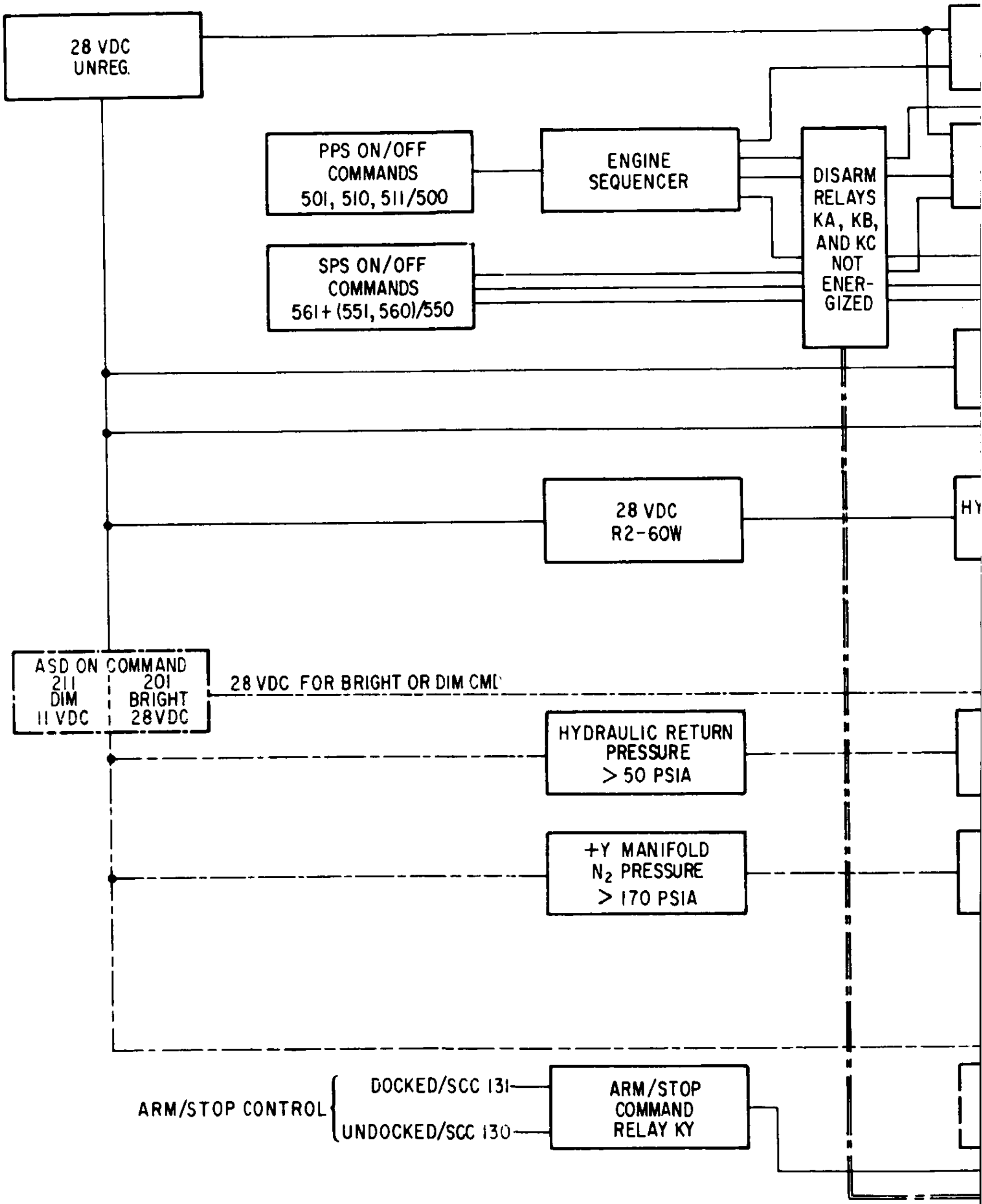
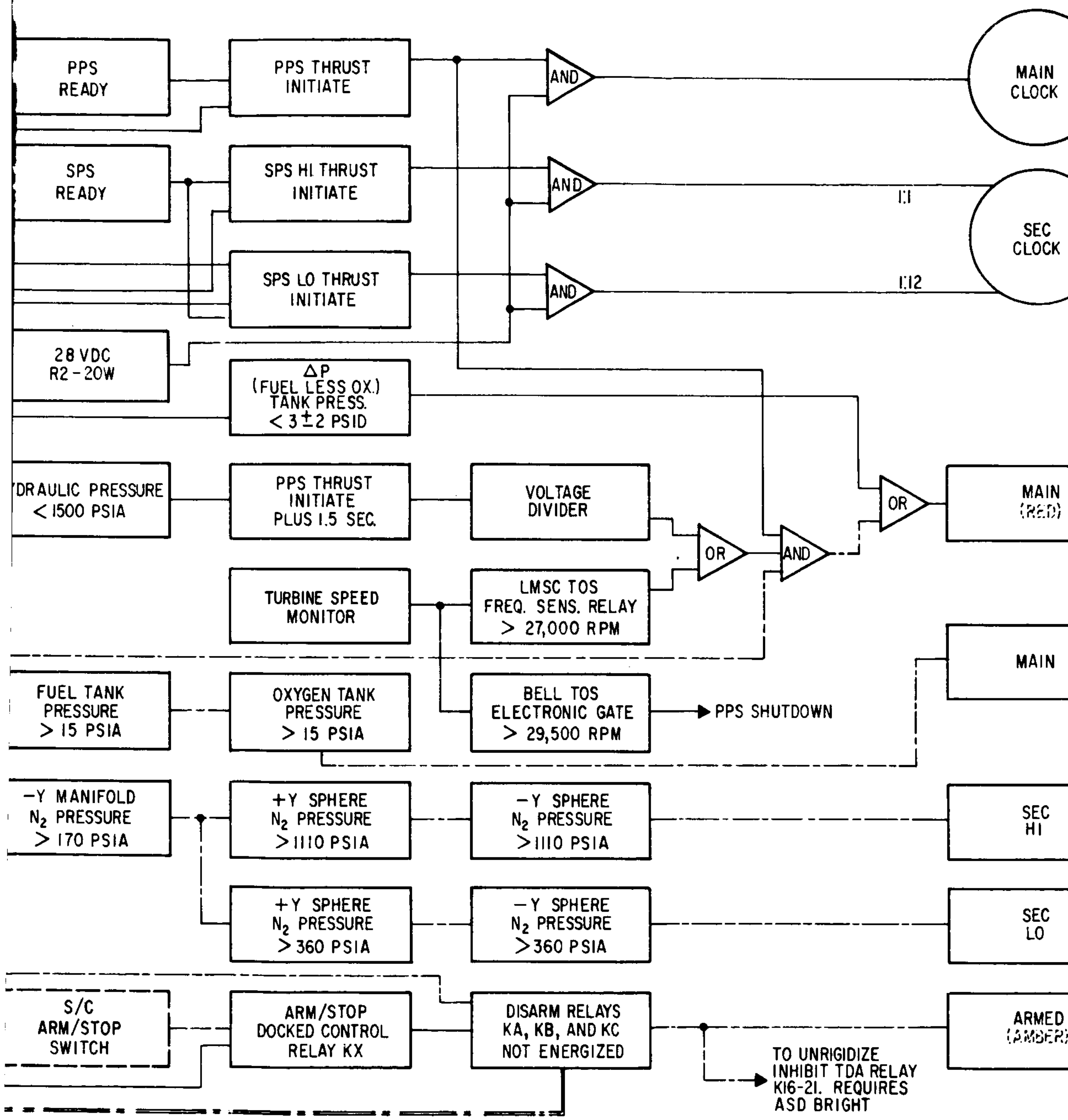
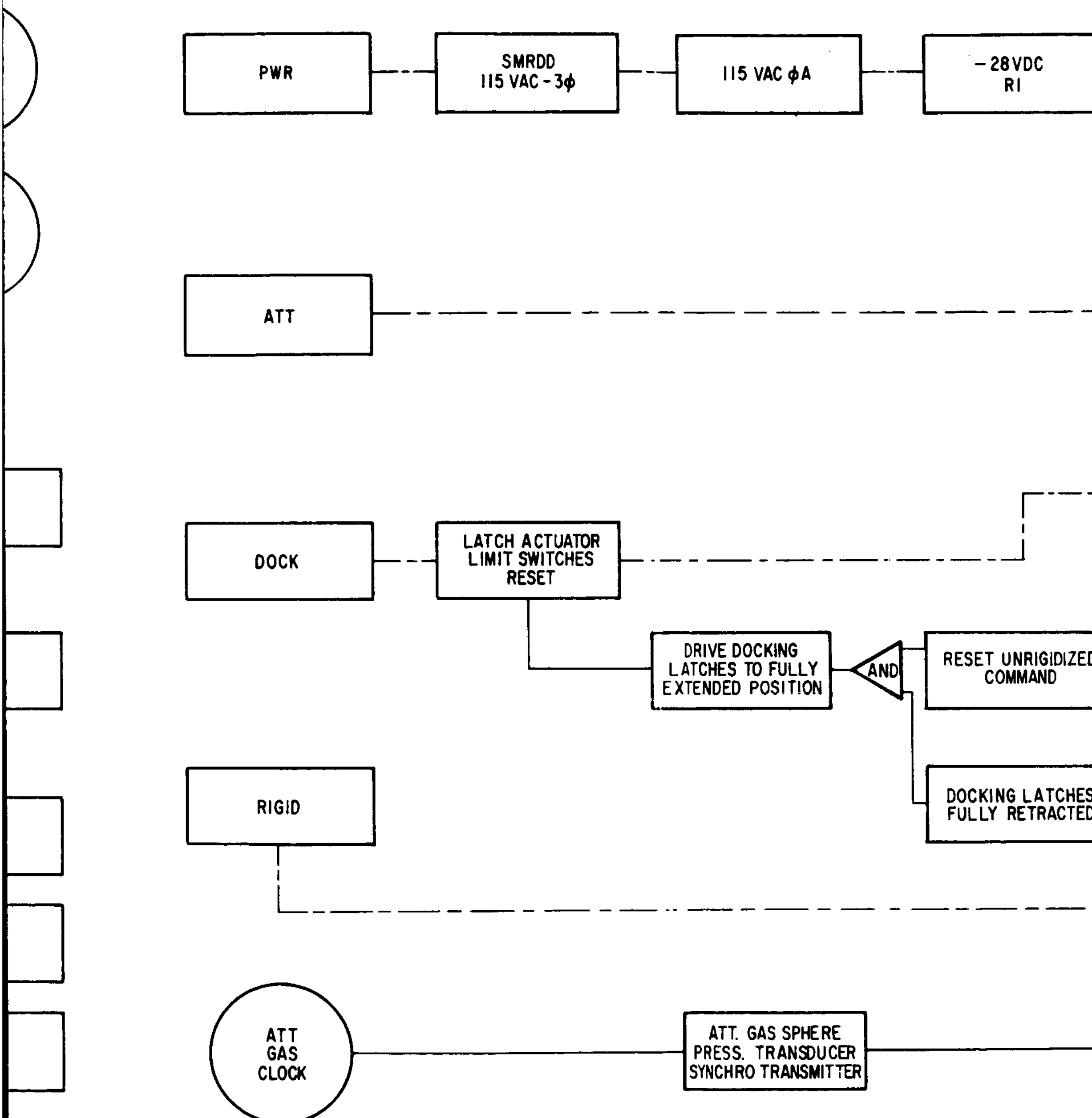


Figure III. C-18. Agena Status Display Panel


28 VDC UNREG.
PPS ON/OFF COMMANDS 501, 510, 511/500
ENGINE SEQUENCER
DISARM RELAYS KA, KB, AND KC NOT ENERGIZED
SPS ON/OFF COMMANDS 561+(551, 560)/550
28 VDC R2-60W
HY
ASD ON COMMAND
211 DIM 11 VDC
201 BRIGHT 28VDC
28 VDC FOR BRIGHT OR DIM CMD
HYDRAULIC RETURN PRESSURE > 50 PSIA
+Y MANIFOLD N_2 PRESSURE > 170 PSIA
ARM/STOP CONTROL
DOCKED/SCC 131
UNDOCKED/SCC 130
ARM/STOP COMMAND RELAY KY

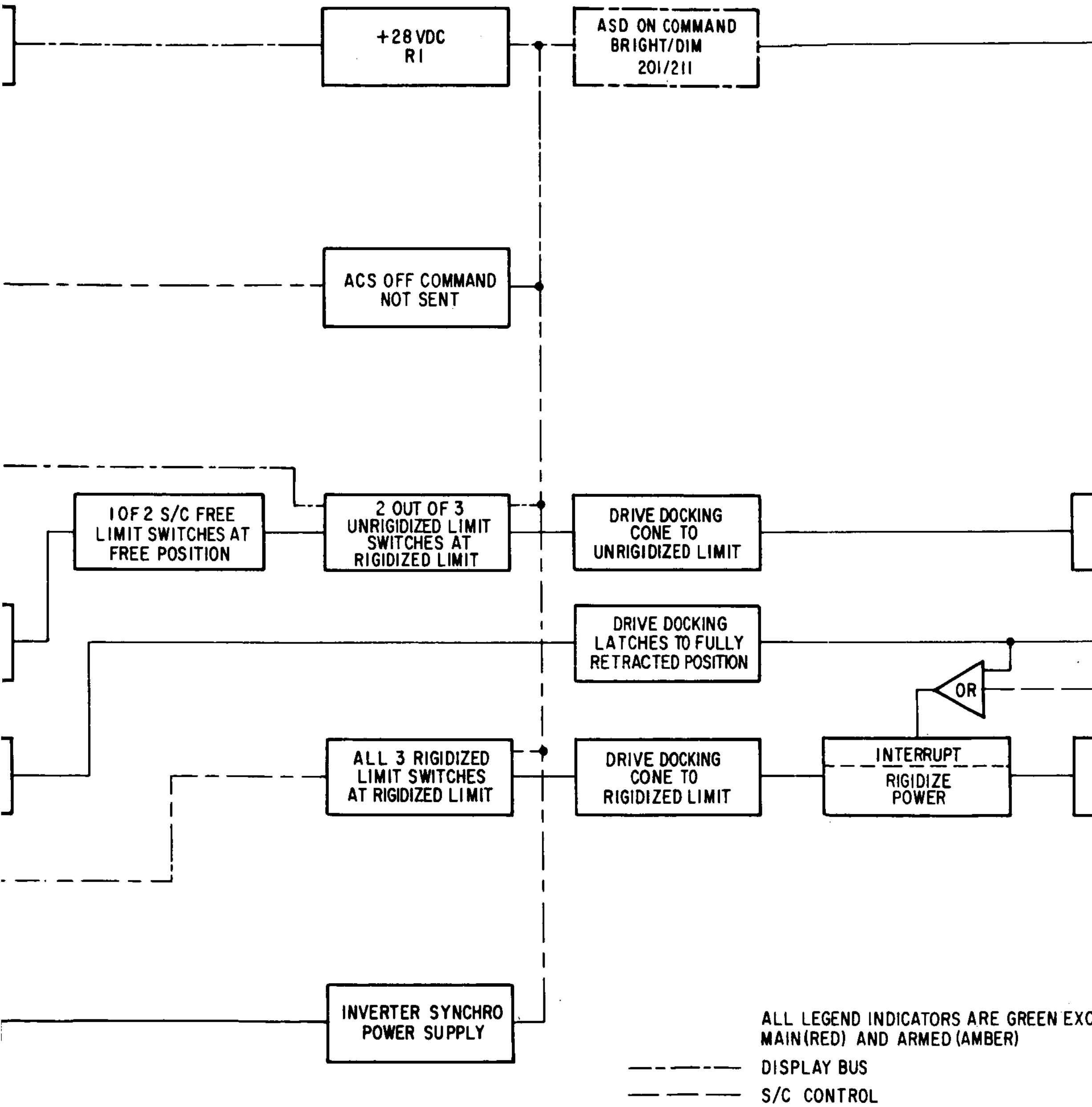


PPS READY
PPS THRUST INITIATE
AND
MAIN CLOCK
SPS READY
SPS HI THRUST INITIATE
AND
I:1
SEC CLOCK
SPS LO THRUST INITIATE
AND
I:12
28 VDC R2-20W
ΔP (FUEL LESS OX.) TANK PRESS. < 3 ± 2 PSID
YDRAULIC PRESSURE < 1500 PSIA
PPS THRUST INITIATE PLUS 1.5 SEC.
VOLTAGE DIVIDER
OR
AND
OR
MAIN (RED)
TURBINE SPEED MONITOR
LMSC TOS FREQ. SENS. RELAY > 27,000 RPM
MAIN
FUEL TANK PRESSURE > 15 PSIA
OXYGEN TANK PRESSURE > 15 PSIA
BELL TOS ELECTRONIC GATE > 29,500 RPM
PPS SHUTDOWN
-Y MANIFOLD N2 PRESSURE > 170 PSIA
+Y SPHERE N2 PRESSURE > 1110 PSIA
-Y SPHERE N2 PRESSURE > 1110 PSIA
SEC HI
+Y SPHERE N2 PRESSURE > 360 PSIA
-Y SPHERE N2 PRESSURE > 360 PSIA
SEC LO
S/C ARM/STOP SWITCH
ARM/STOP DOCKED CONTROL RELAY KX
DISARM RELAYS KA, KB, AND KC NOT ENERGIZED
ARMED (AMBER)
TO UNRIGIDIZE INHIBIT TDA RELAY K16-21. REQUIRES ASD BRIGHT



PWR
SMRDD
115 VAC - 3ϕ
115 VAC ϕA
- 28 VDC
RI
ATT
DOCK
LATCH ACTUATOR
LIMIT SWITCHES
RESET
DRIVE DOCKING
LATCHES TO FULLY
EXTENDED POSITION
AND
RESET UNRIGIDIZED
COMMAND
DOCKING LATCHES
FULLY RETRACTED
RIGID
ATT
GAS
CLOCK
ATT. GAS SPHERE
PRESS. TRANSDUCER
SYNCHRO TRANSMITTER

III G-38-2



+28 VDC
R1
ASD ON COMMAND
BRIGHT/DIM
201/211
ACS OFF COMMAND
NOT SENT
1 OF 2 S/C FREE
LIMIT SWITCHES AT
FREE POSITION
2 OUT OF 3
UNRIGIDIZED LIMIT
SWITCHES AT
RIGIDIZED LIMIT
DRIVE DOCKING
CONE TO
UNRIGIDIZED LIMIT
DRIVE DOCKING
LATCHES TO FULLY
RETRACTED POSITION
OR
ALL 3 RIGIDIZED
LIMIT SWITCHES
AT RIGIDIZED LIMIT
DRIVE DOCKING
CONE TO
RIGIDIZED LIMIT
INTERRUPT
RIGIDIZE
POWER
INVERTER SYNCHRO
POWER SUPPLY
ALL LEGEND INDICATORS ARE GREEN EXC
MAIN (RED) AND ARMED (AMBER)
DISPLAY BUS
S/C CONTROL

