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RENDEZVOUS CONCEPT FOR CIRCUMLUNAR FLYBY IN 1967

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CONTENTS

SUMMARY

This study was made in collaboration with NASA Manned Space Flight Center and the McDonnell Aircraft Corporation to determine how a circumlunar flight could be made during 1967 using existing space hardware building blocks and operational techniques that will have been developed by 1967. Several approaches have been considered using the Titan IIIC booster; e.g., a direct flight approach and an earth-orbit rendezvous. The most practical concept, which is detailed in this report, consists of launching a modified Gemini A capsule by a Gemini launch vehicle (GLV) to rendezvous with a stripped Titan III transtage propulsion module containing a modified Agena target docking adapter (TDA). The propulsion module will be launched into orbit by a standard Titan IIIC booster. The optimum launch technique involves launching the GLV several minutes after the Titan IIIC is launched, with initiation of intercept and rendezvous maneuvers to start at the Titan IIIC orbit injection point. If the GLV launch does not take place, a second opportunity will occur about one orbit period later, when the transtage of the Titan IIIC overflies the launch site.

The stripped transtage provides an adequate propellant load to insert an 8000-pound Gemini capsule into a circumlunar transfer orbit with approximately a 7.5-percent performance margin. Likewise, the GLV is capable of inserting the 8500-pound Gemini capsule into a 100-nautical-mile earth orbit.

No system changes are necessary to the standard Titan IIIC booster or the Gemini launch vehicle for this mission. The interface between the Gemini capsule and the stripped transtage propulsion module is designed to use the standard nine-wire umbilical connector being used in the present Gemini rendezvous program. Investigation of the structural dynamic and flight control aspects associated with the connection of the Gemini capsule and the transtage propulsion module by means of the Agena target docking mechanism has indicated that mixing of rate-derived control signals from the Gemini capsule with rate gyro signals in the transtage will provide adequate stability and load margins.

The program provides for three Titan IIIC and Gemini spacecraft launchings:

1) Heat shield qualification test - Mount a Gemini capsule on a Titan IIIC booster and fly a trajectory designed to duplicate the reentry heating rate expected on the capsule heat shield as it enters the earth's atmosphere after completing the circumlunar flyby;

2) Earth orbit rendezvous checkout - Launch Titan IIIC and manned GLV at near-simultaneous launch time and rendezvous in a near-earth orbit to verify all systems and procedures;

3) Circumlunar flyby mission - Launch Titan IIIC and manned GLV, rendezvous, and inject Gemini capsule into circumlunar transfer orbit, reenter, and recover.

Ground launch system facilities at the Eastern Test Range exist for both Titan IIIC and Gemini launch vehicle and only minor modifications of these facilities will be necessary for this mission.

Launch vehicle hardware can be provided and launch operations accomplished for the complete program within 20 months from program initiation.

If the success of the Gemini program continues at its present pace, it is highly probable that the major program objectives will be accomplished well before the 12th flight. Because of this it appears desirable to think seriously about expanding Gemini program objectives to include additional functions that would enhance and complement the Apollo program with space "firsts" of sufficient significance to greatly improve our national prestige.

With the above objectives in mind, a study was made in collaboration with the NASA Manned Spaceflight Center and the McDonnell Aircraft Corporation to determine how a circumlunar flight could be made during 1967. Several approaches were considered. The one described in this document takes maximum advantage of the space hardware building blocks and operational techniques that will have been developed by 1967. The concept consists of a modified Gemini A capsule launched by a Gemini launch vehicle (GLV); the capsule to rendezvous with a stripped Titan III transtage propulsion module, containing an Agena docking device, that has been launched into orbit by a standard Titan IIIC. This stripped transtage provides an adequate propellant load to insert an 8000-pound Gemini capsule into a lunar orbit with approximately a 7.5-percent performance margin.