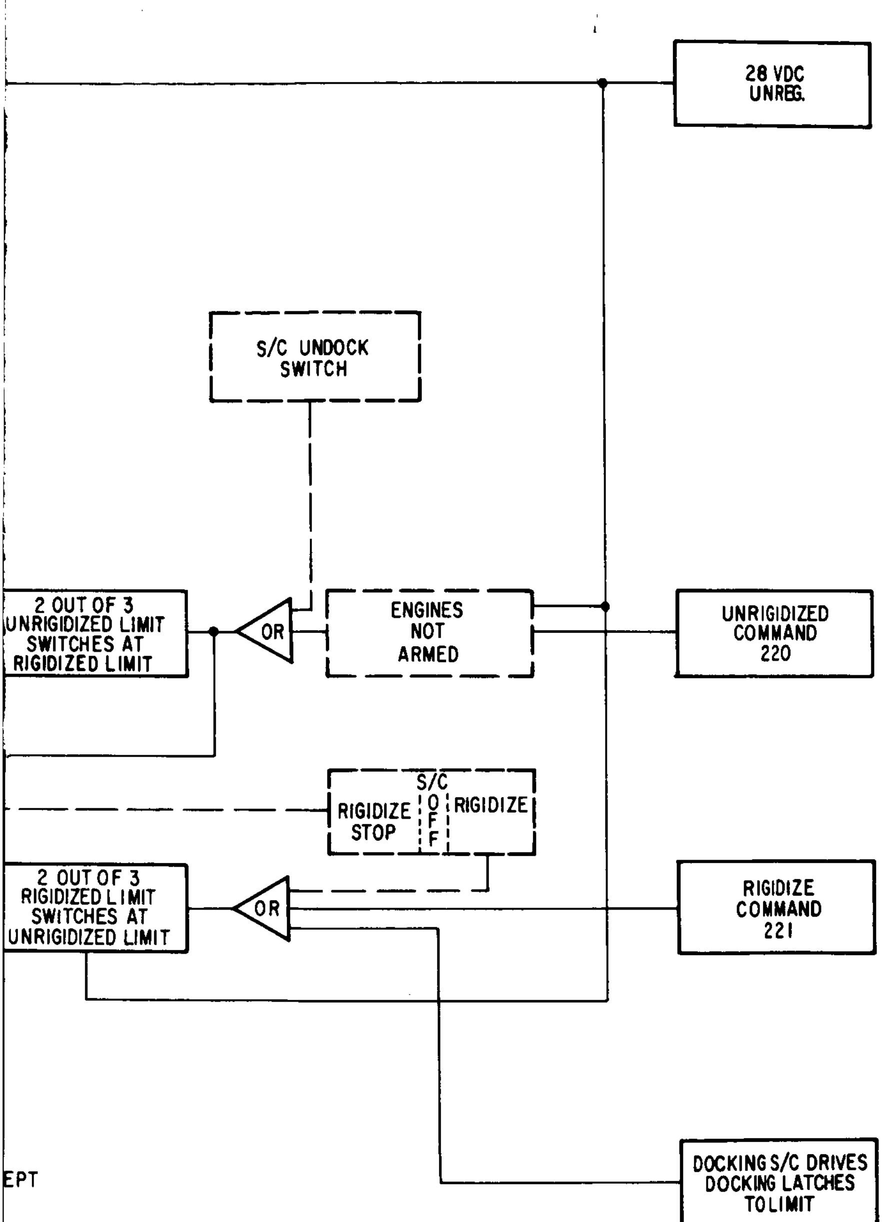


Figure III. C-19. Agena Target Display Logic

III G-38-3

8. ATLAS/AGENA INTERFACE

The mechanical interface between the Atlas SLV-3 and the GATV consisted of a booster adapter section which stayed with the Atlas at the time of GATV separation. Separation was initiated by the detonation of a circumferential primacord and the operation of two retrorockets which slowed the booster section so that the target vehicle could slide out on rails in the adapter. First motion disconnected the pullaway umbilical generating a backup signal to start the ascent sequence timer. Three detents in the booster adapter actuated a switch on the GATV to permit telemetry monitoring of GATV motion in the adapter, while a spring-loaded lever, released as the GATV left the booster adapter rails, actuated two separation switches which enabled the GATV pneumatic attitude control system and gave a telemetry indication. Two other separation switches served the purpose of initiating the self-destruct system in the booster adapter in the event of premature separation of the GATV during the SLV-3 boost phase. The self-destruct circuitry, which could also be actuated by ground command to the Atlas range safety receivers, was disabled by the SECO signal in the SLV-3 with backup provided by the VECO signal. The VECO signal also crossed the interface to the GATV to uncage its gyros.

Operation of the Atlas Agena interface and separation system was faultless with the exception that the separation monitor on GATV 5002 and 5003 failed to transmit the 3 distinct voltage steps representative of GATV motion in the adapter. Laboratory tests conducted after the second malfunction which could have been caused by a shorted capacitor showed that the dynamic characteristics of the switch mechanism generated multiple actuations. Reduction of switch travel and a change in spring characteristics corrected the problem.

SECTION III. C

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SECTION III. C

DEFINITIONS

ATT	Attitude
BELL	Bell Aerosystems Company
BECO	Booster Engine Cutoff
C and C	Command and Communication Subsystem
CG	Center of Gravity
EMI	Electro-Magnetic Interference
ERT	Emergency Reset Timer
FCEP	Flight Controls Electronic Package
FCLP	Flight Command Logic Package
FREQ	Frequency
GAT	Gemini Agena Target
GATV	Gemini Agena Target Vehicle
GFE	Government Furnished Equipment
H/S	Horizon Sensor
IRFNA	Inhibited Red Fuming Nitric Acid
IRP	Inertial Reference Package
LMSC	Lockheed Missiles and Space Company
MAIN	Pertaining to PPS
MAP	Message Acceptance Pulse
MON	Mixed Oxides of Nitrogen
OFPS	Oxidizer Feed Pressure Switch
OMPS	Oxidizer Manifold Pressure Switch
PAM	Pulse Amplitude Modulation
PCM	Pulse Code Modulation
POSV	Pyro-Operated Solenoid Valve
PSIA	Pounds per square inch absolute
PWR	Power
RTC	Real Time Commands
S/C	Spacecraft
SEC	Pertaining to SPS
SEC	200 pounds thrust SPS
SEC LO	16 pounds thrust SPS
SECO	Sustainer Engine Cutoff

SENS	Sensitive
SMRDD	Spin motor rotation and direction detector
SPC	Stored Program Commands
SPS	Secondary Propulsion System
TDA	Target Docking Adapter
TOS	Turbine Overspeed
UDMH	Unsymmetrical Dimethylhydrazine
UHF	Ultra High Frequency
VECO	Vernier Engine Cutoff
VHF	Very High Frequency
V/M	Velocity Meter
W	Watts
28 VDC UNREG	28 VDC unregulated bus
28 VDC R1	28 VDC regulated bus #1
28 VDC R2 - 20 W	20 Watts branch of 28 VDC regulated bus #2
28 VDC R2 - 60 W	60 Watts branch of 28 VDC regulated bus #2

D. GUIDANCE AND PERFORMANCE ANALYSIS

1. GEMINI ATLAS AGENA TARGET VEHICLE SYSTEM ASCENT GUIDANCE

a. Introduction

All Gemini Atlas Agena Target Vehicles (GAATV) were launched by the AF 6555th ATW at Cape Kennedy, Florida. AFSSD had prime responsibility for the development and implementation of GAATV ascent guidance which was accomplished by the Mod III Radio Guidance System consisting of the Burroughs Computers and GE Mod III System. Lockheed Missiles and Space Company (LMSC) as integrating contractor, furnished the reference trajectories, range safety package and flight termination system reports. General Dynamics/Convair (GD/C) provided the SLV-3 flight termination system reports and conducted a pre-flight data exchange with LMSC. The guidance equation contractor, Thompson-Ramo-Wooldridge (TRW), provided GAATV ascent guidance equations and associated documentation as well as tray wiring information. Burroughs Corporation wired the guidance trays and operated the Burroughs computers. General Electric (GE) provided the guidance canisters and operated the GE Mod III System.

b. Injection Requirements

NASA/Manned Spacecraft Center (MSC) specified the first orbit requirements for the GAATV to the Air Force. Their requirements, completed in January of 1965, were that all GATV's would be injected into a near circular 161 n mi orbit at an inclination angle of 28.87 degrees. Two additional requirements were specified as follows:

1) The GAATV guidance equations would have the capability of either steering for the nodal point or for the inclination angle.
2) The GAATV would have the capability of either using or not using the biased launch azimuth (dog-leg).

The requirement for nodal point steering was generated by MSC because control of the inertial longitude of the ascending node was more important than control of the inclination angle. MSC selected nodal point steering for all six GAATV missions and, to accomplish this, specified the desired inertial longitude of the ascending node for each mission. In the fall of 1964, TRW suggested and proved that the launch azimuth of the GAATV could be biased (dog-leg) to permit a 7-1/2 minute extension of the GLV spacecraft launch window. This biased launch azimuth was used on the Gemini VI and VIII missions.

c. Development and Description of GAATV's Guidance Equations

The TRW Systems Group guidance equation program, developed for the AF/NASA Atlas-Agena Ranger-Mariner (Block Change) Contract was used as the basis for the GAATV equations program. The Ranger-Mariner equations program was modified to include deadreckoning, nodal point steering, and yaw look angle constraints. In the GAATV equations, the radar data is filtered (Kalmin filter) to obtain the best estimate (nearly minimum variance) of the vehicle's position, velocity, and acceleration vectors, independent of the amount of time sequence of data received. As a result, an intermittent or complete failure of either the track or rate system does not cause a large decrease in guidance accuracy. By use of this filter, the navigation problem is separated from the control problem resulting in maximum flexibility with respect to different missions. The guidance equations also had capability of steering in either pitch or yaw (or both) during the SLV-3 booster phase. This permitted the correction of 3-sigma booster dispersions without large attitude maneuvers during the sustainer phase.

d. GAATV Ascent Guidance Working Group

To coordinate all GAATV ascent guidance activities and to resolve all ascent guidance problems, SSVT, in May of 1964, formed the GAATV Ascent Guidance Working Group. These meetings, chaired by SSD, consisted of representatives from NASA, SSD, 6555th ATW, LMSC, GD/C, TRW, GE, and Burroughs. These meetings were held as necessary to complete the GAATV ascent guidance tasks.

e. Guidance Equations Design Review Board

The TRW Guidance Equations contract called for Guidance Equation Design Review Boards (first on internal TRW DRB followed at a later date by a Government DRB). The purpose was to review and certify all of the TRW's effort. AFSSD invited MSC and LMSC to the Government DRB's. These meetings were held at TRW Systems Group in Redondo Beach, California. Several were held during the guidance equation development stage and a DRB was held for each GAATV mission.

f. GAATV Ascent Guidance Certificates

In the area of ascent guidance, the Air Force felt that certain tasks and areas of responsibility should have additional emphasis and contractor responsibility. LMSC, TRW, GE, and Burroughs agreed to provide certificates for each flight. These signed certificates were received by SSD prior to each launch and were available at each GAATV Flight Safety Review. Basically, these certificates were as follows:

LMSC: LMSC as integrating contractor was responsible for verifying that the GAATV guidance equations and manual constants satisfied the mission requirements.

TRW Systems Group: TRW certified that they had completed an internal review of the guidance equations and computer program for the upcoming launch and that the equations as programmed for the Burroughs Computer were flight-ready.

General Electric: GE certified that their missile-borne guidance sets assigned to support the upcoming GAATV launch were flight-ready and that the GE Mod III System was ready for launch.

Burroughs: Burroughs ran an independent check of the TRW-supplied computer program and certified it to be correct. They also certified that the Burroughs A-1 computing system was flight-ready.

g. GAATV Ascent Guidance Performance

The ascent performance of the GAATV was excellent. The desired initial injection conditions were met for each successful injection into orbit. Even on the Gemini VI and IX GAATV failures, the ascent guidance system performed satisfactorily until failure occurred. On the four successful launches into orbit, all dispersions, errors, etc., were well within the three sigma limits of GAATV ascent guidance calculated.

2. GEMINI ATLAS AGENA TARGET VEHICLE MISSION PLANNING

a. Definition of First Orbit Requirements

Although NASA/MSC had informally stated the proposed orbital requirements at various Trajectory and Orbits Meetings, no firm decision was reached by MSC until early 1965. To permit maximum flexibility to MSC, the Air Force on 20 October 1964, directed LMSC to study the possibility of a number of first orbits within a certain volume of space. The size of this proposed mission box (volume of space) was as follows:

1) Orbit altitude: 121 to 201 nautical miles

2) Orbit inclination: 28.4913° to 29.87°

3) Eccentricity: from near circular to 0.0111

After discussion, MSC/GPO stated by TWX on 7 Jan 1965 that the target orbit altitude for Mission GT-6 would remain as a 161 nautical mile circular orbit and that this altitude was planned for subsequent missions.

With the initial orbit specified, the only variation from launch to launch would be the amount of biased launch azimuth (dog-leg) and the ascending nodal position requirement which would be defined by NASA for each flight.

b. On-Orbit Requirements for the GATV

The requirements of the GATV on-orbit were more easily defined. At a Trajectory and Orbits Meeting on 14 August 1963, basic ground rules for GATV on-orbit maneuvers were discussed and agreed upon by NASA, Air Force, and LMSC. Primary emphasis in the development of the ground rules was given to establishment of maneuvers assuming greatest probability of rendezvous success. Some of these basic ground rules were:

1) GATV in-plane and out-of-plane maneuvers shall not be attempted during a maneuver sequence with a single thrust.

2) The GATV pitch and yaw capabilities shall be employed individually and never in combination.

3) The maximum GATV apogee of the in-plane dwell maneuver is limited only by the available delta velocity in the GATV.

To summarize, the GATV could be called upon to make a wide variety of maneuvers in-plane and out-of-plane, docked or undocked.

c. Post-Rendezvous Mission Planning

LMSC and the Air Force had forwarded to NASA/MSC in 1963-64 several recommendations on post-rendezvous tests and maneuvers. In September of 1965, MSC officially requested SSD/Aerospace and LMSC assistance in this area. On 30 September 1965, the review of the MSC Post S/C Retro Fire/Recovery Maneuvers Plan for the Agena Target Vehicle was sent to NASA. Several additional maneuvers and tests were also proposed. The importance of this September 1965 activity was that the decisions reached and the maneuvers discussed did not vary during the remainder of the program. Due to the two GAATV failures and other flight problems, there were few opportunities to perform these maneuvers and thus no need to recommend new ones.

d. Trajectory and Orbits Panel

The Trajectory and Orbits Panel was a NASA/MSC mission planning meeting always convened at MSC. LMSC and the Air Force were members of this panel and had representatives at all meetings to which they were invited. The meeting discussed all aspects of the mission: launch, rendezvous, EVA, experiments, post-rendezvous maneuvers, and recovery.

SECTION III. D

DEFINITIONS

AFSSD	Air Force Space Systems Division
ATW	Aerospace Test Wing
DRB	Design Review Board
GAATV	Gemini Atlas Agena Target Vehicle
GATV	Gemini Agena Target Vehicle
GD/C	General Dynamics/Convair
GE	General Electric
GPO	Gemini Program Office
LMSC	Lockheed Missile and Space Company
MSC	Manned Spacecraft Center
NASA	National Aeronautics & Space Administration
SLV	Standard Launch Vehicle
TRW	Thompson Ramo Wooldridge

E. SPECIAL STUDIES

1. TEN POINT PROGRAM

Because of the number of failures and rework problems during initial manufacturing stages of the command and communications subsystem, it became apparent that certain mechanical and electronic design deficiencies existed. Early in 1965 a "ten point program" was initiated to perform circuit and packaging design reviews, 100% electrical testing of C&C piece parts, worst case analysis of critical modules, and several specific quality control actions.

Performance of these tasks resulted in redesign of a number of programmer circuits and several packaging changes to improve manufacturability. In addition, failure analysis and corrective action down to the piece part level was expedited. Closer monitoring of vendor piece part processing was accomplished, and additional mandatory in-process quality control inspection points were established in the manufacturing and rework areas.

The success of the ten point program is reflected in the fact that no failures occurred in the command system during any GATV flights.

2. PROJECT SUREFIRE

The flight of target vehicle 5002 resulted in failure to obtain orbit. A review of flight data strongly indicated that a hard start of the main Agena propulsion system was the initiating cause of the failure. The most probable cause of the hard start was determined to have been the result of a fuel lead during engine ignition. This conclusion was agreed upon by a group of industry propulsion experts who were assembled in a symposium to evaluate this hypothesis. The GATV had been intentionally sequenced for a simultaneous/slight fuel lead propellant start; this was done to conserve oxidizer for the many programmed restarts.[1] All previous standard Agena engines had utilized an oxidizer lead propellant start.

Because of the possible destructive nature of this type of failure and the program objective of utilizing the Agena engines during docked modes, an intensive investigation was undertaken to correct this condition. This program, titled "Project Surefire," was concerned with the investigation of the cause of the failure of GATV 5002, the resulting system modification, redevelopment, and flight worthiness demonstration as required to insure adequacy of the system for manned usage. Surefire was to culminate in the best flight configuration from both the hardware and procedural standpoints. During the investigation and subsequent modification/test evaluation period, a number of studies were conducted. A few of these which did result in hardware modification are presented.

- It was noted from flight records that the 2g accelerometers mounted on the engine cone were damaged by the separation shock to the extent that they ceased to function. This led to the conjecture that the separation shock, caused by firing the primacord used to separate the GATV from the booster adapter structure, might be strong enough to damage components mounted in the aft rack and on the engine cone structure. Shock mounts were designed for these components and a series of shock tests conducted to verify the integrity of the shock-mounted components (Reference: LMSC Report A796480, GATV Pyrotechnic Shock Tests and Dynamic Response Tests, 1 March 1966). The integrity of the shock-mounted components was satisfactorily demonstrated and the airborne accelerometers were replaced with a more shock-resistant 5g unit.
- The propulsion system start sequencing was modified as described in Section III. C. 3, returning the engine to an oxidizer lead configuration. This redesign/retest, both on the component and engine level at BAC and under hard vacuum conditions at AEDC, comprised the major effort of the Surefire program and was successfully concluded.
- During the design and reliability reviews which resulted in utilizing two pressure switches (OMPS, OFPS) in the POSV control circuitry, it became apparent that a lock-in relay

incorporated in the same circuitry had a failure mode which could result in power being prematurely applied to the POSV. This premature activation of the POSV would negate the desired oxidizer lead. The pressure switch relay box was redesigned to eliminate this lock-in relay.

- o As a result of evaluation of 5002 data and review of data from several standard Agena flights which experienced slow-opening fuel valves, it became apparent that with the Gemini start system an inadvertent overspeed shutdown could occur during ascent burn. If such a shutdown did occur during ascent, the effect would be vehicle destruction resulting from overpressurization of the main propellant tank. To eliminate this condition, a wiring change was made so that the circuitry to the engine solenoids did not pass through the overspeed trip and that the overspeed trip was not enabled during the ascent burn portion of the overall mission.
- o In the course of the component review and evaluation, the microswitch elements utilized in the OMPS and OFPS were found to be not hermetically sealed, and hence, subject to a vacuum environment in orbit. Because of the requirement for on-orbit life of up to 5 days, a condition of cold welding of the metallic surfaces in close contact could be experienced. A test program was satisfactorily conducted to verify switch integrity at pressure altitudes of 200 nautical miles.

The Project Surefire test series, including component/subsystem/engine level testing at BAC, and engine level hard vacuum testing at AEDC, was successfully completed and the modified engine verified by the remaining GATV flights. Both Bell and Lockheed have issued detailed reports covering all phases of study/tests on the Surefire program.[2, 3, 4]

3. ATLAS GATV 5002 INTERFACE AND GATV EXPLOSION EFFECTS

During the GATV 5002 failure investigation, the SLV-3/GATV interface and the target vehicle separation time history were carefully reviewed for any anomalies that might have contributed to the GATV 5002 failure. Results of this review were that both the SLV-3 and GATV sides of the interface and separation were completely clean with the exception of the failure of the GATV 5002.

After the failure investigation, a detailed review of SLV-3 data from the time of GATV 5002 separation through primary propulsion system (PPS) start and explosion was made to determine whether this data could be used to help establish a safe separation distance between the S/C and GATV in the event of an impending Agena explosion.

On the basis of Atlas accelerometer, rate gyro and tank pressure data, the following was established:

- Atlas-Agena separation was approximately 190 feet at PPS start, and approximately 270 feet at the time of explosion.
- Mild disturbances without ringing were detected in SLV-3 data 0.9 and 1.4 seconds after PPS start and vehicle explosion respectively, representing pressure waves traveling at an average of 200 feet/second. Absence of ringing and of tank pressure decay indicated strongly that the SLV-3 was not hit by any debris of significant mass.
- SLV-3 attitude at the time of these events was essentially end-on, representing a minimal surface area comparable to that of the spacecraft.

In summary, the data from this single event constituted too small a sample to consider a reduction in the 1500 foot minimum stand-off distance in the event of a threatened GATV explosion in orbit.

4. SHROUD SEPARATION

The nose shroud system used on the GATV was a modified A-12 Shroud System designed and built by Douglas Aircraft Co. The original modifications for the GATV consisted of removing thermal insulation from the interior of the shroud. As a result of a failure of the A-12 system on a Comsat flight, the ability of the system to separate properly in the GATV environment was questioned. LMSC conducted an extensive failure analysis of the Comsat system which was not successful in identifying the failure mode; however, they recommended three modifications to preclude the most probable failure modes. These modifications were:

- Installation of a tension band to correct warpage and ensure roundness at the mating fitting.
- Modification of the pivot brackets to provide greater travel distance and stronger local structure.
- Changing of the lower actuator springs to provide more force.

Aerospace Corporation conducted a complete evaluation of these modifications and recommended the adoption of the first two, but not the third. This recommendation was accepted and the modified system was tested at the Lockheed Rye Canyon facility. The tests (ref. Lockheed Calif. Co. Report no. LR 19111, dated 15 September) were completely successful and the shroud system functioned satisfactorily on all GATV flights. It is noted that the shroud failure during the Gemini IX mission (The "Angry Alligator") was the result of personnel error and in no way reflects on the reliability of the shroud separation system.

5. DOCKED BURN STABILITY

The GATV controls system was originally designed to provide stable flight for an Agena D and payload. In the Gemini mission, it was required to provide stability during docked burns. The original system was designed to filter out all Agena body bending modes which were greater than 8 cps. The system could be modified by a gain change to handle modal frequencies as low as 5 cps; however, the docked GATV and spacecraft had a fundamental body bending mode with a frequency between 2 and 4 cps. A lead-lag circuit described in III.E.6 was designed by LMSC to cope with this mode, and stability studies were run at MIT Instrumentation Labs to check out the modified system.

The fundamental mode in question involved rigid body motion of the GATV and spacecraft with a flexible spring (the Target Docking Adapter) connecting them. Preliminary stiffness data from McDonnell Aircraft Corporation showing both in-plane and out-of-plane response, when incorporated in the MIT model, indicated the inability of the modified system to provide stability. It was decided to run a dynamic response test at McDonnell to provide better data for the analysis. The results of this test showed considerably more out-of-plane coupling in the fundamental mode than expected. The frequency of this mode was found to be between 2.5 and 3.0 cps depending on the weight condition. Structural damping varied between 2.0 and 5.0 percent. In the course of evaluating the test data, errors in handling the out-of-plane response were discovered in the MIT model. With the model corrected and with the use of lower bound damping values, the lead-lag modification as proposed by LMSC was shown to provide adequate stability. The modification was flown on 5003 and subsequent GATV's.

6. FLIGHT CONTROLS ELECTRONIC PACKAGE - HYDRAULIC CHANNEL LEAD-LAG SHAPING NETWORK

As soon as the modal response of the docked Agena-Spacecraft had been established by studies at Massachusetts Institute of Technology (MIT) and these results accepted by the affected contractors, the flight control electronics compensation was established. Previous studies by LMSC had shown that a modification to the lead-lag shaping already in existence could handle both the ascent dynamics and the docked dynamics with a minor change in loop gain between two flight modes. The MIT simulation of the vehicle was increased to include the flight control system and the potential of the revised lead-lag was confirmed.

Lockheed proceeded to mechanize and optimize the lead-lag design with the use of a single axis digital computer simulation. Hardware components and hardware tolerances were evaluated. The most difficult development item in the change was the perfection of the temperature-stabilized operational amplifier. Actual bread-boarded parts were tied into the single axis simulator for making temperature tests as well as system performance evaluations. This phase of the testing also was used to perfect test procedures and test tolerances that would ensure proper system performance.

The physical configuration of the Standard Agena electronic card modifications were smaller satellite cards mounted on standoffs above the basic pitch and yaw hydraulic channel cards. The only problems encountered in building and checking the production lead-lag boards were with wiring and soldering. The harnesses which attached the satellite boards to the first units were cut too short. This caused lead breakage in assembly and test. Removal of the Standard Agena piece parts prior to the modification resulted in broken printed circuit traces and lifted eyelets. These difficulties were overcome in sufficient time to meet the GATV 5003 effectivity.

Qualification was accomplished with vibration difficulty experienced on the first attempt. An old laboratory unit was used which had non-production assembly techniques used on some transformers and capacitors. These parts could not pass the vibration test. New production cards were modified and vibration and temperature/altitude tests were completed with no further problems.

7. CENTER OF GRAVITY (C-G) OFFSET EFFECT

A major problem occurred on GATV 5003 during undocked on-orbit PPS powered flight. A significant vehicle yaw heading error existed (hence a velocity vector error) which affected the orbital guidance computations and resulted in adverse orbital ephemeris accuracies when making out-of-plane orbit changes. This yaw heading error was due to a combination of yaw c-g offset, slow control system response time and the vehicle dynamics. The yaw c-g offset was approximately twice that of the Standard Agena due to the addition and location of two running light batteries (Type XVII). The slow control system response time was an effect caused by the redesign of the flight control electronics package. This redesign had been required to provide stable control system operation during the docked mode.

Orbital altitude errors ranged up to approximately 120 miles during PPS operation. The errors were much more pronounced when the vehicle was in a ± 90 degree configuration and a plane change was attempted. This was due to the offset being in the yaw direction and the velocity component error combining directly with the orbital velocity. These errors greatly exceeded 3 sigma values derived in prior error analyses and on-orbit guidance computations. The mechanics of this problem and the interactions of c-g offset, attitude error, actuator position, vehicle mass properties, time, etc. are clearly explained in the referenced document.[5]

Various solutions to the c-g problem were investigated. These consisted of removing batteries, realigning the engine, adding ballast, off-loading SPS propellant, correction tables and combinations of the aforementioned. The following changes were made to the GATV to correct the problem by realigning the vehicle c-g:

1) Removed one type IC battery (Prime Power)
2) Removed one type XVII battery (Running Light Power)
3) Replaced magnesium access door with stainless steel door
4) Installed two dummy battery spacers
5) Added lead ballast (approximately 158 pounds)
6) Prepared correction tables (for use in trimming out potential c-g dispersions)

The referenced report contains correction tables and a method of making on-orbit corrections to guidance calculations for both docked and undocked modes. A parametric study was performed which related pitch and yaw attitude errors to c-g offsets for the GATV during PPS operation. Attitude errors were determined as a function of burn time, vehicle c-g offsets and vehicle weight. Results were plotted as a family of curves to provide programmed attitude correction data for desired orbit changes. Average attitude error and actuator position for various PPS burn times, along with transient attitude and actuator position response curves were presented.

The above solution was implemented successfully and good results were obtained for the remaining GATV flights.

8. PROBABILITY OF METEOROID IMPACT AND PROPELLANT TANK BULKHEAD REVERSAL

In reviewing what failures of the GATV could lead to catastrophic consequences to the spacecraft when docked or in the immediate vicinity, it was determined that the reversal of the inner propellant bulkhead, separating the fuel and oxidizer tanks, would be the most probable cause[6].

Independent studies by LMSC[7,8] and Aerospace were conducted with the purpose of determining the mechanism of such failure, the probability of it occurring, and the warning time available to the crew. Concurrently, a test program was undertaken by LMSC[9] to determine the tank pressure differential at which the bulkhead would actually collapse.

While the quantitative results of these studies showed a fairly wide spread mainly due to the lack of good empirical data in the flight regime of interest, qualitatively they were in agreement:

- Most probable mechanism of failure would be meteoroid puncture of the fuel tank.
- Most probable hole size would result in fuel tank pressure decay rates low enough to allow the S/C to separate from the GATV and move to a safe distance, provided the crew was informed as soon as the Δp reached a critical level.
- Probability of fuel tank puncture during 8 hour docked sleep periods, when the crew could not take evasive action, was well below the maximum acceptable level of risk of 0.001.

Some consideration was given to the feasibility of providing for an audible alarm in the spacecraft operating off the Red MAIN circuit, or for venting of the oxidizer tank on the Agena, but both considerations were dropped on the basis of cost, schedule, and low probability of meteoroid penetration.

A summary report on meteoroid penetration probability, in the light of latest empirical and analytical data as applied to the Gemini Program, was published by Aerospace Corporation[10].

9. CHAMBER PRESSURE DROPOFF

During the course of the GATV program and before the last GATV flight there were four standard Agena flights that experienced a Pc dropoff anomaly. This anomaly consisted of a chamber pressure drop of 30 to 50 psi, usually occurring at 100 to 150 seconds after start of engine burn. Within two seconds the chamber pressure would recover to within 10 psi of its original value where it remained with no further disturbances for the duration of the firing. There have been no unsuccessful missions related to this anomaly and three of the four flights successfully accomplished restarts.

This anomaly has been continuously under study since the first occurrence; a sea level turbopump test led to the first possibility of the cause. During this test the pump speed dropped and recovered similar to the flight chamber pressure traces. A subsequent pump disassembly showed that the fuel ball bearing had 3 flat balls. A further study revealed that this bearing, prior to assembly, was heavily packed with a grease that was non-compatible with the rubber lipseal next to it. It was felt that the bearing failure might have been caused either by the grease effecting a lipseal failure thereby allowing the fuel to contaminate the bearing or that the heavy grease packing caused the balls to skid instead of roll. As a corrective action, future pump bearings were not to be packed in grease prior to assembly. In addition, the outer race bearing carrier of the failed fuel bearing was slightly undersized causing a tighter press fit than normal. All future bearing carriers were dimensionally checked prior to pump assembly.

A Pc dropoff similar to the previous four cases occurred on the last GATV flight (5001). The turbopump on this engine was assembled with all of the improvements previously discussed. A further study effort by Lockheed, Bell and Aerospace has concluded that additional flight instrumentation on Standard Agena flights is required in addition to a ground simulated altitude pump test program. One theory that resulted from the study was the possibility that hot gas leakage past the turbine seal could be causing a turbine bearing failure.

10. EXTENDED AGENA TARGET VEHICLE LIFETIME STUDY

The purpose of this study[11] was to examine the existing data to determine the engineering feasibility of extending the GAT orbital lifetime from 5 days up to a maximum of 105 days by employment of a deactivation mode. The results showed that, with minor changes, this extension of orbital lifetime was feasible; however, no additional action was taken by MSC in this area.

11. ORBIT DECAY STUDY FOR GEMINI VI AGENA TARGET VEHICLE 5002

The purpose of this study[12] was to determine the shutdown mode and recommend an altitude at which the Gemini VI GATV could be maneuvered on 30 October 1965 (assuming a 25 October 1965 launch) so that it could be available as a non-powered, non-stabilized back-up target for the Gemini VIII mission. The LMSC recommendation was that the GATV be slowly tumbled in the orbit plane at a rate of 1.5 degrees per second after injection into a 208 nm circular orbit on 30 October 1965.

12. GEMINI AGENA TARGET ALTITUDE LIMITATION STUDY

This study[13] was initiated by SSVT primarily to ensure that the only limitation in making high altitude maneuvers should be that limitation dictated by the delta velocity capability of the primary propulsion system. The first part of the study consisted of the effect of the upper limit orbit (perigee - 161 nmi, apogee - 2400 nmi) on the guidance and control system of the Gemini Agena Target. The second and major part of the study was concerned with the thermal response of the GATV operating in elliptical orbits having perigee at 161 nmi and apogees at altitudes up to 2200 nmi. The results showed that the vehicle could perform all required maneuvers but that for orbits having apogees greater than 560 nmi, close monitoring of temperatures of equipment in the aft rack would be required.

13. AGENA MANEUVER ACCURACY FOR THE GEMINI PROGRAM

The purpose of this study[14] was to evaluate the soundness of the MSC ground rules for maneuvering the GAT. The results presented errors in the orbital elements as well as position and velocity dispersions relative to the desired nominal orbits. These errors were within the accuracy limitations set up by the MSC ground rules.

14. GEMINI AGENA TARGET VEHICLE GUIDANCE AND PROPULSION SYSTEM ERRORS

This study[15, 16] presented PPS and SPS velocity magnitude errors as well as steady state and transient errors associated with pitch and yaw maneuvers (docked and undocked).

15. REVISED GEMINI/AGENA RENDEZVOUS DISPERSION ANALYSIS

This report[17] presented a general analysis of the possible dispersion resulting from GATV on-orbit maneuvers made for the Gemini rendezvous mission. Three basic GATV on-orbit maneuvers were considered: the orbital plane change maneuvers; the phasing maneuvers; and the retro maneuver into the rendezvous orbit.

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SECTION III. E

DEFINITIONS

AEDC	Arnold Engineering Development Center
BAC	Bell Aerosystems Corporation
C and C	Command and Communication Subsystem
Comsat	Communications Satellite
GATV	Gemini Agena Target Vehicle
LMSC	Lockheed Missiles and Space Company
MIT	Massachusetts Institute of Technology
MSC	Manned Spacecraft Center
OFPS	Oxidizer Feed Pressure Switch
OMPS	Oxidizer Manifold Pressure Switch
PC	Chamber Pressure
POSV	Pilot Operated Solenoid Valve
PPS	Primary Propulsion System
S/C	Spacecraft
SLV	Standard Launch Vehicle
SPS	Secondary Propulsion System

F. SYSTEMS PRODUCTION AND TEST

1. AT CONTRACTOR FACILITY

a. Standard Agena

The Standard Agena is a production line vehicle. It was supplied to the Gemini program as GFE through the standard DD-250 procedure. When this vehicle was transferred to the Gemini program, it had successfully completed all systems tests as required by contract and was considered to be a flight-ready vehicle. The testing which had brought the vehicle to this status was basically the same as that used to recertify the flight readiness of the Gemini Agena Target Vehicle (GATV) following its modification from the Standard Agena.

b. GATV

(1) General

The test program for the GATV was conducted on both the component and systems level. Testing was divided into (a) development testing to establish the functional characteristics of the components, (b) qualification testing to qualify the components for flight, (c) acceptance testing to ensure that the system was functioning properly, and (d) vehicle testing to gather information for systems evaluation and to ensure that the vehicle would meet the flight objectives.

(2) Manufacturing and Final Assembly Tests

During the manufacturing and final assembly test phase, electrical bonding tests were conducted to assure compliance with specifications. Electrical harness continuity and resistance checks were made. All harnesses were visually inspected for nicks, abrasions, unnecessary twists or other signs of damage, and for physical damage to the connectors and plugs. Automatic circuit analysis of electrical continuity, resistance, and insulation voltage breakdown was conducted. Alignments were performed and verified to assure compliance with the engineering drawings. Frequency, impedance, insertion loss, and voltage standing wave ratio tests were conducted after the transmission lines were installed in their final configuration. Guidance system validation tests were conducted before and after installation of the system in the vehicle. Leak checks were performed on the flight control pneumatic system, the hydraulic system, and the primary propulsion system.

(3) Vehicle Systems Test

The complete Vehicle (less the target docking adapter (TDA) and secondary propulsion system (SPS)) was subjected to a systems test similar to that conducted on all Standard Agena vehicles. Preliminary subsystem tests (Tasks I through VI) were run to verify the operation of the electrical guidance and control, and communication and control subsystems. The SLV-3 (Atlas)/GATV adapter and the GATV nose shroud were also checked during these tests. Finally, a simulated countdown and flight test was conducted to check the operation of the subsystem components according to the sequence of events established for ascent and orbit. During this test, electrical power was supplied by vehicle batteries and the vehicle hydraulics and control gas systems were pressurized. Orbit commands were fed into the vehicle both manually and by tape. Data were received both by landline and by telemetry link. Propulsion and pyrotechnic firings did not take place during these tests.

(4) Special Tests

Several special test programs were included in the Gemini Agena program in addition to the normal engine development program. These are discussed in brief in the following paragraphs.

(a) Propulsion Test Assembly Program. Because of the extensive changes to the propulsion system, a propulsion test assembly was fabricated and tests conducted at the Lockheed Santa Cruz Test Base (SCTB) facility. The propulsion test assembly consisted of a basic Agena structure with propellant pressurization, feed-and-load system, the primary propulsion system and two secondary propulsion system units attached to the aft rack. Propellant loading operations and hot firings of both propulsion systems were then conducted to establish the adequacy of the propellant loading system and the associated ground equipment for both the SPS and PPS, to demonstrate proper overall system operation, and to obtain engineering data on the system and the resulting environment.

The first test series acquired base-line performance on the PPS and SPS and subjected one SPS module to the dynamic and acoustic environment created by 55 seconds of PPS firing. The second series simulated a possible Gemini mission profile including multiple firings and various coast and burn times on both PPS and SPS units. Twelve firing cycles were completed on this series with maximum burn times of 184 seconds of the PPS, 167 seconds on the 16 lb thrust SPS engine and 50 seconds on the 200 lb thrust SPS engine. The third series involved a maximum number of starts and minimum-impulse firings on both PPS and SPS units. Sixteen PPS firings were conducted with times ranging from 2.0 to 145 seconds. Sixteen lb thrust SPS firings were conducted with times ranging from 5 to 70 seconds and nine 200 lb thrust firings were conducted with times ranging from 0.3 to 48 seconds.

(b) Captive Flight Test Program. The first GATV (5001) was taken to the Lockheed Santa Cruz Test Base for a captive flight test program designed to simulate actual on-orbit operation. The primary objective of the static hot fire test program was to verify operation capabilities of the Agena Target Vehicle under dynamic conditions during actual firing of the primary and secondary propulsion systems. The propulsion system firings were accomplished in accordance with a simulated mission profile sequence of events. This test included multiple firings of the primary and secondary propulsion systems. Subsequent to the final propulsion system firing, and prior to removal of the vehicle from the test stand, communications and control revalidation checks were accomplished to assure that vibration and acoustic levels reached during engine firings did not adversely affect other vehicle systems. In addition, the target docking adapter (TDA) built by McDonnell Aircraft Corporation was installed and tested as an integral system during this program. The secondary objective of this program was to demonstrate the integrity of the peculiar AGE, including the PCM telemetry ground station. Details of the preparation for the tests and the results can be found in the LMSC final report.[1]

(c) EMI/RFI Compatibility Test Program. Following completion of the captive test program at SCTB, the GATV (5001) was returned to Sunnyvale and placed in the anechoic chamber for EMI/RFI compatibility test. The objective of these tests was to demonstrate Agena electromagnetic compatibility in each of four configurations: (1) on pad at the launch site, (2) on ascent, (3) on orbit, (4) during predock, docking, and docked maneuvers with simulated r-f radiation expected from the Gemini spacecraft. The vehicle was suspended in the chamber as shown in Figure III. F-1. The spacecraft r-f generating devices were placed outside the chamber with the antennas inside the chamber and properly directed and attenuated to simulate spacecraft r-f radiation during orbital maneuvers. The testing program was divided into the following phases: self compatibility of the GATV r-f system; compatibility of the GATV

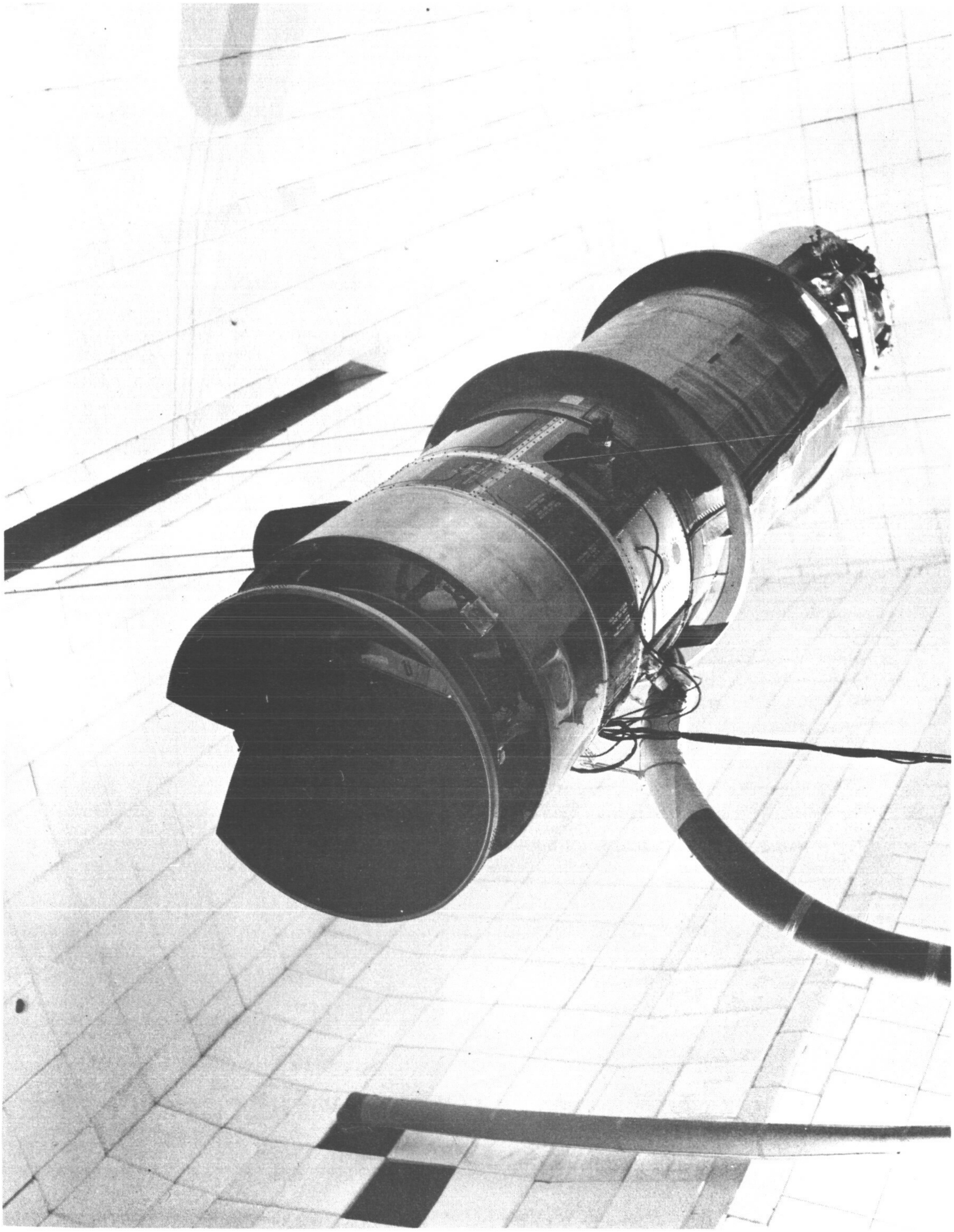

Figure III. F-1. Anechoic Chamber-5001.

and Gemini r-f systems; self compatibility of the GATV during high conducted transient noise conditions; compatibility of the GATV to simulated launch site r-f sources; and immunity of the GATV to static discharges. The latter test employed a capacitor charged to 50 volts and was based on the assumption that a static potential would exist between the GATV and Gemini spacecraft docking. Details of the tests and results are available in LMSC Report A 744002.[2]

2. VEHICLE TESTING AT ETR

Vehicle tests at ETR fulfilled four general requirements.