Based on the present schedule of activities, by the second quarter of 1967 the Gemini A capsule, after some 10 manned flights, will have logged mission times considerably in excess of a minimum-energy lunar flyby. The GLV will have flown 12 times. The transtage will have flown 13 times -- nine times with the complete Titan IIIC launch vehicle. Rendezvous between the Gemini vehicle and the Agena docking device will have been accomplished several times. In summary, all major building blocks of the concept described herein will have been exercised many times before their proposed use in the circumlunar mission.

Technical aspects of a Gemini lunar flyby program have many parallels to the present near-earth Gemini program, e.g., exercise of the global net for the lunar mission. Voice communications at lunar distances as well as the many aspects of guidance, navigation, and general supporting operations can be exercised. Such a program can also provide useful lunar reconnaissance data. Many other semispectacular feats can also be accomplished, e.g., large maneuvers in space, highly elliptical orbits, flyby inspections of 24-hour orbital period communication satellites, and demonstration of in-space maintenance.

The chapters that follow provide backup details for a building-block approach to a lunar flyby in 1967 in these technical areas: launch vehicle configurations, stripped transtage, circumlunar transfer configuration, mission profile, rendezvous operations, launch vehicle performance, and program planning.

A. LAUNCH VEHICLE CONFIGURATION

1. Titan IIIC Vehicle

For the Gemini mission, the Titan IIIC standard space launch vehicle (SSLV) is used to place in orbit a payload comprising a modified transtage and docking mechanism. The Titan IIIC consists of a solid rocket motor (SRM) Stage 0, a liquid rocket Stage I, a liquid rocket Stage II, and a liquid rocket Stage III (transtage). The first three stages of Titan IIIC are used without modification. The transtage is modified to reduce weight by removing some of the R&D instrumentation. An interstage skirt is added to the forward end of the SSLV transtage (Sta 77.0), extending the structure as required to accommodate a second transtage as part of the earth-orbital payload. The overall vehicle configuration at launch is shown in Figure II-1.

Stage 0 consists of two solid rocket motors (SRM) strapped on in parallel to Stage I. Each SRM has five cylindrical center segments, an aft closure containing a rocket nozzle, and a forward closure containing thrust termination ports. Each SRM develops an initial sea-level thrust of approximately 1,190,000 pounds. A regressive burning characteristic provides a thrust decay to approximately 800,000 pounds per motor under vacuum conditions at the start of thrust tailoff. Attitude is controlled by a fluid injection thrust vector control system that receives signals from an upper-stage flight control computer.

Each SRM is 10 feet in diameter and approximately 85 feet long. The aft ends of the SRMs are attached to the two longerons on the aft skirt of Stage I. The forward end of each SRM is attached by horizontal stabilizing supports to the forward skirt of Stage I. Staging is accomplished at Stage 0 burnout by pyrotechnic release of the SRM and lateral thrusting by four small solid rockets on each end of the SRM.

A separate ordnance destruct system is provided for range safety in the event of early flight termination.

Stage I is a liquid-propellant rocket stage 10 feet in diameter and approximately 71 feet long, including the interstage adapter to Stage II. The propellants are nitrogen tetroxide oxidizer and a 50/50 hydrazine/UDMH fuel. The engine assembly consists of two subassemblies, each composed of a regeneratively cooled thrust chamber, gimbals, a turbopump, a gas generator, and attendant connecting plumbing. The engine assembly is rated at 430,000 pounds of thrust at sea level. The in-tandem propellant tanks form an integral part of the vehicle structure and are pressurized in flight by an autogenous pressurization system.

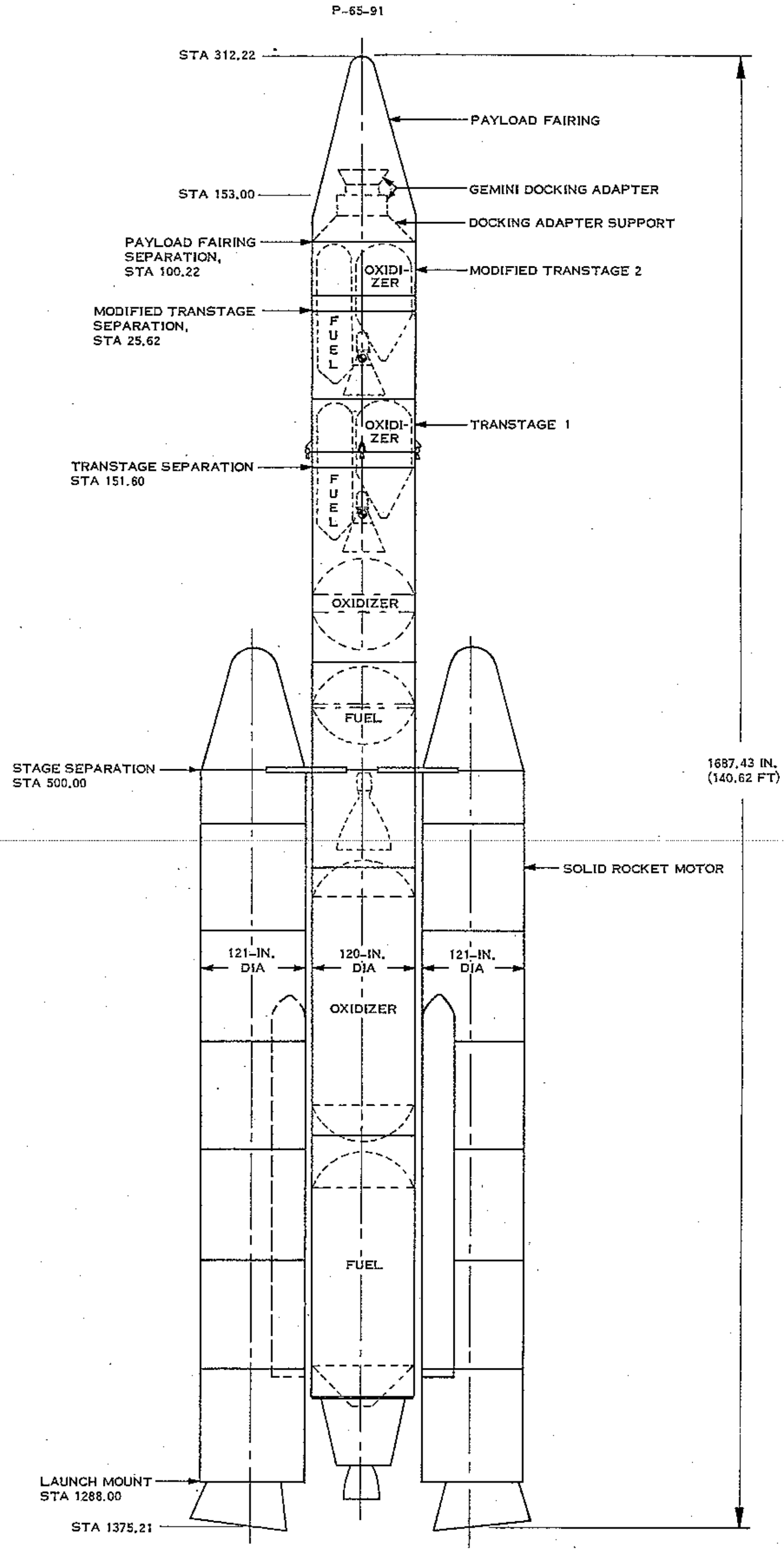


FIGURE II-1 TITAN IIIC OVERALL CONFIGURATION

Stage I is controlled by gimbaling the engine subassemblies with hydraulic actuators in response to signals from an upper-stage flight control computer. A rate gyro package is provided in Stage I for stability, and a lateral-acceleration-sensing system is provided for load relief (when required) during Stage 0 flight.