1) Assured the over-all integrity of the vehicle and associated hardware after shipment to ETR by receiving inspection, preliminary functionals, and fit checks.
2) Provided troubleshooting and retest, as required, to revalidate malfunctioning equipment and/or equipment added or changed out at ETR.
3) Verified operational readiness of all vehicle systems prior to erection and again after erection and mating to the Atlas vehicle on the pad.
4) Supported special tests such as Plan X and SLD.

The normal flow of vehicle tests began in Hangar "E" with the receiving inspection, fit checks, preliminary leak checks, power-on, and interface functionals.

The vehicle was then delivered to KSC on Merritt Island and positioned on the "Timber Tower" along with the Gemini spacecraft to support an r-f and functional mission compatibility test conducted by NASA.

Objectives which were accomplished in this test included:

1) Verified spacecraft/GATV hardline and r-f command capability.
2) Verified functionally satisfactory operation of spacecraft/GATV and MCC Houston/Cape command, data and communication links in proximity and docked modes of operation.
3) Provided astronaut familiarization in operating the spacecraft/GATV combination in a simulated rendezvous mission.

The vehicle was then returned to Hangar "E" for the Combined Interface Test (CIT). The purpose of this series of tests was to verify the operational readiness of all vehicle systems prior to erection and mating to the Atlas. Major emphasis was on exercising all vehicle functions on an end-to-end basis. Within the limits of practicability and equipment availability, test procedures and AGE utilized in CIT were identical to those employed in Final Systems Test at Sunnyvale.

Following the CIT, the vehicle underwent SPS and PPS functional checks. The SPS installation and checks at ETR were peculiar to the Gemini Program. A requirement[3] to repeat the PPS functional and leak checks 40 calendar days after initiation of the original checks established the on-pad time capability for GAATV after which demate would be required. This provided an on-pad capability of 14 days subsequent to the initially scheduled launch day.

After mating to the SLV-3 on Complex 14, the GAATV was subjected to a Joint Flight Acceptance Composite Test (J-FACT). This test was a combined check of all contractors, the Range, the vehicle, and AGE in a simulated countdown and flight. Propellants and high pressure gases were not loaded and the gantry was not removed for this test.

Next, the vehicle supported a simultaneous launch demonstration (SLD). This was a Gemini Program peculiar test which constituted a full dress rehearsal of the countdown. This test demonstrated the coordination required to conduct a single countdown on two vehicles, the GAATV located on

Complex 14, and the GLV-S/C on Complex 19. The MCC Houston/Cape link and ETR Range support were an integral part of the combined countdown and were satisfactorily demonstrated during this test. The actual launch countdown procedures were used to the point of committment with GAATV termination at T-18 seconds. The initial SLD, which was conducted with vehicle 5001, was a complete simulation including tanking of propellants. Subsequent tests merely simulated the tanking sequence.

The actual launch countdown represented the final systems test for the GATV. If all systems operated satisfactorily and within the redline parameter limits[3], the vehicle was committed for flight.

SECTION III. F

REFERENCES

1. Captive Flight Test Program (Final Report), Santa Cruz Facilities, LMSC 72235, Lockheed Missiles and Space Company, 24 March 1965.

2. Gemini Agena Target Vehicle Anechoic Chamber Test, LMSC A740002, Lockheed Missiles and Space Company, 2 April 1965.

3. Agena Target Vehicle Launch and Hold Limitations, Vehicle 5001, LMSC/A732564F, Lockheed Missiles and Space Company, October 1966.

DEFINITIONS

AGE	Aerospace Ground Equipment
CIT	Combined Interface Test
EMI	Electromagnetic Interference
ETR	Eastern Test Range
GAATV	Gemini Atlas Agena Target Vehicle
GATV	Gemini Agena Target Vehicle
GFE	Government Furnished Equipment
GLV	Gemini Launch Vehicle
J-FACT	Joint Flight Acceptance Composite Test
KSC	Kennedy Space Center
MCC	Mission Control Center
PCM	Pulse Code Modulation
PPS	Primary Propulsion System
RFI	Radio Frequency Interference
S/C	Spacecraft
SCTB	Santa Cruz Test Base, LMSC
SLD	Simultaneous Launch Demonstration
SLV	Standard Launch Vehicle
SPS	Secondary Propulsion System
TDA	Target Docking Adapter

G. GEMINI AGENA TARGET VEHICLE (GATV) HISTORY

1. MISSION

The mission of the Gemini Agena Target Vehicle (GATV) was to provide a stable platform with an on-orbit capability of commanded attitude changes and propulsion maneuvers for use by the Gemini Astronauts as a target vehicle for rendezvous and docking missions. Six GATV's were used for Gemini missions VI and VIII through XII. The history of these vehicles is shown in Figure III. G-1.

2. OBJECTIVES

The test objectives for the GATV remained the same throughout the Gemini program. The primary objectives were as follows:

a) To achieve a 161 nm circular orbit with an orbital inclination angle of 28.87 degrees. The SLV-3 (Atlas) was to place the GATV into a coast ellipse.

b) To maintain a stable attitude for a nominal 5-day active orbital life.

c) To receive, store, and execute ground-initiated commands and receive and execute spacecraft initiated commands for orbital control of guidance, propulsion, docking, and communications systems.

d) To maneuver into a new orbit in response to commands from the ground or the spacecraft.

e) To provide a safe environment during the rendezvous phases of the 5-day mission and to participate in docking and undocking operations with the Gemini spacecraft.

f) To provide a safe environment in the docked configuration while operating the primary and secondary propulsion systems.

3. FACTORY HISTORY

The manufacture and checkout of the GATV was accomplished entirely within the Lockheed Missiles and Space Company facilities at Sunnyvale, California. Standard Agena D vehicles were provided to the Gemini program by the normal delivery procedures. Vehicles were then disassembled and modified as necessary and the additional components required by the Gemini mission were added. Figure III. G-1 presents a summary of the manufacturing and test history of the six GATV's.

4. 5001 REFURBISHMENT[1]

The highly specialized activity in connection with Vehicle 5001 is worthy of note. The need to refurbish this vehicle was directly related to its history. The vehicle came into existence, during April 1964, as Standard Agena #AD71. After systems testing and delivery by DD-250, the vehicle underwent the modifications necessary for the Gemini Target Vehicle Configuration. During September 1964, these modifications were completed and vehicle systems level testing started and continued until July 1965. During this period over 1600 hours of vehicle power-on time was accumulated. Testing encompassed the following:

Sunnyvale Systems Test	- 4 months
Sunnyvale RFI	- 2 months
Santa Cruz Test Base Hot Fire	- 2 months
Launch Base Test Activity	- 2 months

After completion of the test activity at the Air Force Eastern Test Range, (ETR) the vehicle was retained as a backup for the first GATV launch (Vehicle 5002). Subsequent to this launch, the decision was made to return 5001 to Sunnyvale for refurbishment and updating to the latest GATV configuration for flight use later in the Program.

VEHICLE NUMBER	MANUFACTURING AND	
	PROGRAM MODIFICATION FINAL ASSEMBLY	
5001 AD–71, DD–250 30 April 64	30 April 1964/24 September 1964 418 In-process discrepancies 363 CART Inspection discrepancies CART to VST with 49 open items	VS Te sy SC (S An (Se VS 139 110 940 fro
5002 AD–82, DD–250 17 Dec 64	18 January 1965/18 May 1965 81 In-process discrepancies 204 CART Inspection discrepancies CART to VST with 54 open items 4 ECP's worked	18 48 101 124 171 36 9 S 2 E Tap TD TM Sne TM
5003 AD–108, DD–250 20 Jul 65	20 July 1965/14 October 1965 Received with 1 open discrepancy 76 In-process discrepancies 109 CART Inspection discrepancies CART to VST with 48 open items plus 75 ABNI items 31 ECP's worked	15 71 309 104 75 F 37 T 11 S 28 "Su 18 Tap FC PIV ind
5004 AD–109, DD–250 25 Oct 65	25 October 1965/26 January 1966 190 In-process discrepancies 102 CART Inspection discrepancies CART to VST with 37 open items plus 59 ABNI items Modified PPS installed 18 ECP's worked	27 J 18 D 34 In 32 P 117 17 T 9 ST 11 E
5005 AD–129, DD–250 2 Feb 66	2 February 1966/12 April 1966 Received with 4 open items 209 In-process discrepancies 176 CART Inspection discrepancies CART to VST with 31 open items plus 35 ABNI items Modified PPS installed 13 ECP's worked	12 A 16 D 52 In 22 P 88 P 8 Tr 7 ST 5 EC Com Stan
5006 AD–130, DD–250 22 Mar 66	22 March 1966/6 June 1966 Received with 3 open items 198 In-process discrepancies 133 CART Inspection discrepancies CART to VST with 43 open items plus 52 ABNI items Modified PPS installed 20 ECP's worked	6 Ju 4 De 40 28 F 51 F 10 12 7 E AM Yaw
5001R AD–71, DD–250 30 April 64	Returned from ETR 23 November 1965 for refurbishing IRAN activity began in December of 1965 826 Squawks written, only 150 required MRB action Approximately 40% component replacement and 35% electrical harness replacement Modified PPS installed 195 CART Inspection discrepancies CART to VST with 40 open items plus 40 ABNI items 41 ECP's worked	21 2 D 49 31 91 10 8 S 6 E

III (

CEPTANCE	POST ACCEPTANCE ACTIVITY (ETR)		REMARKS	ETR TEST HISTORY		
VAT ACTIVITY	DD-250 ITEMS	ADDITIONAL ITEMS		INTERFACE TESTS PLAN X	SYSTEMS TESTS (Hanger E)	
10 May 1965/27 May 1965 46 Discrepancies on data and component packages FACI conducted 21 May to 25 May – 818 discrepancies written DD-250 not signed due to lack-of-confidence in C & C system	Sent to ETR 28 May 1965 to support SLD activities 17 Design changes 11 Vehicle discrepancies 12 Open removals 1 UAS 6 Non-flt components 10 Unqual components	23 STI's 7 Engineering changes	SCTB- Mated with TDA and SPS On 20 January 1965, PPS fired 5 times, SPS I fired 5 times, and SPS II fired once – firings satisfactory Only major anomaly was command programmer clock jump due to EMI 62 Total data discrepancies Anechoic Chamber – EMI shown due to AGE, suitable protection incorporated into vehicle and AGE Previously planned EMI & RFI testing complete without incident	No testing with this vehicle at this time	29 May 1965/8 July 1965 Helium system contamination	8 Jul ACS
6 July 1965/23 July 1965 41 Discrepancies on data and component packages Limited physical inspection in conjunction with AFQA DD-250 23 July 1965	7 Open Engineering 1 Open operation 9 Vehicle discrepancies 4 Open removals 1 UAS 62 5001 FACI discrepancies 2 5002 FACI discrepancies 10 Shortages	4 Engineering changes 5 Unqual components 3 Non-flt components 23 STI's 9 ABNI items 9 ECP's	Command and Communication subsystem redesign efforts March – June Corning glass capacitor body cracking problem	23 August 1965/1 September 1965 Command programmer anomaly TM Sync loss Spurious MAP problem ERT reset anomaly	25 July 1965/23 August 1965 and 2 September 1965/30 September 1965 ACS N_2 Reg replaced Command programmer failure SPS ox fill valve leakage	1 Oc Prem
5 January 1966/18 January 1966 23 Discrepancies on data and component packages 24 Discrepancies on physical inspection 61 Configuration Management discrepancies DD-250 18 January 1966	5 Vehicle discrepancies 11 Open removals 4 Unqual components 4 Non-flt components 9 Config Mgt discrepancies 6 open Failure analyses 13 Shortages	2 Engineering changes 25 STI's 1 UAS 22 ABNI items 11 ECP's	GE Capacitor leakage problem Babcock relay getter problem Kemet capacitor problem Potter & Brumfield relay contamination problem Fairchild transistor leakage problem TI Integrated circuit welding problem Special pyro shock tests of aft rack equipment	26 January 1966/28 January 1966 25 Minor discrepancies 3 GATV problems	21 January 1966/26 January 1966 and 29 January 1966/28 February 1966 Turbine exhaust nozzle extension clearance problem FCEP Capacitor investigation Ox Lip Seal Press problem	1 Mar S-bar TDA
23 February 1966/11 March 1966 8 Data discrepancies 15 Component package discrepancies 7 Physical inspection discrepancies 11 Configuration Management discrepancies DD-250 11 Mar 1966	3 Open engineering 4 Vehicle discrepancies 2 Open removals 2 Shortages	Torquing UAS 15 STI's 3 Engineering changes 22 ABNI items 6 ECP's	One physical inspection discrepancy resulted in test and subsequent redesign of the hyd power package UDMH lines support TI integrated circuit problem Deutsch connector locking ring problem Dale resistor manufacturing problem Kemet capacitor problem	21 March 1966/23 March 1966 Cold solder joint in guidance J-box	14 March 1966/21 March 1966 and 24 March 1966/1 May 1966 D-Timer faulty switch CG mod worked Command receiver failure – no shutdown on command	2 Ma UHF to s on V
26 April 1966/14 May 1966 7 Data discrepancies 9 Component package discrepancies 4 Physical inspection discrepancies DD-250 14 May 1966	2 Open engineers 5 Vehicle discrepancies 5 Configuration Management squawks 4 Data package squawks 3 Shortages 6 Open ECP's 15 Open FEDR's 1 Unqual comp	Torquing UAS 13 STI's 4 Engineering changes 26 ABNI items	Pyro operated helium control valve orifice survey accomplished post acceptance on 5005, 5001 and 5006 valves also checked Command receiver survey for manufacturing anomaly Thrust valve cluster flow path contamination problem	1 June 1966/8 June 1966 No GATV problems	16 May 1966/1 June 1966 and 8 June 1966/1 July 1966 C-band xsponder frequency low, replaced, PPS Fuel pump assembly seal leakage, replaced Leaky fuel PIV, replaced	2 Ju Inte Ho
26 June 1966/13 July 1966 6 Data discrepancies 8 Component package discrepancies 8 Physical inspection discrepancies DD-250 13 July 1966	1 Vehicle discrepancy 2 Shortages 29 Open FEDR's 2 Unqual components 1 Non-flt component 5 Data pkg squawks	13 STI's 1 Engineering change 33 ABNI items 9 ECP's	Hughes quad diode problem Semcor capacitory leakage problem	22 July 1966/26 Jul 1966 No GATV problems	15 July 1966/21 July 1966 and 26 July 1966/20 August 1966 SPS pad misalignment problem C-band transponder sensitivity out-of-spec, HPP replaced, leak in reservoir seal	22 A Prog cap Spur
8 August 1966/2 September 1966 5 Data discrepancies 16 Component package discrepancies 11 Physical inspection discrepancies DD-250 2 September 1966	3 Open operations 1 Shortage 26 Open FEDR's 1 Non-flt component	22 STI's 1 UAS 27 ABNI items 24 ECP's	Thrust valve cluster flow path contamination Special Configuration Control Board, established to control vehicle configuration during refurbishment	16 September 1966/20 September 1966 No GATV problems	4 September 1966/16 September 1966 and 20 September 1966/22 October 1966 SPS bi-propellant valve bellow leak Shroud latching mechanism problem Shroud lanyard replaced due to kinks Performed engine align verification as a result of the 5006 flight anomalies	23 (IRP tra Hyd Sus Spu fail

	FLIGHT SUMMARY			REMARKS
SYSTEMS TESTS (Complex 14)	MISSION NUMBER	RESULTS	ANOMALIES	
1965/26 July 1965 ₂ Reg replaced	N/A	Maintained as a back-up for 5002 Subsequently returned to SV for refurbishment on 19 November 1965		Safety problem-SPS N_2 sphere pressure with personnel in area PPS removed and sent to BAC for modification
ber 1965/20 October 1965 ure ERT clockout anomaly	Gemini VI 25 October 1965 1500:04 GMT	Orbit not achieved due to GATV PPS malfunction during ascent		N_2 sphere pressure problem continued, waiver granted for 5002 only Excessive problem with out-of-spec MON for SPS
1966/9 March 1966 ksponder failure iamolies	Gemini VIII 16 March 1966 1500:03 GMT	Successful orbit Rendezvous with S/C on 5th orbit First docking in space 8–PPS/SPS–I and 2–SPS II undocked burns Maximum apogee 407 n mi	Yaw CG offset ACS Control gas regulator pressure drop during 1st yaw maneuver S/C problem terminated S/C mission early	V/M Counter contamination
966/10 May 1966 mmand Receiver replaced due ect manufacturing error found cle 5005 units	Gemini IX 17 May 1966 1515:03 GMT	Orbit not achieved due to SLV–3 pitch control malfunction	Excessive inverter temperature following SLV–3 problem	Out-of-spec MON Inverter temp probable cause – arcing at main umbilical due to Atlas exhaust gases IRFNA AGE Filter contaminated
1966/12 July 1966 ence between Complex 14 and n during V/M loading tests	Gemini X 18 July 1966 2039:46 GMT	Successful orbit Rendezvous on 4th orbit First docked PPS burn 3–PPS/SPS I & 3 SPS II docked burns Manned altitude record 413 n mi 2–PPS/SPS–I & 1–SPS II undocked burns Maximum apogee 751 n mi	+Y SPS II Indicated low Pc –Y SPS II Shutdown impulse above nominal on last 2 burns V/M With neg null torque gave greater Δ V than programmed Tape recorder problem following active mission	Tank pressure switch cleaning history investigation AGE power loss during CIT, no vehicle damage MON Spec H_2O content increased
ıst 1966/31 September 1966 ımer failure due to Semcor tor problem s programmer command problem	Gemini XI 12 September 1966 1305:02 GMT	Successful orbit Rendezvous on 1st S/C orbit 3–PPS/SPS I docked burns Manned altitude record 741 n mi 3–SPS II undocked burns Artificial gravity created by spinning tethered S/C, GATV configuration	Yaw heading error greater than expected during PPS burns Roll attitude excursion larger than expected during SPS II burns Time accumulator skip/L–band transponder failure; H/S pitch and roll error transients Tether interference	Spurious command problems determined to be design problem – eliminated by procedural change 1st launch attempt scrubbed due to GLV propellant leakage problem 2nd launch attempt scrubbed due to SLV–3 autopilot problem.
ıber 1966/1 November 1966 vorked to change-out questionable :tors ic system leak t oil contamination of PPS POSV s cmd due to TDA control panel	Gemini XII 11 November 1966 1907:59 GMT	Successful orbit Rendezvous with S/C on 4th orbit 2–SPS II Docked burns SPS I undocked burn prior to attempted PPS Number 2 Excessive use of ACS during docked maneuvers and EVA depleted gas supply	Momentary thrust decay during PPS ascent burn Turbine speed measurement anomaly Overspeed shutdown of PPS at start of burn Number 2 Excessive inverter temperature following loss of attitude control V/M Anomaly, instrument failure	V/M Electronics replaced by Extra Care Team due to excessive shelf life Investigation of Babcock relays for suspect overstress during LMSC receiving TI Transistors suspected of contamination

Figure III-G-1. Gemini Agena Target Vehicle Histories

[redacted]

Several major decisions had to be made prior to the start of the vehicle refurbishment activity:

- How far should the vehicle be disassembled in order to effect refurbishment?
- Who would manage and direct the refurbishment?
- What controls were required to ensure the final vehicle configuration?
- What requirements were necessary to ensure proper control of vehicle components during refurbishment?