An ordnance destruct system is provided for range safety in the event of early flight termination.

Staging is accomplished by thrust chamber pressure switch initiation of engine shutdown, pyrotechnic release of Stage I, and fire-in-the-hole separation of Stage II.

Special environmental protection of Stage I includes an ablative coating in the forward section to accommodate Stage II firing and a heat shield around Stage I engines to provide thermal radiation and hot gas protection during Stage 0 burning.

Stage II is also 10 feet in diameter and, including the forward skirt, is approximately 37 feet long. The propellants are the same type as in Stage I and the tankage configuration is similar. The turbopump-fed engine is regeneratively cooled, has an ablative skirt, is gimbaled for pitch and yaw control, and has a gas generator. Developed thrust is 100,000 pounds at altitude. Roll is controlled by directing gas generator exhaust through a rotatable nozzle.

Stage II is controlled by gimbaling the engine and swiveling the roll nozzle with hydraulic actuators in response to signals from an upper-stage flight control computer. A rate gyro package provides stability.

An ordnance destruct system is provided for range safety in the event of early flight termination.

Staging is accomplished by thrust chamber pressure switch-initiated or guidance-initiated engine shutdown, pyrotechnic release of Stage II, and retrothrust from three solid rockets. Guides in Stage II that operate in conjunction with Stage III rails are provided to ensure separation clearance.

Stage III is the transtage (Fig. II-2) referred to as transtage 1 in this report, which performs as the final stage of the launch vehicle.