It was decided to do the job adequately, the vehicle would have to be disassembled completely. This was necessary in order to (1) uncover and correct all structural corrosion; and (2) properly clean the main propellant tanks. The level of disassembly went to the point of a permanently attached assembly, in other words, riveted or welded. This decision to completely disassemble the vehicle proved to be sound. It not only revealed areas of corrosion unsuspected at the time of the vehicle's return, but established confidence that refurbishment would be thorough.

In order to ensure proper reassembly of the vehicle, it became apparent that the total configuration of the reassembled vehicle must be known as early as possible in the refurbishment activity. Attempts were made to establish a configuration utilizing existing CCN's and engineering documentation. This system, however, was not manageable because of the extensive research required to establish a configuration of detailed components. The final method of establishing vehicle configuration was by means of an integrated numerical parts list (INPL) covering both the program-peculiar and Standard Agena components of the Gemini Agena Target. It was also decided that special configuration inspections of the vehicle should be accomplished at certain major milestones. During these inspections, the vehicle was inspected for conformance to existing engineering and the INPL was used as a reference and updated. The inspection points chosen were as follows:

1. Completion of major structural assembly
2. Completion of wire harness installation
3. Completion of "black box" installations.

These inspections proved advantageous, aside from assuring the proper configuration of the vehicle. The inspections also allowed the paper system to have a complete audit and any errors corrected. It is of interest that throughout these inspections the hardware was never found in error, but invariably there were numerous errors in the various paper systems. The original premise is still valid that the "as-built" configuration must be known as soon as possible. Experience gained in this refurbishment activity indicated that the delivered vehicle configuration be known prior to disassembly, preferably by some means like an INPL.

Because of the nature of the refurbishment activity, it was decided that it would be most expeditiously handled by a Task Force mode of operation under the direction of the program office. Each participating organization assigned people to the Task Force for the duration of that organization's responsibility. Each organization also assigned a member to act as its team leader, responsible for that organization's effort.

All components were retested, updated, or repaired as necessary; however, the decision was made to use existing hydraulic or pneumatic tubing whenever possible. Subsequent test history proved this decision to be grossly in error. After reassembly of the vehicle, an additional 5 shifts were expended to complete the scheduled 4-shift leak checks. Upon delivery to the launch base and the ensuing high pressure checks, numerous leaks were found. This condition can only be attributable to the use of used tubing. Even though the tubing was reinspected for pressure and flare condition, no attempt was made to compare it to its original bend data. The vehicle, after being completely disassembled and reassembled was therefore not in exactly the same alignment as the tubing had previously experienced. This resulted in adjustments of the tubing to ensure proper fit.

5. ETR HISTORY

After the signing of the formal acceptance by the Air Force (Form DD-250), vehicles were transferred to the Eastern Test Range where additional confidence tests and spacecraft interface tests were conducted prior to launch.

6. LAUNCH

The GATV's were launched from Complex 14 at Cape Kennedy. Initial boost for the GATV was provided by the SLV-3 (Atlas). The SLV-3 placed the GATV into a coast ellipse and initiated the discrete start-up command for the GATV sequence timer. Final orbit attainment was achieved by firing the GATV primary propulsion system.

7. FLIGHT SUMMARY

The flight history of the GATV was impressive, but not entirely successful. Of the six GATV launches attempted, only four resulted in the attainment of the desired orbit by the GAT. The failure of the Gemini VI mission was attribúted to malfunction to the GATV primary propulsion system during the ascent burn. The failure of the Gemini IX mission was the result of an Atlas (SLV-3) guidance control malfunction.

On the brighter side, however, the four GATV's which attained orbit did so to near perfection and performed very well throughout the active portion of their respective missions. The significant events accomplished in the Gemini Program which the GATV made possible were:

a) G-VIII - first docking of two vehicles in space.
b) G-X - a new manned altitude record of 413 n mi.
c) G-XI - a new manned altitude record of 741 n mi.
d) G-XI - a creation of an artificial gravity by rotating the GATV/SC combination connected by a tether.

SECTION III. G

REFERENCES

1. Vehicle 5001 Refurbishment Summary Report, A 832599A, Lockheed Missiles and Space Company, 15 October 1966

2. Guidelines and Requirements Document for Gemini Atlas Agena Target Vehicle System, NASA DOD PR H-30247, H-50576, T-15482G, May 1965

3. Systems Test Objectives Gemini Atlas Agena Target Vehicle System, LMSC-A-727460 through -A727460E

SECTION III. G

DEFINITIONS

ABNI	Available but Not Installed
ACS	Attitude Control System
AFQA	Air Force Quality Assurance
AGE	Aerospace Ground Equipment
APDJB	Aft Power Distribution J-Box
CART	Conditions of Assembly for Release and Transfer
CG	Center of Gravity
CCN	Contract Change Notice
CMP	Configuration Management Plan
ECD	Engineering Change Directive
EMI	Electromagnetic Interference
ERT	Emergency Reset Timer
ERF	Engineering Reconciliation Form
FACI	First Article Configuration Inspection
FCEP	Flight Control Electronics Package
FCJB	Flight Control J-Box
FCLP	Flight Command Logic Package
FEDR	Failed Equipment Discrepancy Report
GMT	Greenwich Mean Time
H/S	Horizon Sensor
INPL	Integrated Numerical Parts List
IRP	Inertial Reference Package
MAP	Message Acceptance Pulse
MON	Mixed Oxides of Nitrogen
MRB	Materials Review Board
PIV	Propellant Isolation Valve
PPS	Primary Propulsion System
RFI	Radio Frequency Interference
SCTB	Santa Cruz Test Base, LMSC

SPS	Secondary Propulsion System
STI	Special Test Instruction
TI	Texas Instrument
TLM	Telemetry
UAS	Urgent Action Survey
UDMH	Unsymmetrical Dimethylhydrazine
UHF	Ultra-high Frequency
V/M	Velocity Meter
VST	Vehicle Systems Test

SECTION IV

GEMINI TARGET BOOSTER

A. INTRODUCTION

This portion of the report presents a summary of the historical background and the role played by the SLV-3 (Atlas) Target Booster in the Gemini Program. Since the standard mission of the SLV-3 is to place an Agena vehicle into a specified coast ellipse, the SLV-3 is discussed primarily in terms of aspects that were peculiar to the Gemini mission. The part it played in the backup Augmented Target Docking Adaptor (ATDA) mission - a different application in which the SLV-3 placed the target (ATDA) into a direct-ascent earth orbit - will also be discussed.

B. SELECTION OF THE SLV-3

The requirements of a booster for the Agena Target Vehicle included consideration of the Titan II. Factors that made the Atlas SLV-3 vehicle a desirable choice follow. The Atlas D (LV-3) vehicles:

- Were capable of meeting the target mission requirements.
- Were in quantity production and currently used as a standard workhorse booster for Air Force and NASA payloads.
- Had a demonstrated compatability with Agena vehicles, and were in routine use on USAF and NASA Programs.
- Constituted the most cost effective vehicle system.

Although these considerations in themselves were sufficient to determine the launch vehicle to be used, there was one additional fact that reinforced the decision. At that time, the Air Force had contracted with the Convair Division of General Dynamics for a new, improved version of the Atlas to be identified as the SLV-3. The modified Atlas was to incorporate a large number of mechanical and electrical design changes in an effort to eliminate the identifiable problem areas and failure modes experienced with the Atlas D (LV-3) space booster vehicle. To offset the weight increase due to these changes, the booster total engine thrust was increased by approximately 10 percent. The objective of the modifications was to achieve a vehicle reliability of 95 percent or better. The modifications[1] established a standard vehicle configuration for use with any space program. This standardized vehicle was expected to reduce cost and increase the rate of production and reliability. The Gemini Program would benefit from these improvements since there would be at least twelve SLV-3 launches before the first target mission.

C. SLV-3 PROGRAM MANAGEMENT

Concurrent with the decision to use the SLV-3 as the target booster, it was decided to use the management organization within AFSSD for the procurement and management of the Gemini target boosters. Figures IV.C-1 and IV.C-2 show the standard organizational relationships inherent in the SSD program management for SLV-3 type vehicles.

The Air Force SLV-3 Directorate (Figure IV.C-1) was used for the procurement and management of the SLV-3 for the Gemini Target Program; being a standard program office organization, it requires no description. Funding for procurement of SLV-3's comes from the various program offices using them, in this case, the Agena Target Vehicle Directorate. The relationship with NASA for implementation of SLV-3 purchase and launch support services is covered in Section III.A, which deals with Agena program management.

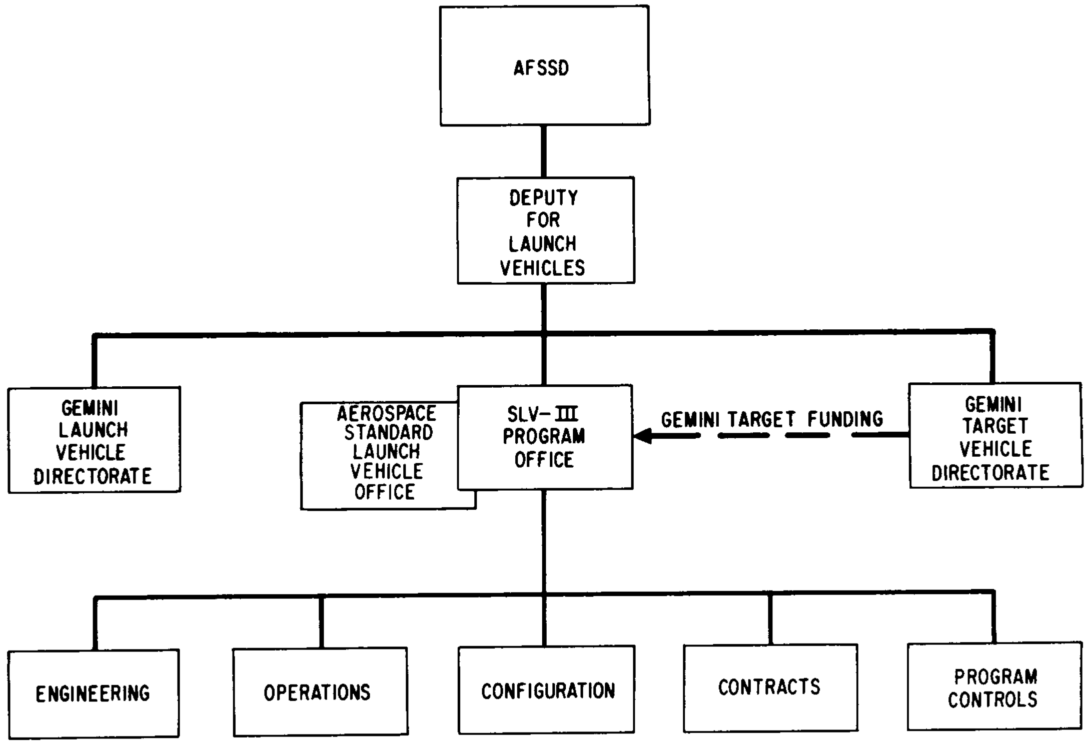

Figure IV.C-1. The Air Force SLV-3 Directorate.

The Aerospace Corporation provided technical surveillance over the SLV-3 in an advisory capacity to the Air Force. The Air Force and Aerospace program offices managing the over-all SLV-3 program provided combined support to the Gemini Program. Figure V. C-2 shows the contractor relationships in the SLV-3 program. In general, and because of the standard vehicle concept, none of the SLV-3 contractors established program offices specifically for Gemini. The Convair Division of General Dynamics, however, did have a special Gemini project engineer who had been given management authority. The mission planning and integration of the SLV-3 and associated activities were the responsibility of Lockheed except for the ATDA effort, which was completely GD/C's. ATDA management is discussed in D-4 of this section. The responsibilities of SSD/Lockheed are discussed in Section III. A.

The SLV-3 used as the Gemini Atlas Target Vehicle (GAATV) required a minimum of hardware changes to satisfy the Gemini program mission requirements. The program-peculiar modifications were handled by the same contractor engineering/management organizations that supported the over-all SLV-3 program, with SSD directing the effort and Aerospace supporting SSD, as required. The launch preparation was handled by the contractor's field organizations. Established field procedures were used and SSD (6555th Test Wing) directed the over-all effort at the ETR. Aerospace and vehicle contractor personnel participated in all of the GAATV design and technical reviews and in the flight readiness presentations made to the Flight Safety Review Board at Cape Kennedy prior to each launch. In addition, Aerospace Corporation in support of SSD, provided a special team of system experts at ETR, which reviewed the vehicle prelaunch test data to further assure vehicle flight readiness on all vehicles subsequent to the 2nd (5302) GAATV launch.

D. GEMINI TARGET BOOSTER HISTORICAL SUMMARY

1. VEHICLE CONFIGURATIONS

In accordance with the original program management decisions, the SLV-3's assigned to the Gemini program were standard vehicles selected from the production line by matching their scheduled delivery dates to the site need dates in support of Gemini. The Gemini target boosters were not consecutively produced, and the normal scheduled reliability improvement changes for SLV-3 were incorporated in the production articles as the need arose. These changes, which were minimal and of the product improvement type, generally resulted in no two Gemini SLV-3 booster vehicles having exactly the same configuration. However, all modifications were made solely to improve the over-all reliability of the SLV-3 family.

The standardization of design in the SLV-3 minimized the need for special modifications to accommodate the different programs. For most missions (using an Agena upper stage), the only significant difference between SLV-3 configurations is in the displacement gyro canister of the autopilot system. The Gemini program used identical canisters for all its flights except on the ATDA mission. Program-peculiar modifications consisted of gain changes in the displacement gyro to attain the required trajectories. In all other respects, the Gemini Target Boosters were standard SLV-3 vehicles.

The ATDA mission, which involved direct injection of the payload (ATDA) into orbit, required a number of special modifications to the SLV-3. They were adopted from the Mercury and Ballistic Missile Programs and are tabulated in Table IV. D-1.

2. OVER-ALL FLIGHT SUMMARY

The mission of the SLV-3 terminated at Agena target vehicle separation, and the only result significant in the program history was the success of the SLV-3 in satisfying mission injection parameters. Table IV. D-2 summarizes the results of the SLV-3 launches. The failure of SLV-3/5303 led directly to the decision to use the ATDA concept, and these two events are described in the following sections.

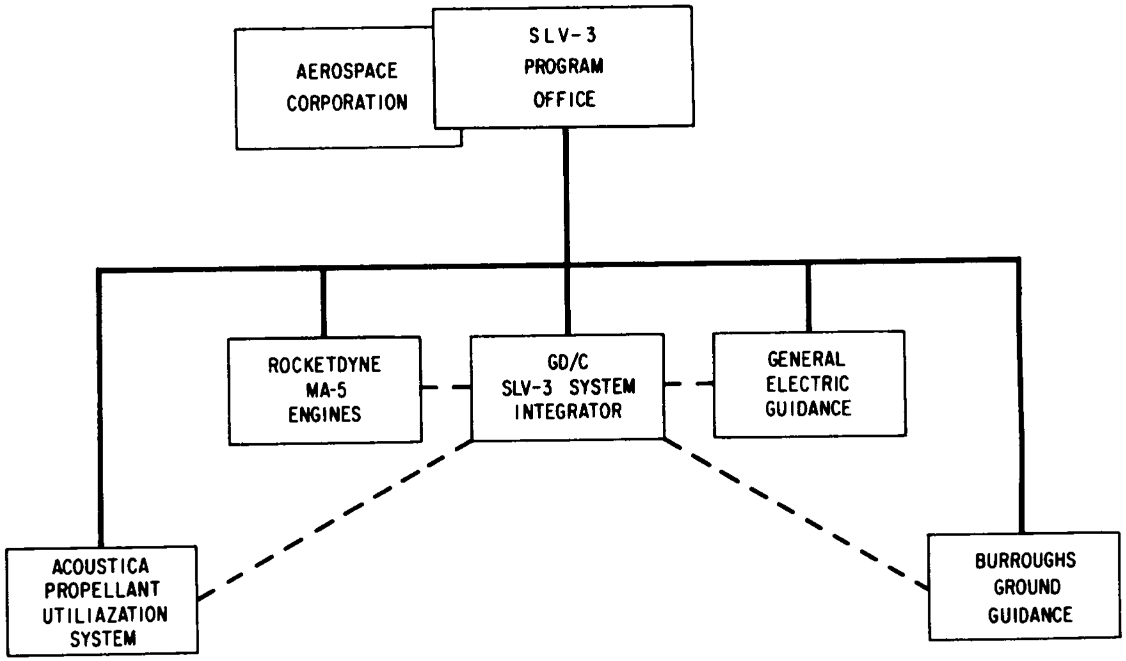

Figure IV. C-2. Contractor Relationships in the SLV Program.

TABLE IV. D-1

SUMMARY OF SLV-3/5304 FOR THE ATDA MISSION CONFIGURATION CHANGES

SYSTEM	MODIFICATION
Booster Adapter	The ATDA Kit modifies the Gemini target equipment adapter forward of the existing booster adapter extension as follows: 1. Removes the four existing jettison rails. 2. Removes the retro-rocket fairings and support bracketry. 3. Removes LMSC pyrotechnic ring and detonator housing at station 361.5 and bracketry in the destruct package area. 4. Removes the discrete destruct box and all electrical harnesses. 5. Vent holes in 4 access doors at station 436.6 are plugged up and the vent holes in the 4 access doors at station 391.05 are enlarged from 1-1/4-in. to 3-in. diameter.
Flight Control	1. Modifies the gyro canister rate and displacement gains to implement the requirements of the ATDA Bulletin No. 2-00197. 2. Changes the Pitch Attenuation Factor to 0.69. 3. Changes the staging backup accelerometer to implement a staging backup level of 6.1 g. 4. Modifies the programmer canister as follows: a. Revises the pitch program amplitude and time slots. b. Retimes the Subroutine 11 (SECO to Reset) to begin at 464 sec to accommodate longer sustainer burn time. c. Revises the VECO B/U and ISS B/U activation time. d. Revises the guidance enable time to L + 95 sec. e. Revises the integrator status by eliminating the null command during staging sequence. f. Revises staging enable time slot to L + 114 sec.