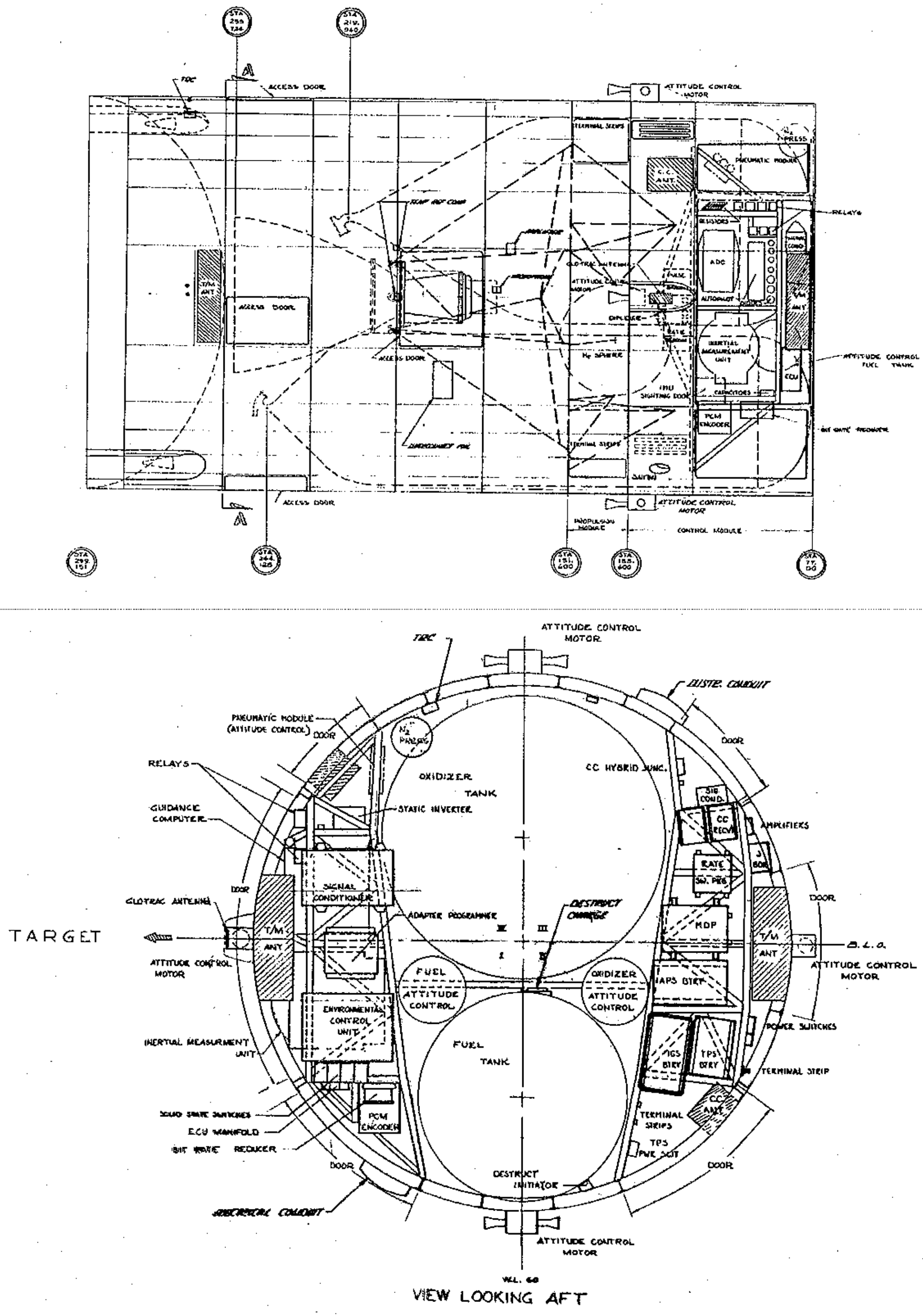


Figure II-2 Transtage 1

The basic transtage includes two structural modules:

1) A propulsion module consisting of two side-by-side propellant tanks and trusses, trusses for two engines, and an 18-inch-long, 10-foot-diameter cylindrical skin section;

2) A control module consisting of equipment trusses and a 56.6-inch-long, 10-foot-diameter cylindrical skin section. The top of the transtage is Station 77.0. A new interstage skirt is added above Station 77.0 to adapt to the upper transtage of the payload. This cylindrical skirt is constructed of aluminum honeycomb for weight saving. It is similar in design but thicker than the skirt described later for the upper transtage (2). Access doors in the skirt are provided for servicing transtage 2.

The subsystems of transtage 1 are the same as in the existing Titan IIIC SSLV, except for instrumentation. The main propulsion is provided by two gimbaled, 8000-pound-thrust engines. Coast attitude control is provided by an attitude control system (located in the equipment module) that consists of four aft-pointing, 45-pound-thrust engines and two pairs of tangential, 25-pound-thrust engines. In transtage 1, guidance and flight control are provided by an inertial guidance system and an analog flight control system. Electric power is provided by batteries. A range safety tracking and command system and an operational telemetry system are provided.

The transtage's capability to operate in orbit for longer than the specified 6½ hours is discussed in the following subsections.

a. Transtage 7½-Hour Capability

In a recent operational flight, the transtage was attitude-stabilized for 8.3 hours. When the attitude control system ceased operating, the ACSP guidance system was still functioning properly. Flight data have not been sufficiently analyzed to determine the reason the attitude control system shut down. Though preliminary investigations indicate that battery exhaustion was the most likely cause, other possibilities are:

1) Propellant-line freezeup;

2) ACS engine malfunction or freezeup;

3) Autopilot or other electronic malfunction.

Ground tests for the transtage included thermal balance testing in Lockheed's HIVOS facility. Test results indicated that there are certain temperature limitations beyond 6½ hours. However, the tests were conducted with no orbital heat simulation, which represents an ultraconservative case. The actual orbital flight of the transtage revealed that in near-earth orbits, temperature drop is not as rapid as that induced for the HIVOS tests. In these tests, the operations of certain components and parts became marginal at the low-temperature portion of the cold-soak test (no external heat). These include:

1) Hydraulic motor pump power switch;

2) Hydraulic reservoir;

3) Supply/return hydraulic lines near the actuators (no insulation);

4) Main engine propellant lines;

5) ACS roll/yaw modules.