TABLE IV. D-1 (concluded)

SUMMARY OF SLV-3/5304 FOR THE ATDA MISSION CONFIGURATION CHANGES

SYSTEM	MODIFICATION
Flight Control	g. Revises time slot to provide liftoff filter switch at stg. + 0.1 sec and changes harness to provide liftoff filter switch signal from low-power Switch 26 instead of Switch 6.
	h. Revises displacement switch timing by adding low-power Switch 8 timing for displacement gain change at L + 90 sec.
	i. Revises rate switch timing by modifying low-power Switch 9 timing to achieve a rate gain change at L + 90 sec and SECO.
	5. Provides a servoamplifier assembly similar to the OAO configuration.
Pneumatic	1. Modifies the boiloff valve installation by readjustment of the valve controller for a 3.0 to 4.0 psig operating pressure and a larger bellows stroke.
	2. Modified the Lox regulators and relief valve installation for lowered Lox tank flight pressures (new range of 24.7 - 26.0 psig).
Electrical	1. Replaces the existing MOD III guidance antenna and associated waveguide with a Mercury booster guidance antenna and associated waveguide.
	2. Revises electrical harness by rerouting one wire to carry autopilot programmer Switch 8 output to the gyro package.
Telemetry & Instrumentation	1. Addition ot one landline measurement to the ATDA kit: S1325X, VECO Programmer Test Output.
	2. Deletion of the following measurements for the ATDA kit: Y41X Start D Timer Y1041X Start D Timer Y1040X Uncage LMSC Gyros S248X Release Payload S1248X Release Payload
Propulsion	Installs a larger-in-diameter and longer Rocketdyne sustainer lube oil tank (similar to the one used in the Mercury Program) and associated modified tubes per Rocketdyne ECP MA5-88. This sustainer lube oil tank has an increased capacity of approximately 1.7 gallons.
Range Safety	The ATDA kit deletes the interface requirement for upper stage. The harness at P/J106 is disconnected and tied back.

TABLE V.D-2

SLV-3 TARGET LAUNCH VEHICLE OVER-ALL FLIGHT SUMMARY

SLV S/N	Gemini Mission	Date of Launch	Sequence Among All Atlas Space Launches	SLV-3 Launch Sequence Number	Atlas Mission Objectives
5301[2]	VI	10-25-65	86	14	Successful (Agena Failure)
5302[3]	VIII	3-16-66	90	18	Successful
5303[4]	IX	5-17-66	96	23	Failure
5304[5]	IX-A (ATDA)	6-01-66	98	24	Successful
5305[6]	X	7-18-66	103	28	Successful
5306[7]	XI	9-12-66	107	32	Successful
5307[8]	XII	11-11-66	115	38	Successful

3. SLV-3 S/N 5303 FAILURE

Vehicle 5303 experienced the only SLV-3 flight failure in the Gemini program, and was one of the only two flight failures at that time in the SLV-3 program, which had an over-all success of better than 95 percent in 42 launches.

The flight of SLV-3/5303 proceeded normally until approximately 120 seconds (10 seconds prior to nominal Atlas booster engine cutoff). At this point, control of the B2 engine was lost. It gimballed rapidly to the full pitchdown position and remained against the stop. The autopilot loop, sensing the incorrect engine position, gave a full pitchup correction command; however, only the B1 engine responded. The B1 engine's response merely decreased the rate of angular acceleration to near zero. The initial rate of pitchdown angular rotation continued essentially unchanged and had reached a value of over 20 deg/sec. The vehicle continued out of control until after booster separation, at which time the sustainer engine regained control and stabilized the vehicle (150 sec after liftoff).

The vehicle performed a 216-deg pitchdown maneuver and pointed toward Cape Kennedy in a climbing attitude about 13 deg above the horizontal. During the maneuver, both radio guidance lock and pitch attitude reference were lost, resulting in the vehicle continuing on a new trajectory following normal autopilot programmer sequencing until vernier engine cutoff. Normal Agena separation was accomplished by programmer-timed backup command; however, because of the various abnormal conditions, the Agena did not attain orbit and fell into the Atlantic Ocean some 90 miles off the coast of Florida.

As a result of the flight failure, an intensive two-week failure analysis was undertaken. The conclusion of the investigation was that a short circuit occurred in the servo control circuit somewhere between the autopilot servo amplifier and the B2 pitch actuator. Corrective action taken for the following SLV-3 launches furnished protection against shorting of the control harnesses and consisted of the following measures:

- X-ray all soldered electrical connectors of less than eight pins. Include all servo valve and feedback transducer plugs.
- X-ray all servo valve torque motor wiring (Survey 38-66) to establish that pinching does not exist.
- All servo-actuator assemblies are to be subjected to a run-in of 4000 cycles of operation of 50 to 95 percent full stroke at 1/2 cycles/sec.
- Perform a manual flex test on the servo amplifier and excitation transformer connectors.

4. AUGMENTED TARGET DOCKING ADAPTOR (ATDA)

After the failure of the first target mission (Agena 5003), there was concern as to whether the Agena failure mode could be defined and corrected without causing a serious delay in the Gemini program. The possibility of another Agena failure would have a major impact on the program. Because of the potential problems, a parallel program, Augmented Target Docking Adapter (ATDA), was instituted by NASA to be used in the event of another Agena Target Vehicle failure. The SLV-3 was chosen as the boost vehicle since it was a standard production item that could be made available on relatively short notice.

a. Concept

McDonnell Aircraft Company proposed a modified target docking adapter mounted on a simplified airframe, which would utilize an attitude control system adapted from existing Gemini vehicle hardware. The ATDA would be a passive docking vehicle as compared to the active Agena target vehicle. This ATDA would offer an alternate capability for testing the docking functions in the event the current Agena problems were not resolved within a reasonable time. The ATDA could be built in a relatively short time since it utilized existing hardware and would be light enough so that an SLV-3 could inject it directly into orbit without requiring a second stage. The plan was to make the ATDA and the modified SLV-3 available in time to support the scheduled March 1966 mission. The ATDA, if not used, would then be kept in a state of readiness throughout the remainder of the Gemini program.

b. Management

The tight schedule available for the development and production of the ATDA dictated a highly responsive management system to insure no time delays in implementation. Within their organizations, the Air Force established a streamlined project management team and identified project officers in each key function to work as part of the ATDA management team. Personnel identified as part of this team were given authority and support to insure that there would be no delays in decision-making or contractual activity, which might adversely affect the program. In retrospect, the ATDA program stands out as an excellent example of the effectiveness of this type of project control for rapid response missions.

c. Implementation

From the time that NASA decided to implement the ATDA program, it was possible to have the affected contractors/organizations briefed and authorized to start work within a period of four working days. Convair, as the systems integrating contractor, performed the necessary vehicle integration studies, the open- and closed-loop trajectory analyses, and the identification and fabrication of modified SLV-3 hardware within 45 days from go-ahead. McDonnell Aircraft Co. was to provide the shroud and all engineering services associated with its installation and use. The guidance equations were modified by the TRW Systems Division and validated by GD/Convair during this period. GD/C studies, supplementing the McDonnell Company's efforts, identified a potential problem in the ATDA shroud in sufficient time for corrective recommendations to be implemented. The program schedule was met with all of the program-peculiar SLV-3 kits at Cape Kennedy by 1 March 1966. The SLV-3 booster vehicle kits included all changes in the site procedures necessary to launch the ATDA vehicle, as well as the complete hardware, instructions for installation, and drawing changes. As of 1 March 1966 everything associated with the SLV-3 booster vehicle was in readiness for application of the ATDA modifications to any Gemini SLV-3 for a launch within 18 days from go ahead.

d. Flight

The day after the failure of SLV-3 S/N 5303, NASA decided to launch the ATDA. Although the original schedule called for an 18-day turnaround, SLV-3/5304 was modified, erected at Complex 14, and prepared for launch within 14 days. The launch took place on 1 June, the 15th day following the 5303 failure. The flight of 5304 [5] was routinely precise and the ATDA was injected into orbit. The planned insertion conditions were met within close tolerances.

The ATDA[10] was not used for on-orbit docking maneuvers due to a failure of the nose shroud to separate.

E. CONCLUSIONS

The application of a standard launch vehicle, in this case, the SLV-3, which requires only minor program-peculiar modifications, proved very satisfactory for the Gemini target booster.

The injection parameters obtained during each mission, with the exception of 5303, were extremely accurate.

Considerable cost savings were realized, primarily because of the low cost of the SLV-3. Furthermore, since no extensive program modifications were required, it was not necessary to man the program office with additional personnel to manage the SLV-3.

The flexibility of a standard vehicle and the utilization of the kit concept for program-peculiar modifications were mandatory for the ATDA mission. Only through the kit concept was it possible to modify the SLV-3 so rapidly for such a mission.

SECTION IV

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SECTION IV

DEFINITIONS

AFSSD	Air Force Space Systems Division
ATDA	Augmented Target Docking Adapter
B1	Booster 1
B2	Booster 2
B/U	Backup
ECP	Engineering Change Proposal
g	Gravitation Acceleration
GD/C	General Dynamics/Convair
ISS	Integrated Start System
LMSC	Lockheed Missiles and Space Company
LV-3	Launch Vehicle 3
NASA	National Aeronautics and Space Administration
OAO	Orbiting Astronomical Observatory
SECO	Sustainer Engine Cutoff
SLV-3	Standard Launch Vehicle 3
SSD	Space Systems Division
TRW	Thompson Ramo Wooldridge
VECO	Vernier Engine Cutoff

V. CONCLUSIONS

As this report was prepared, conclusions specifically applying to a particular area or subject were developed and appear with the related material. The following conclusions represent general reflections of lessons learned and techniques used successfully during the course of the program.

MANAGEMENT

The Martin Company maintained an aggressive and technically capable reliability organization reporting <u>independently to program management</u>. This was extremely effective for Gemini and this particular organization contributed much to the company's success on Gemini.

Incentive contracting as used on Gemini was administered quite successfully from a management standpoint; however, the manned aspect of the program and the associated publicity provided a more effective incentive of a personal nature to the work forces in the factories and test areas. This motivating factor was encouraged by management and it added greatly to overall pride of workmanship.

TEST - ACCEPTANCE

Contractual control of ground testing must be tailored to meet the special situation. It is very easy for managing agencies to impose approval requirements on testing at an improper level. Loss of control can occur by approving contractor test plans at too general a level or by selecting a detail level which quickly results in an unmanageable administrative burden as the program becomes operational.

Interface agreements must be contractual to be effective and should contain test requirements and criteria as well as configuration.

The Airborne Systems Functional Test Stand (ASFTS) at Martin/Baltimore continually provided technical support to the GLV program. This extremely comprehensive and flexible engineering tool functioned in a "pilot plant" capacity throughout the program in areas of system integration, component and procedure development, and AGE compatibility. As basic studies were completed, this facility was used effectively to support failure analysis investigations. The existence of this engineering and test tool was unquestionably justifiable for Gemini and the concept could be highly advantageous for future programs.

A highly disciplined contractual hardware acceptance program is extremely important to a program where reliability is paramount and launch preparation time is critical. The question of incremental or end product acceptance of significant systems must be given careful evaluation by a customer agency before an approach is chosen.

TREND ANALYSIS

The observation of selected parameters monitored at regular intervals during normal factory and prelaunch testing can reveal useful performance trends. These trends may be used to advantage:

To express normal operating drift and test-to-test characteristics for baseline tolerances.

To identify incipient failures and allow timely replacement in the test cycle.

As a corrective action method in situations where suspect parts cannot be retrofitted because of cost, schedule, or accessibility.

A great deal of emphasis was placed on establishing nominal behavior patterns of Gemini critical components and systems.

RELIABILITY - QUALITY

The importance of a highly disciplined hardware oriented reliability program cannot be overemphasized. Realistic qualification testing and a relentless closed loop failure analysis effort must be part of the basic contracts and have the support of management at all times.

Periodic problem reporting and status meetings among all agencies involved in the program provided a good avenue for rapid, up-to-date information dissemination on real and potential problems with the end result of minimizing their effect on the program. The regularly scheduled

GPIS reviews and the GPIS accounting system as described in this report provided a very useful technique for assuring that all problems received the timely and proper amount of attention from all agencies concerned. The prelaunch technical reviews between Aerospace and the contractors for a final vehicle statusing prior to launch greatly reduced the possibility of an item "dropping through the crack."

Piece-part traceability is an extremely useful corrective action tool. It was not formally accepted in Gemini for economy reasons; however, on many occasions, the Gemini contractors at their own expense found ways to isolate suspect parts by long tedious research of manufacturing and procurement records.

Vendor control is by far the most annoying program problem. Many problems with Atlas, Agena, and GLV contractors were ultimately traced to vendor weaknesses. Extremely tight procurement specifications, effective source and receiving inspection techniques, vendor audits, and motivation briefings are just a few of the measures needed to control this problem. Any program with a high reliability objective must specifically attack this area aggressively with maximum management attention.

Following the successful conclusion of a program such as this, the highly disciplined techniques and mechanisms that were developed disappear rapidly as the people disperse. Every facet of the Gemini program was complex and challenging: the missions; the procurement, production, and testing; the launches; the communications; the geography; and especially the management. This report has related some of the techniques used in the hope that present and future programs might benefit.

The common goal of providing safe and predictable vehicles which would permit the astronauts maximum concentration on the scientific objectives of the Gemini program was achieved. There is no attempt at accolades in this report. Gemini was a highly successful engineering endeavor. Many of the reasons are contained herein, but the methodology demands competent contractor personnel and extremely capable management agencies. Such was the case on Gemini!

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VII. ENGINEERING CHANGE PROPOSAL (ECP) TABULATION

This section contains a tabulation of airborne ECP's. Section A lists those changes incorporated on the Gemini Launch Vehicle (Martin Company and Aerojet General). Section B lists the changes incorporated on the Gemini Agena Target Vehicle (Lockheed Missiles and Space Company).

Table VII. A-1. GLV Airborne Systems - ECP Tabulation

Martin Company, Baltimore

ECP	TITLE	LAUNCH VEHICLES											
		1	2	3	4	5	6	7	8	9	10	11	12
3	Modification of MDSTS Isolation Circuitry	x	x	x	x	x	x	x	x	x	x	x	x
12	Verification of Stage II 1.8 Sec. Ox. Shutdown TD Relay	x	x	x	x	x	x	x	x	x	x	x	x
19	Improved LV Capability for Propellant Temp. from 35 to 65 Deg. Fahrenheit	x	x	x	x	x	x	x	x	x	x	x	x
20	Rev. of Overvoltage Protection Devices	x	x	x	x	x	x	x	x	x	x	x	x
24	Rev. PCS Shutdown Sequence and Firing Mode Switch	x	x	x	x	x	x	x	x	x	x	x	x
28	Incorporation of Aux. Second Stage Engine Cut-Off Signal	x	x	x	x	x	x	x	x	x	x	x	x
32	APS and IPS Bus Voltage	x	x	x	x	x	x	x	x	x	x	x	x
35R1	FM/FM Telm. System for Trans. of Environmental Staging Data	x	x	x	x								
36R1	Telm. Pickup of IGS Output Signals		x	x	x	x	x	x	x	x	x	x	x
42	Revision to Pressure Switch Assembly	x	x	x	x	x	x	x	x	x	x	x	x
43	Mod. of Flight Term. System	x	x	x	x	x	x	x	x	x	x	x	x
44	Provision for Checkout to TARS Program	x	x	x	x	x	x	x	x	x	x	x	x
45	Spacecraft Abort Switch, Add to LV Test Console	x	x	x	x	x	x	x	x	x	x	x	x
46R1	Mod. of PDCS Hold/Kill Status Summary Circuit		x	x	x	x	x	x	x	x	x	x	x
47	Relocation of Stage II Engine Shutdown Monitor Point	x	x	x	x	x	x	x	x	x	x	x	x

Table VII. A-1 (Continued)

ECP	TITLE	LAUNCH VEHICLES											
		1	2	3	4	5	6	7	8	9	10	11	12
53	Mod. of Stage I Hydraulic System	x	x	x	x	x	x	x	x	x	x	x	x
54R1	Flight Control Sys. Hold Parameter Chg. Fail-Go to Fail-Safe		x	x	x	x	x	x	x	x	x	x	x
57	Safing of Stage I Destruct Initiator at Normal Staging	x	x	x	x	x	x	x	x	x	x	x	x
58	Make Provisions to Lock Out Flight Control Switchover Capability	x											
63R1	Engine Prevalve Redesign and Requalification	x	x	x	x	x	x	x	x	x	x	x	x
64	Reduction of Stage I Propellant Tank Mini-mum Ullage Volumes	x	x	x	x	x	x	x	x	x	x	x	x
66	Rev. to Stage II Oxidizer Feed Line Clamps (Compt. 2)	x											
67	PCM/FM Telm. Measurement Reg. for all GLV Flight	x	x	x	x	x	x	x	x	x	x	x	x
69	Relocation of Rate Gyro Isolation Resistors		x	x	x	x	x	x	x	x	x	x	x
70R1	Revise Scope of Qual. Test for Destruct Initiator	x	x	x	x	x	x	x	x	x	x	x	x
72R2	Installation of MOD III G Secondary Antenna		x	x	x	x	x	x	x	x	x	x	x
75	Limit Torquing Voltages to Rate Gyros During Programmed Test	x	x	x	x	x	x	x	x	x	x	x	x
79	Replace Polar Rings, Revise Cpl. on Fuel Fill Conn.	x	x	x	x	x	x	x	x	x	x	x	x
84	Installation of MK III Command Control Receiver (CP2850 at ETR only) (Replaced by ACI Command Control Receivers on LV 2 and subsequent)	x											

Table VII. A-1 (Continued)

ECP	TITLE	LAUNCH VEHICLES											
		1	2	3	4	5	6	7	8	9	10	11	12
86	Revision to Vent Topping Solenoid Valve	x	x	x	x	x	x	x	x	x	x	x	x
92	Relocate A/B Air Cond. Ext. Temp. Probe Measurement	x	x	x	x	x	x	x	x	x	x	x	x
93R2	Add Surge Suppression Devices to Stg. I Prop. Suction Line	x	x										
93R3	Add Surge Suppression Devices to Stg. I Prop Suction Line			x									
95R1	Shorting Rate Gyros During Static Firing	x	x	x	x	x	x	x	x	x	x	x	x
101	Verif. of Switchover Status and Gain State at Lift-Off	x	x	x	x	x	x	x	x	x	x	x	x
103	Change Autogenous Kill Parameter Check Time	x	x	x	x	x	x	x	x	x	x	x	x
110	Switchover to Secondary Autopilot Hyd. System GLV #1	x											
115	Pre Hydro-Helium Leak Test St. I and II Tanks (Test Accomplished by MMB, SSD disapproved ECP)			x									
115R1	Add Vapor Emission Tests for St. I and II Tanks				x	x							
116	Redesign Engine Exhaust Duct Life from 2 sec to 6 sec		x	x	x	x	x	x	x	x	x	x	x
117	Removal of Additional Rivets in Skin Splice			x	x	x	x	x	x	x	x	x	x
119R2	Replacement of Level Sensors	x	x	x	x	x	x	x	x	x	x	x	x
121	Rev. Method of Monitoring Staging on PCM/FM Telm.	x	x	x	x	x	x	x	x	x	x	x	x

Table VII. A-1 (Continued)

ECP	TITLE	LAUNCH VEHICLES											
		1	2	3	4	5	6	7	8	9	10	11	12
123	Mod. of Vent and Pressure Flex Hose	x	x	x	x	x	x	x	x	x	x	x	x
124	Manhole Cover Sealing			x	x	x	x	x	x	x	x	x	x
125	Adapter Cover Modification	x	x	x	x	x	x	x	x	x	x	x	x
126R2	Secondary Autopilot Gain Change Relay		x	x	x	x	x	x	x	x	x	x	x
127R1	Addition of ACI Command Destruct Receiver		x	x	x	x	x	x	x	x	x	x	x
129R1	Forward Skirt Ox. Tank Stage II/Spacecraft Sealing	x	x	x	x	x	x	x	x	x	x	x	x
130	Tandem Actuator Forgings and Internal Leakage	x	x	x	x	x	x	x	x	x	x	x	x
131	Rate Switch and Rate Gyros Contamination Control	x	x	x	x	x	x	x	x	x	x	x	x
132	Rev. to 115 VAC and 26 VAC Discriminator		x	x	x	x	x	x	x	x	x	x	x
134	TARS Gyro Fix. Redesigned Bellows	x	x	x	x	x	x	x	x	x	x	x	x
135	Inst. Low Range Axial Accel. to determine Stage II Thrust Tail-Off	x	x	x	x	x	x	x	x	x	x	x	x
137	Tandem Actuator Velocity Limits		x	x	x	x	x	x	x	x	x	x	x
138R2	Redesign of Oxidizer Feed Lines		x	x	x	x	x	x	x	x	x	x	x
141	Warning Plate for AGE Hyd. Power Supply	x	x	x	x	x	x	x	x	x	x	x	x
143	Telemetry Provisions of LV #2 and up		x	x	x	x	x	x	x	x	x	x	x
146	Incorporation of Shock Mounted Mod III-C Equipment	x											
150	Incorporate New Power Amp. Tube in T/M Transmitter	x	x	x	x	x	x	x	x	x	x	x	x