Generally, however, operational results and ground tests indicate the feasibility of using a standard transtage for the proposed 7½-hour mission. The possibility of parts of the ACS freezing would be lessened by having a propulsion module on the forward end of the standard transtage, since the loaded propellant tanks of this module would represent a large heat source.

Mission limitations would be:

1) Battery exhaustion;

2) ACS engine malfunction;

3) Electronic equipment malfunction;

4) Some of the main engine components such as the hydraulic system and propellant lines dropping below design temperature.

Assuming that there is no requirement for starting the main engines as late as 7½ hours, and keeping the above limitations in mind, it appears that the standard transtage can meet mission requirements.

The special propulsion module (transtage 2) also appears to have no thermal problems for the mission duration contemplated. The fact that the main propellant tanks are full gives a very large thermal mass that should help maintain component temperatures in acceptable ranges. The surrounding structure will also act to prevent heat loss in a manner similar to the aft shroud being used for the 30-day transtage. Studies should be conducted to determine whether the skin of the special propulsion module should have some special surface coatings, and whether it is necessary to apply some multilayer insulation to the inside of the skin.

b. Proposed Modifications for 30-Day-Mission Transtage

In our efforts to develop a long-life transtage, Martin Company is investigating the following possible modifications:

1) Add thermostats to control ACS engine module heaters;

2) Remove main propellant tank insulation and ACS propellant line water jackets;

3) Remove hydraulic component insulation;

4) Remove main propellant line insulation and main engine insulation;

5) Add electrical disconnects, motor-driven switches, control wiring, and two 100-pole switches to permit control and monitoring of various transtage functions and to supply electric power to transtage;

6) Add aft shroud and shroud mounting structure;

7) Modify paint patterns (forward cylinder);

8) Replace destruct system detonator cord with all-electrical initiation system;

9) Braze or weld selected plumbing joints to minimize leakage.

These modifications are estimated to weigh 400 pounds.

c. Titan IIIC Launch Hold Capability

The capability of Titan IIIC to perform a mission requiring a specific launch time is indicated by the following hold and reaction requirements of the standard system. The Titan IIIC is capable of the following launch hold times:

1) Up to 30 days at T - 195;

2) Up to six hours at T - 45;

3) Up to one hour at T - 1 (if repeated instrumentation and range safety checks are waived).

2. Gemini Launch Vehicle

The Gemini launch vehicle is a Titan II ICBM booster with certain basic modifications to accommodate the Gemini mission and to provide man-rating. Fundamentally, the modifications can be broken down into the following 11 categories:

1) A redundant flight control system is provided to give adequate time for safe crew ejection from the spacecraft. This is a parallel system composed of the spacecraft inertial guidance system (IGS), a second autopilot identical to the primary autopilot, and allied equipment that feeds into a tandem hydraulic actuator;

2) A malfunction detection system (MDS) is provided to assure pilot safety. The following items are sensed and displayed,

   a) Oxidizer and fuel tank pressure, Stages I and II,

   b) Pitch, roll, and yaw overrate,

   c) Staging bolt firing,

   d) Stage I combustion chamber pressure,

   e) Stage II combustion chamber pressure,

f) Low voltage,

g) Command;

3) All MDS circuits are fully redundant;

4) Provision is made for engine shutdown from the spacecraft;

5) A fully redundant method of giving the staging signal and actuating the Stage II engine starter cartridge is provided;

6) A 120-inch-diameter skirt is provided forward of the Stage II oxidizer tank to furnish an interface and mating plane between Stage II and the spacecraft adapter;

7) A redundant electrical system is provided to furnish backup circuits for guidance, engine shutdown, and staging;

8) Titan II retrorockets, vernier rockets, and allied equipment are removed;

9) The Titan II IGS is replaced by a GE Mod III-G radio guidance system (RGS);

10) A three-axis reference system (TARS) is added to furnish attitude stabilization and programing;

11) Titan II equipment support trusses are replaced with trusses designed to meet Gemini requirements.