Table VII. A-1 (Continued)

ECP	TITLE	LAUNCH VEHICLES											
		1	2	3	4	5	6	7	8	9	10	11	12
151R1	Redesign of St. 1 & St. II Fuel Tank Internal Conduits (Stage I Only for GLV 4, Stage I & II for GLV 5 and subsequent)				x	x	x	x	x	x	x	x	x
153R1	Selected Items of GLV Weight Reduction Program					x	x	x	x	x	x	x	x
154R1	TARS Test Improvement and Stiction Test Requirement	x	x	x	x	x	x	x	x	x	x	x	x
155	Add Clamping for Staging Switch Bundle	x	x	x	x	x	x	x	x	x	x	x	x
156	Addition of Support Cable to Disconnect Cordage	x	x	x	x	x	x	x	x	x	x	x	x
159-3	Elim. Beat Frequency and Increase Demod. Dynamic Response	x	x	x	x	x	x	x	x	x	x	x	x
161	Mod of Gyros for Rate Switch Package	x	x	x	x	x	x	x	x	x	x	x	x
162	Reduce Stage II Fuel and Ox. Minimum Ullage Volumes	x	x	x	x	x	x	x	x	x	x	x	x
163	Sustainer and Roll Nozzle Actuator Cleaning Requirements	x	x	x	x	x	x	x	x	x	x	x	x
167	Radio Guidance System, Mounting Alternate B		x	x	x	x	x	x	x	x	x	x	x
170	Rework of No. 2 Truss Receptacle Screw Holes	x	x	x	x	x	x	x	x	x	x	x	x
171	Add Terminal-Quick Disconect for Grounding Missile to Transtainer	x	x	x	x	x	x	x	x	x	x	x	x
172	MDS Pressure Transducer Cleaning Requirements	x	x	x	x	x	x	x	x	x	x	x	x
173	Revise Mounting Hardware for Transistors in Power Amplifier	x	x	x	x	x	x	x	x	x	x	x	x

Table VII. A-1 (Continued)

ECP	TITLE	LAUNCH VEHICLES											
		1	2	3	4	5	6	7	8	9	10	11	12
174	Replace Lock Bolts in Truss Assembly	x	x	x	x	x	x	x	x	x	x	x	x
176	Revision to Pressure Transducer	x	x	x	x	x	x	x	x	x	x	x	x
178	Rev. for Longeron Supports & Incorporation of Denver DCN's Stage I Trailer	x	x	x	x	x	x	x	x	x	x	x	x
180	Mod. and Retest of Static Inverter	x	x	x	x	x	x	x	x	x	x	x	x
181R1	RGS/TARS Torquing Sensitivity	x	x	x	x	x	x	x	x	x	x	x	x
190R1	Replacement of Transistor & Diode Bridge in SMRD Module of MDP		x	x	x	x	x	x	x	x	x	x	x
191	Cap & Stow Umbilical End of A/B Coax Cable as Spare	x											
193	Addition of Redundant Propellant Level Sensors					x	x	x	x				
194R1	Incorp. New Filter in A/P, Stage II Slosh Condition	x	x	x	x	x	x	x	x	x	x	x	x
197	Revise Tank Breather Installation to Provide Clearance		x	x	x	x	x	x	x	x	x	x	x
200	Dynamic Response Testing GLV #1 at AMR	x											
201	Add Circuit Isolation for Staging Cont. Relays	x	x	x	x	x	x	x	x	x	x	x	x
202	Incorporate 4 Additional Telemetered Voltage Measurements for GE Mod III Guidance System	x											
203	Relocation of Filter Capacitor for Measurement 0355	x	x	x	x	x	x	x	x	x	x	x	x

Table VII. A-1 (Continued)

ECP	TITLE	LAUNCH VEHICLES											
		1	2	3	4	5	6	7	8	9	10	11	12
213	Mistram Test Set Primary Power Regulation		x	x	x	x	x	x	x	x	x	x	x
214R1	Tandem Actuator Redesign Position Switch	x	x	x	x	x	x	x	x	x	x	x	x
215	Suppression of RSP False Signals on 3 Telm. Channels		x	x	x	x	x	x	x	x	x	x	x
216R2	PCM/FM Telemetry Measurement Program - Add 16 Measurements		x	x									
221	Define Config. Identification for Launch Vehicles	x	x	x	x	x	x	x	x	x	x	x	x
222	Define Config. Identification for Launch Vehicles	x	x	x	x	x	x	x	x	x	x	x	x
223	Define Config. Identification for Launch Vehicles	x	x	x	x	x	x	x	x	x	x	x	x
228	Replace Primary 6-inch Aperture Antenna with 4-inch	x	x	x	x	x	x	x	x	x	x	x	x
230	Hydraulic Tube Assembly Modification	x	x	x	x	x	x	x	x	x	x	x	x
235	Revise Diode Mounting in Power Amplifier of MDP		x	x	x	x	x	x	x	x	x	x	x
239R1	Coupling Assembly Manual Self-Sealing Quick Disconnect			x	x	x	x	x	x	x	x	x	x
240	Remove 3. 5 Second Time Delay Relay	x											
241	Relocate Accelerometer	x	x	x	x	x	x	x	x	x	x	x	x
242	Revision to F/C Test Document 424-1112003 RGS/TARS	x	x	x	x	x	x	x	x	x	x	x	x
251-1R1	Rev. of EEI Control Plan to Reflect Expanded EEI Test Requirements	x	x										

Table VII. A-1 (Continued)

ECP	TITLE	LAUNCH VEHICLES											
		1	2	3	4	5	6	7	8	9	10	11	12
253R1	Revision to PS746000001 Pressure Transducer		x	x	x								
262R1	Removal of Level Sensor Shutdown Capability and Rev. TCPS Check	x	x	x	x	x	x	x	x	x	x	x	x
265	Redesign Stabilizing Line Support Bracket		x	x	x	x	x	x	x	x	x	x	x
267	Rev. to Flight Control/Guidance Redundancy Scheme			x	x	x	x	x	x	x	x	x	x
268	Removal of Destruct Initiator Auto Safety Feature	x	x	x	x	x	x	x	x	x	x	x	x
271R1	New Dampener Support Bracket to Clear AGC Pump Fins		x	x									
272	Increase Length of Launch Vehicle Ground Cable	x	x	x	x	x	x	x	x	x	x	x	x
277	Accum. Res. Transducer Change									x	x	x	x
278	Real Time Telm. Monitor Requirements - FCS					x	x	x	x	x	x	x	x
280R1	Replace Multiplexer Encoder Switch Capacitors to more Reliable Type	x	x	x	x	x	x	x	x	x	x	x	x
282	Relocate Stage I Destruct Initiator		x	x	x	x	x	x	x	x	x	x	x
284	Destruct Initiator Change	x											
286	Installation of Gemini Start Cartridge Conditioning Unit	x											
287	Add Tape Wrapping to Primacord Leads, Conduits, and Destruct Charges	x	x	x	x	x	x	x	x	x	x	x	x

Table VII. A-1 (Continued)

ECP	TITLE	LAUNCH VEHICLE											
		1	2	3	4	5	6	7	8	9	10	11	12
289R1	Transducer Change, Tandem Actuators		x	x	x	x	x	x	x	x	x	x	x
291	MDS/AGE Problem Dry Circuit Conditions		x	x	x	x	x	x	x	x	x	x	x
294	Confirmation of Baseline Ident. of GLV-5					x	x	x	x	x	x	x	x
299	Revise TARS Pitch Program Settings & Rates		x	x	x	x	x	x	x	x	x	x	x
304R1	PCM Encoder Modification		x	x	x	x	x	x	x	x	x	x	x
305	Implement S/C Interface Spec.; Remove 2 Wires from APS Bus		x	x	x	x	x	x	x	x	x	x	x
307	Redesign of Rotary Potentiometer for Fuel Accumulator		x	x									
312	Add Resistor to Martin Side of Interface to eliminate Oscillation		x	x	x								
320	Addition of LV & AGE Destruct Motor Driven Switch Control Interlock			x	x	x	x	x	x	x	x	x	x
321	Addition of Stage I Oxidizer Pump Suction Temperature Probe			x	x	x	x	x	x	x	x	x	x
324	Revise ESS/PVS Control Monitor Circuits to Prevent Cycling and Provide Positive Monitoring			x	x	x	x	x	x	x	x	x	x
325	Addition of Surge Suppression Devices in Stg. I Propellant Suction Lines (POGO)				x	x	x	x	x	x	x	x	x
327	Transient Voltage Limiting Resistence Change												
328	Dummy Propellant Prevalves, Stage I and II			x	x	x	x	x	x	x	x	x	x
329	Voltage Transients on L/V APS & IPS Power Busses		x	x	x	x	x	x	x	x	x	x	x

Table VII. A-1 (Continued)

ECP	TITLE	LAUNCH VEHICLES											
		1	2	3	4	5	6	7	8	9	10	11	12
330	Revision of MDS Rate Switch Package Settings		x	x	x	x	x	x	x	x	x	x	x
331	Revise Compensator Configuration of Stage I Hydraulic Pump		x	x	x	x	x	x	x	x	x	x	x
334	Application of Power to Destruct Battery Heater			x	x	x	x	x	x	x	x	x	x
337	Redundance for Starting the TARS Program Initiate Function		x	x	x	x	x	x	x	x	x	x	x
339	Provide Separate Grounds for Redundant Components		x	x	x	x	x	x	x	x	x	x	x
342R1	Revise FM/FM Tel. Patch to Provide more reliable data			x	x								
344	Add Resistance to Rate Beacon RF Power Telemetry Circuit		x	x	x	x	x	x	x	x	x	x	x
345	Selected Items of GLV Weight Reduction							x	x	x	x	x	x
349R1	Removal of Receiver Time Delay in Range Safety System Circuit		x										
349R2	Removal of Receiver Time Delay in Range Safety System Circuit			x	x	x	x	x	x	x	x	x	x
351	Transistor Recovery		x										
352	Increase Frequency Response on POGO Instrumentation Measurements		x	x									
353R2	Programming PCM/FM Telemetry Measurements, Package A			x	x	x	x	x	x	x	x	x	x
354	Connector Stowage Brackets Revision					x	x	x	x	x	x	x	x

Table VII. A-1 (Continued)

ECP	TITLE	LAUNCH VEHICLES											
		1	2	3	4	5	6	7	8	9	10	11	12
356	Add Shield to Level Sensors to Prevent False "Recovered" Readings		x										
357	Launch Shutdown Parameter Requirement for GLV #2 and Subsequent		x	x	x	x	x	x	x	x	x	x	x
358	New Clamp for Reservoir Accumulator Installation - Stage II			x	x	x	x	x	x	x	x	x	x
363	Adapter Package Transistor Recovery			x	x	x	x	x	x	x	x	x	x
364	Thermal Protection for Stage I Engine Driven Pumps		x	x	x	x	x	x	x	x	x	x	x
365	Rework and Replace Diodes - Autopilot			x	x	x	x	x	x	x	x	x	x
366	Rework and Replace Diodes - TARS Package			x	x	x	x	x	x	x	x	x	x
368	Engine Start Cartridge Conditioning, Stage I and II		x	x	x	x	x	x	x	x	x	x	x
370	Elimination of SECO Signal to Spacecraft During Countdown		x										
372	Prevention of Premature Staging Due to APS Battery Failure		x	x	x	x	x	x	x	x	x	x	x
373R1	Install Fire Protection Insulation over Selected Critical Components in Compartment 5			x	x	x	x	x	x	x	x	x	x
374	Martin Hardcoat Piston Shaft for POGO Fuel Accumulator		x	x									
379	A/P Channel Amplifier Capacitor Replacement			x	x	x	x	x	x	x	x	x	x

Table VII. A-1 (Continued)

ECP	TITLE	LAUNCH VEHICLES											
		1	2	3	4	5	6	7	8	9	10	11	12
380	Stage II Rate Gyro Signal Generator Capacitive Load Change			x	x	x	x	x	x	x	x	x	x
382	Incorporation of Redundant Shutdown System			x	x	x	x	x	x	x	x	x	x
383	GE Decoder Part Number Change		x	x	x	x	x	x	x	x	x	x	x
384	Conversion of Ground Prevalves to Flight Units								x	x	x	x	x
386	Reprogramming T/M Measurements, Package B		x	x	x								
387	Reprogramming T/M Measurements, Package C			x									
388	Reprogramming T/M Measurements, Package D			x	x								
390	Sustainer Pitch Actuator Rigging Length Revisions		x	x	x	x	x	x	x	x	x	x	x
392	Increase of Instrumentation Effectivity					x	x	x	x	x	x	x	x
393	Remove 40 Second Engine Shutdown Inhibit Timer		x	x	x	x	x	x	x	x	x	x	x
399	Change MDS to Monitor Fuel Injector Pressure		x	x	x	x	x	x	x	x	x	x	x
405R1	Change of Null Error at Zero		x	x	x	x	x	x	x	x	x	x	x
406	Redesign of Cover for Multiplexer Encoder		x	x	x	x	x	x	x	x	x	x	x
407	Redesign of MDP to Provide Insulation for Wire Bundle		x	x	x	x	x	x	x	x	x	x	x
409	Remote Changing Concept for Oxidizer Standpipe				x	x	x	x	x	x	x	x	x

Table VII. A-1 (Continued)

ECP	TITLE	LAUNCH VEHICLES											
		1	2	3	4	5	6	7	8	9	10	11	12
410	Launch Vehicle Grounding Strap		x	x	x	x	x	x	x	x	x	x	x
411	Maintain Power on LV Busses after T ± 15 Seconds		x	x	x	x	x	x	x	x	x	x	x
413	Revise Wiring Identification on MDS Pressure Transducer		x										
419	Modification of Microfuse Envelope		x	x	x	x	x	x	x	x	x	x	x
422	GLV TCS Console Mod. for T.C. Abort Command Capability		x	x	x	x	x	x	x	x	x	x	x
424	MDS Wiring Changes			x	x	x	x	x	x	x	x	x	x
425	Revision to Fuel Level Sensors				x	x	x	x	x	x	x	x	x
425R1	Revision to Fuel Level Sensors				x	x	x	x	x	x	x	x	x
426	Revise Wiring Identification Process MDS Pressure Transducer			x	x	x	x	x	x	x	x	x	x
428	Removal of Elapsed Time Indicator from the Rate Gyro System*												
429	Gyro Motor Bearing Pre-Load Process Control & Measurements*												
442	Replacement of Relays and Addition of a Redundant Diode in Switchover Module			x	x	x	x	x	x	x	x	x	x
446	Stage I Oxidizer Feedline Bellows Assy.						x	x	x	x	x	x	x
447	Redesign of Breather Sets, Propellant Tanks					x	x	x	x	x	x	x	x
451R1	Tandem Actuator Servo Design Change		x	x	x	x	x	x	x	x	x	x	x
454	Revise Stage I Actuator Null Shutdown Parameter		x	x	x	x	x	x	x	x	x	x	x

*See incorporation status, end of table.

Table VII. A-1 (Continued)

ECP	TITLE	LAUNCH VEHICLES											
		1	2	3	4	5	6	7	8	9	10	11	12
455	Add Insulation to Start Cartridge Ducting				x	x	x	x	x	x	x	x	x
461	Provide Positive Mechanical Attachment for Potentiometer		x	x	x	x	x	x	x	x	x	x	x
464	Increase Actuator Sampling Rates, Instrumentation Change		x										
468	Transtainer Modification			x									
469R1	Removal of Spare Instrumentation Wires								x	x	x	x	x
470	TCPS Circuit Modification			x	x	x	x	x	x	x	x	x	x
471	Vibration Isolation of A/P Adapter Switch-over Module			x	x	x	x	x	x	x	x	x	x
472R3	Add Tandem Flowmeters						x	x	x	x	x	x	x
473	Remove Thermal Coating Stage II Oxidizer Tank Forward Skirt				x	x	x	x	x	x	x	x	x
474	Add Heat Shroud to Fuel Accumulator - POGO			x	x	x	x	x	x	x	x	x	x
475R2	Revision to Propellant Fuel and Oxidizer Level Sensor			x	x	x	x	x	x	x	x	x	x
479	Revise S/C Abort Indication Circuitry			x	x	x	x	x	x	x	x	x	x
485	Redevelop Hydraulic Tube Due to Inter. with Autogenous Tube					x	x	x	x	x	x	x	x
495	Revise TARS Pitch Program Rates				x	x	x	x	x	x	x	x	x
496	Installation of Flashing Lights, Stage II				x								
505	Revise Sustainer Actuator Bushings and Spacer				x	x	x	x	x	x	x	x	x

Table VII. A-1 (Continued)

ECP	TITLE	LAUNCH VEHICLES											
		1	2	3	4	5	6	7	8	9	10	11	12
507	Revise Secondary Pressure Switch Setting, Reservoir Accum.					x	x	x	x	x	x	x	x
508	Removal of IPS DC Monitor as a Shutdown Parameter				x	x	x	x	x	x	x	x	x
512	Revise 140-Second Time Delay Relay				x	x	x	x	x	x	x	x	x
513	Tandem Actuator Position Transducer Revision								x	x	x	x	x
514	Landline Monitoring of Destruct Battery Voltage					x	x	x	x	x	x	x	x
517R2	FCS Gain Change Time Revision						x	x					
525	Revise Prevalve Open Times During Launch Countdown					x	x	x	x	x	x	x	x
526	Delete Stg. I Eng. TCPS - Incorp. in MDTCPS									x	x	x	x
534	Flashing Light - Stage II Installation							x					
537	Replace Tube Sockets in PCM/FM Power Amplifiers						x	x	x	x	x	x	x
539	Add Check Valve to Hydraulic Motor Pump Discharge Port						x	x	x	x	x	x	x
541	Station 923 Beef-Up									x	x	x	x
542	Revise FCSTS Functional C/O Capability						x	x	x	x	x	x	x
543	Revision to Stg. I - II Fuel Ox. Flowmeter								x	x	x	x	x
546R1	TARS Gyro Temperature Monitor						x	x	x	x	x	x	x
550	Add Hole in POGO Bracket								x	x	x	x	x
554	Bushing for Oversized Clevis Hole									x			

Table VII. A-1 (Continued)

ECP	TITLE	LAUNCH VEHICLES 1	2	3	4	5	6	7	8	9	10	11	12
560R1	Add Cutout and Lockwire to Disconnects						x		x	x	x	x	x
561R1	Disconnect Modification								x	x	x	x	x
564	T/M Antenna Change								x	x	x	x	x
566	Modification of Holdfires A8 and A10								x	x	x	x	x
568	Bushing for Oversized Clevis Hole											x	
569	PSV Drain Line Revision								x	x	x	x	x
573	Back-up Signal for Holdfire Turn-Off									x	x	x	x
576R1	System Test Selector Valve										x	x	x
577	Destruct Initiator Rework										x	x	x

ECP's 428 and 429 were optional for GLV-2 and subsequent. The incorporation status of these ECP's is as follows:

COMPONENT	PART NUMBER	ECP	GLV 2	GLV 3	GLV 4	GLV 5	GLV 6	GLV 7	GLV 8	GLV 9	GLV 10	GLV 11	GLV 12
Autopilots	424-7562200-079		X	X									
	424-7562200-099	428				X							
	424-7562200-109	429			X		X	X	X	X	X	X	X
Stage I Rate Gyros	PS962000001-049		X	X	X								
	PS962000001-109	428				X							
	PS962000001-129	429					X	X	X	X	X	X	X

Table VII. A-1. GLV Airborne Systems - ECP Tabulation
Aerojet-General Corporation, Sacramento

ECP	TITLE	STAGE I ENGINES														
		1001	1002	1003	1004	1005	1006	1007	1008	1009	1010	1011	1012	1013	1014	1015
AJ-009	Stage I Gearbox Modification	-*	R*	P*	P	P	P	P	P	P	P	P	P	P	P	P
AJ-012	FPDPS Qual. Test Failure	R	R	P	P	P	P	P	P	P	P	P	P	P	P	P
AJ-013	PSV Mod. - Incorporating New Solenoid Valve, Sleeve and Spool Assembly	R	R	P	P	P	P	P	P	P	P	P	P	P	P	P
AJ-016 R2	Replacement of Autogenous Restrictors (Back pressure)	R	-	-	-	-	-	-	-	-	-	-	-	-	-	-
AJ-017 R1	Clamping of PSV Conduit, Control Harness Clamping	R	R	P	P	P	P	P	P	P	P	P	P	P	P	P
AJ-018	Thrust Chamber Assembly Flange Bolt Mod. - Oxidizer and Fuel TCV Flange	R	R	P	P	P	P	P	P	P	P	P	P	P	P	P
AJ-023 R1	Thrust Chamber Valve Assembly Mod. "87"	R	R	P	P	P	P	P	P	P	P	P	P	P	P	P
AJ-029 R4	Fuel Suction Line Modification (POGO) Frequency Converter	R	-	-	-	-	-	-	-	-	-	-	-	-	-	-
AJ-031	Mod. of TCV Assemblies by Replacement of Clevises and Actuation Arms	-	R	P	P	P	P	P	P	P	P	P	P	P	P	P
AJ-033	Addition of Turbine Exhaust Stack Closure	-	R	P	P	P	P	P	P	P	P	P	P	P	P	P
AJ-039 R2	Replacement of Frame Bolt at Frame 5th Point	R	R	P	P	P	P	P	P	P	P	P	P	P	P	P
AJ-043	Modification of GGA Fuel Check Valve	-	R	R	R	P	P	P	P	P	P	P	P	P	P	P
AJ-044 R1	Oxidizer Suction Line Modification (Short)	R	-	-	-	-	-	-	-	-	-	-	-	-	-	-
AJ-045	Mod. of Autogenous Tank Pressure & TLM Instrumentation Installation	R	P	P	P	P	P	P	P	P	P	P	P	P	P	P
AJ-078	Mod. of MDS Cable Assemblies to Incorporate Smaller O. D. Hose Assemblies	-	S*	S	S	P	P	P	P	P	P	P	P	P	P	P
AJ-082	Mod. of PMDS Cable Assemblies to Incorporate Smaller O. D. Assemblies	-	S	S	S	P	P	P	P	P	P	P	P	P	P	P
AJ-090	Replacement of Telemetry Instrumentation Dust Covers	R	R	P	P	P	P	P	P	P	P	P	P	P	P	P
AJ-083 R1	Mod. of TPA to Permit Installation of POGO Hardware	-	R	-	-	P	P	P	P	P	P	P	P	P	P	P
AJ-101	Mod. of Fuel Accumulators & Support Bracket Installation	-	-	R	R	-	-	-	-	-	-	-	-	-	-	-
AJ-102	Instrumentation Pressure Transducer (Statham added as an approved Vendor)	-	S	S	S	P	P	P	P	P	P	P	P	P	P	P
AJ-035	Replacement of Frequency to DC Converter	R	R	R	P	P	P	P	P	P	P	P	P	P	P	P
AJ-117	First Stage Engine Reidentification per TLM Requirements	-	-	-	-	P	P	P	P	P	P	P	P	P	P	P
AJ-118	TLM Kit Switch GLV-1002 to GLV-1005	-	R	-	-	P	-	-	-	-	-	-	-	-	-	-
AJ-057 R1	Replacement of Turbine Interstage Labyrinth Seal with one of Improved Material	-	R	R	R	R	P	P	P	P	P	P	P	P	P	P
AJ-094 R1	Modification of Fuel Accumulator Support Bracket Installation	-	-	-	-	R	-	R	-	-	-	-	-	-	-	-
AJ-086	Replacement of Solenoid Mounting Screws on Pressure Sequence Valve	-	R	R	R	R	R	P	P	P	P	P	P	P	P	P
AJ-122	Replacement of Telemetry Instrumentation Interface Bracket	-	-	R	R	R	-	-	-	-	-	-	-	-	-	-
AJ-144	Stage I SSC Thermal Conditioner	-	R	R	R	R	R	R	R	R	R	R	R	R	R	R
AJ-146		-	R	-	-	-	R	-	R	R	R	R	R	R	R	R
AJ-149	Addition of Fire Protection Insulation to the YLR87-AJ-7 Rocket Engine	-	R	-	R	R	R	R	R	-	-	-	-	-	-	-
AJ-149R	P_C5_C Transmitter Add. Insulating Wrap	-	-	-	-	-	-	-	-	R	R	R	R	R	R	R
AJ-162	Replacement of Telemetry Inst. Interface Bracket	-	R	-	-	-	R	R	R							
AJ-165	Replacement of Hot Gas Cooler Outlet Tube Assembly	-	R	-	-	S	R	R	R	R	R	R	R	R	R	R
AJ-166	Replacement of Marman Clamps	-	S	-	-	S	S	S	S	S	P	P	P	P	P	P
AJ-175R	Lockwire Gas Generator Oxidizer Line "B" Nut and POBTV Pressure Cap. (Stage II)	-	R	-	-	-	R	R	R	R	R	R	R	R	R	R
AJ-181	Replacement of the Stage I Sub Assy I Oxidizer Pressure Discharge (POD) Boss	-*	R*	-	-	-	R	-	R	R	R	R	R	R	R	R
AJ-176	Replacement of Lube Oil Cooler Rigid Inlet and Outlet Tube Assy with Flex Metal Hose Assy	-	-	-	-	-	-	-	R	R	R	R	R	R	R	R
AJ-183R	Replacement of Oxidizer Autogenous Back Pressure Restrictor (OPBPO)	-	-	-	-	-	R	-	R	R	R	R	R	R	R	R
AJ-182	Integration of TCPS Functions with MDTCPS	-	-	-	-	-	-	-	-	R	R	R	R	R	R	R
AJ-184R	Replacement of Oxidizer and Fuel Thrust Chamber Valve "Thru" Bolts (Stage I)	-	-	-	-	-	-	-	S*	R	S	R	R	R	R	R

*R = Retrofit
P = Production line installation
- = Non-applicable serial numbers
S = Spares

Table VII.A-2. GLV Airborne Systems - ECP Tabulation
Aerojet-General Corporation, Sacramento

ECP	TITLE	STAGE II ENGINES														
		2001	2002	2003	2004	2005	2006	2007	2008	2009	2010	2011	2012	2013	2014	2015
AJ-007	Replacement of Turbine Manifold	R*	P*	P	P	P	P	P	P	P	P	P	P	P	P	P
AJ-014	Mod. & Replacement of Pressure Sequence Valve	R	R	P	P	P	P	P	P	P	P	P	P	P	P	P
AJ-015	Add. of "91" Roll Control Nozzle Burst Diaphragm	R	R	P	P	P	P	P	P	P	P	P	P	P	P	P
AJ-019	TCA Flange Bolt Modification	R	R	P	P	P	P	P	P	P	P	P	P	P	P	P
AJ-021 R2	Rerouting of Oxidizer Bootstrap Line & Clamp Changes	R	R	P	P	P	P	P	P	P	P	P	P	P	P	P
AJ-024 R1	Thrust Chamber Valve Assembly Mod. "91"	R	R	P	P	P	P	P	P	P	P	P	P	P	P	P
AJ-028	Turbine Seal Assembly Replacement (Design Chg. of TPA)	R	R	P	P	P	P	P	P	P	P	P	P	P	P	P
AJ-032	Mod. of TCV Assemblies by Replacement of Clevises and Actuation Arms	-	R	P	P	P	P	P	P	P	P	P	P	P	P	P
AJ-036	Replacement of Frequency to DC Converter	R	R	P	P	P	P	P	P	P	P	P	P	P	P	P
AJ-040 R1	TLM-JB-Mounting Bracket	R	R	P	P	P	P	P	P	P	P	P	P	P	P	P
AJ-048	Replacement of MDS and Telemetry Pressure Tube Assemblies	-*	R	R	R	R	P	P	P	P	P	P	P	P	P	P
AJ-048-1	Mod. of Adapter Box to provide for engine MDS Switch Relocation															
AJ-050	PSV Override Solenoid Monitoring Rocket Engine Test Set Mod.	R	R	P	P	P	P	P	P	P	P	P	P	P	P	P
AJ-055	Incorporation of Ultrasonic Inspection and Magnetic Particle Inspection Engine Requirements for Frame Assembly	-	S*	S	S	S	S	S	P	P	P	P	P	P	P	P
AJ-059 R1	Mod. of TPA Turbine Kit by Replacing Safety Wire with Lock Ring	-	S	S	S	S	S	S	S	P	P	P	P	P	P	P
AJ-065	Lube Oil Pump Redesign	R	R	R	P	P	P	P	P	P	P	P	P	P	P	P
AJ-081	Replacement of Malfunction Detection System Cable Assemblies	-	S	S	S	P	P	P	P	P	P	P	P	P	P	P
AJ-087	Replacement of Solenoid Mounting Screws on Pressure Sequence Valve	-	R	R	R	R	R	P	P	P	P	P	P	P	P	P
AJ-091	Replacement of Telemetry Instrumentation Dust Covers	R	R	P	P	P	P	P	P	P	P	P	P	P	P	P
AJ-103	Instrumentation Change (Statham Transducer)	-	S	S	S	P	P	P	P	P	P	P	P	P	P	P
AJ-116	Replacement of Nuts and Washers on Frame Rod End Connectors	-	S	S	S	R	P	P	P	P	P	P	P	P	P	P
AJ-119	Second Stage Engine Reidentification per TLM Requirement	-	-	-	-	P	P	P	P	P	P	P	P	P	P	P
AJ-123	Modification of Telemetry Instrumentation Installation	-	R	R	R	-	-	-	-	-	-	-	-	-	-	-
AJ-141 R1	Stage II Redundant Shutdown	-	-	R	R	R	R	R	R	P	P	P	P	P	P	P
AJ-142	Mod. of Telemetry Instrumentation Interface Bracket	-	R	R	R	R	P	P	P	P	P	P	P	P	P	P
AJ-145	Stage II SSC Thermal Conditioner	-	R	R	R	R	R	R	R	R	R	R	R	R	R	R
AJ-154R2-C	Incorporation of the YLR91-AJ-7 GEMSIP Injector into Gemini Engines	-	-	-	-	-	R	-	-	P	P	P	P	P	P	P
AJ-164	Modification of Clamping of Oxidizer and Fuel Gas Generator Supply Line Installation	-	-	R	R	R	R	R	R	P	P	P	P	P	P	P
AJ-167	Replacement of Marman Clamps	-	-	S	-	S	S	S	S	P	P	P	P	P	P	P
AJ-174R	Lockwire Gas Generator Oxidizer Line "B" Nut and POBTV Pressure Cap. (Stage II)	-	-	-	-	R	R	R	R	R	R	R	R	R	R	R
AJ-185R	Replacement of Oxidizer anf Fuel Thrust Chamber Valve "Thru" Bolts (Stage II)	-	-	-	-	-	R	-	-	S	S	R	R	R	R	R

*R = Retrofit
P = Production line installation
- = Non-applicable serial numbers
S = Spares

Table VII. B-1. GATV Airborne Systems - ECP Tabulation
Lockheed Missiles and Space Company, Sunnyvale

ECP	TITLE	TARGET VEHICLES					
		5001	5002	5003	5004	5005	5006
10	Programmer - Type VI						
11	Hi-Rel. Parts - F/C Sequence Timer			P*	P	P	P
12C2	GATV Arm/Stop Circuitry Change	R*	R	P	P	P	P
15	Running Lights	R	R	P	P	P	P
16	L Band Xponder Cover Jettison Time Change	R	R	P	P	P	P
17	Hi-Rel Parts F/C J-Box Doc. Change			P	P	P	P
18	Hi-Rel Parts Guidance J-Box			P	P	P	P
19	Reloc. Contact Pin Bracket - Horizon Sensor Fairing Temp. Sen.			P	P	P	P
22	Destruct Shield	R	R	P			
30	Bonding Acquisition Light Circuit Shield	R		R	P	P	P
31R1	Flight Control Electronics Mod.	R		P	P	P	P
32	Battery Temp. Monitor	R	R	P	P	P	P
33	Programmer XVI - Design Improvement	R		R	P	P	P
37cc	Horizontal Sensor Bias Angle	R					
38R1	MAP Generator Grounding	R	P	P	P	P	P
39	Message Logic Amp. Mod.						

*P = Production line installation
R = Retrofit

Table VII. B-1 (Continued)

ECP	TITLE	TARGET VEHICLES					
		5001	5002	5003	5004	5005	5006
40	OR Gate Amp. Change - Programmer XVI	R	R	P	P	P	P
43cc	PPS Turbine Exhaust Support Brkt Mod.	P	P	P	P	P	P
44cc	PPS Design Changes	P	P	P	P	P	P
45cc	TLM Control Unit - Rework	R	R	P	P	P	P
46	C Band Antenna Ass'y Mod.			P	P	P	P
49	DC/DC Converter Temp. Cycle Test	R	R	P			
50	SPS Gas Regulator Temp. Recond. & Setting	P	P	P	P	P	P
51R1	Reactivate GATV Running Lights		R	R	P	P	
53	Booster Adapter Strain Gage Change						
57cc	SPS Thermal Shield	P	P	P	P	P	P
58cc	MAC Battery Mounting Change	R	R	P	P	P	P
59cc	Transducer Change - B-71 Measurement	R	R	P	P	P	P
60R1	Vehicle Horizon Sensor Bias Angle		R	P	P	P	P
61	Electronic Filter Box	R		R	P	P	P
65cc	Encoder Programmer Module Redesign	P		P	P	P	P
66	Battery Installation Structure Mod.	R	R	P	P	P	P
67FC-1	Attach Bolt Torque Values	R	R	R	P	P	P
68	Shroud Fitting Normalizing	R	R	P	P	P	P

Table VII. B-1 (Continued)

ECP	TITLE	TARGET VEHICLES					
		5001	5002	5003	5004	5005	5006
73cc	Running Lights & Approach Lights		R				
74	Installation Procedure - Explosive Bolt Ass'y	R	R	P	P	P	P
75	Replacement - Texas Instrument Integrated Circuit Mod.	R		R	P	P	P
78	Attitude Control N_2 Regulator Thermal Surf.	R	R	P	P	P	P
79	Command Controller IV (5002)		P				
80R1	Guidance "J" Box Temp. Testing					P	P
82	SPS Unit #1 Valve Cover - Thermal Surface	R	R	P	P	P	P
83	Guidance Module Ass'y Mod.			R	R	P	P
85	Gemini/Agena Shroud Separation Debris Shield	R	R	P	P	P	P
86	Eliminate TLM Status Bit 50 Error	R		R	P	P	P
87	Coating of Bulbs - Running Lights	R	R	P	P	P	P
89C1	Reactivate GATV Running Lights	R		P	P	P	P
91^{FC1}_{cc}	Mod. - P7 Harness - Ground Loop	R	R	R	P	P	P
92cc	SPS Heat Shield Paint Pattern	R	R	P	P	P	P
93cc	Shroud Harness Mod.	R	R	P	P	P	P
94	Command Controller Refurb. (3 units)						

Table VII. B-1 (Continued)

ECP	TITLE	TARGET VEHICLES					
		5001	5002	5003	5004	5005	5006
95	5002 Test Plan Change		R				
97	Command Destruct Kit Improvement					P	P
98	Flight Control J-Box Temp. Testing	R				P	P
99cc	Mod. - Z4 Harness Ass'y	R		R	P	P	P
101-R1	Project "Sure-Fire" Engine Mod.	R		R	P	P	P
103	Electrostatic Discharge Measurements			R	R		
104	Sequencer Timer Documentation Changes					P	P
106R1	Fwd & Aft Pwr Dist. J-Box Relay Change	R		R	R	P	P
107cc	B71 Measurement - (ECP 59) Plumbing	R		R	R	P	P
111	Programmer Traj #2 Mod.	R		R	P	P	
112cc	Hydraulic Channel Grding Change	R		R	P	P	P
114-C1	FCEP LS8488 Transistor Replacement	R		R	P	P	P
115	Guidance Module J-Box Temp. Testing				Spare Unit Only		
118	Direct Control ERT Reset						
119cc	Addition of Isolation Relay	R		R	P	P	P
121	C-Band Xponder	R		R	P	P	P
122cc	Tape Recorder Sync. Loss (Harness P-10)	R		R	P	P	P

Table VII. B-1 (Continued)

ECP	TITLE	TARGET VEHICLES					
		5001	5002	5003	5004	5005	5006
124cc	Harness Z-4 Mod.	R		R	P	P	P
125	Fwd Pwr Dist J-Box Babcock Relays	R		R	P	P	P
126R1	Programmer & Controller (CCN 52)	R		R	P	P	P
131-1 -4	Hyd. Gain Switching & FCLP Digital Cmd. Disable						
132	Fwd Pwr Dist. J-Box, Increase Current Capacity	R		R	R	P	P
133-2	AGE-Battery Trickle Charger Mod						
134-2R1	AGE-"Hangar E" PCM Grd. Stat. Synch Pulse						
135	Testing - Thermal Surfaces						
139E-R1	GATV Engine Electrical Controls	P		R	R	P	P
140	5001 Refurbishment	R					
141	Vehicle 5003 SPS Unit Thruster Alignment			R			
143	Vehicle 5003 Sequence of Events & H/S Angle Change			R			
146	Shroud Ejection Spring Bracket Mod.	P		R	P	P	P
148	Gemini/Agena Vehicle Detail Spec. Changes	P		R	R	P	P
149	PCM/TM System Change	P		R	P	P	P

Table VII. B-1 (Continued)

ECP	TITLE	TARGET VEHICLES					
		5001	5002	5003	5004	5005	5006
152cc	FAR/FER Thermal Control Surface Change	P		R	P	P	P
154	Pyro Operated Helium Valve Change	R		R	R	P	P
159	Bracket for UDMH Line	R			R	P	P
162R1	Gemini Peculiar Changes Due to Agena D Derived					P	P
165	VHF Types TM Orbit Antenna Change	R			R	P	P
167	Installation of Mod. Velocity Meter Counter	P			R	P	P
171	Sequencer Timer Switch Mod. 1 (B) only	R				R	R
179	Gyro Monitor Range Extension	R			R	R	R
180R1	CG Offset Correction	P			R	P	P
183	MAC/GDC Derived Changes (paper only)	P			P	P	P
184	Mod. Agena Booster Separation Monitor	P			R	P	P
187	Add. of Ascent Sequence Timer Functions	P				R	P
188R1	Harness Ass'y F-3 Mod.	P				P	P
189	Propellant Venting Instrumentation	P			R	R	R
191	Add. of Flag Emblems to GATV's	P			R	R	R

Table VII. B-1 (Continued)

ECP	TITLE	TARGET VEHICLES					
		5001	5002	5003	5004	5005	5006
192	Vehicle Accept. Test Documents & GATV Detail Spec. Changes					P	P
193	Vehicle 5004 Horizon Sensor Bias Angle Change				R		
195	Electro Static Discharge Measurement Vehicle NO. 5005					R	
196	Vehicle 5005 Horizon Sensor Bias Angle Change					R	
197	Vehicle System Test Sequences					P	P
198	Maintenance Testing of Program Peculiar C&C Equip						Spares
199	Velocity Meter Counter Change, Vehicle 5005					R	
201R1	Velocity Meter Mod. IIA Counter Replacement	R					R
202	TDA (MAC) Reidentification Vehicle 5001	R					
203	Vehicle 5006 Horizon Sensor Bias Angle Change & Sequence of Events						R
204	Vehicle 5001 Horizon Sensor Bias Angle Change &Sequence of Events	R					

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