The GLV configuration is shown in Figure II-3.

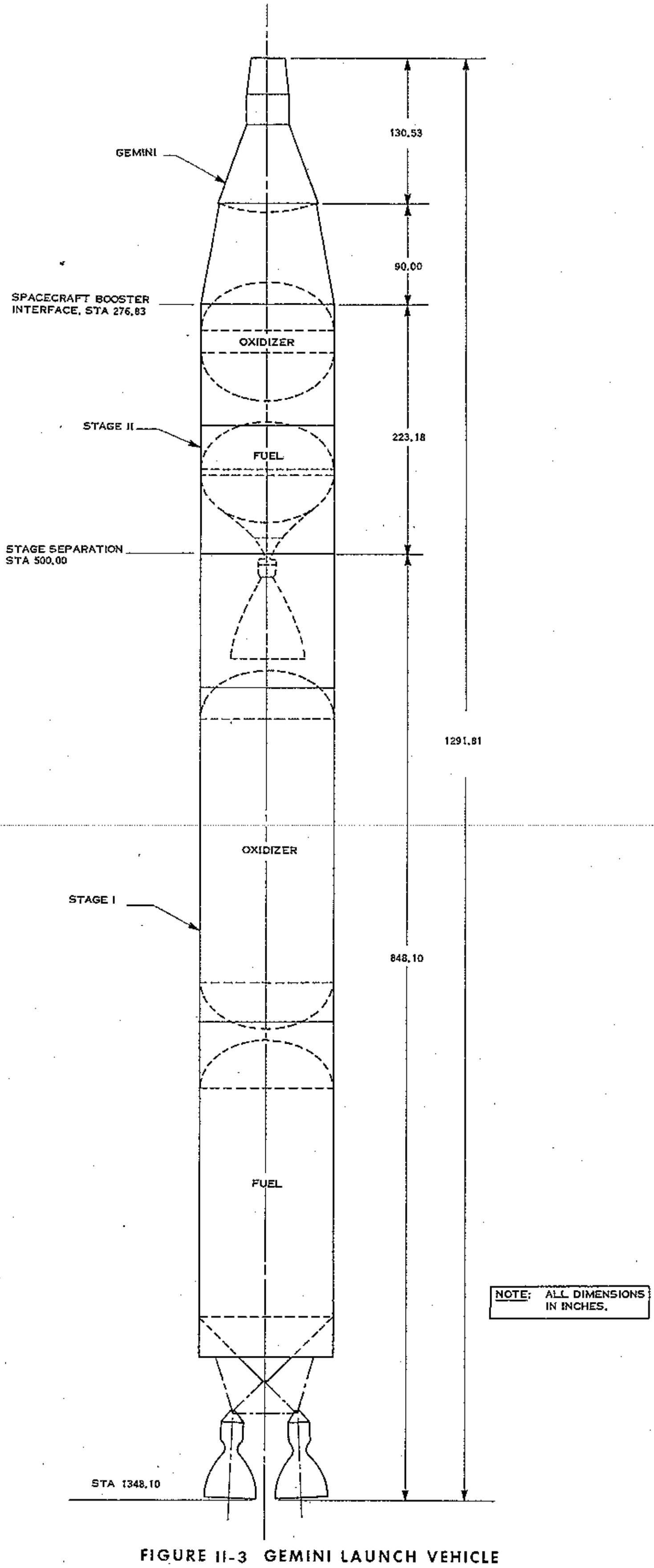


FIGURE II-3 GEMINI LAUNCH VEHICLE

3. Transtage 2

The second transtage is added above the Titan IIIC to provide the propulsion for circumlunar transfer of the Gemini. Transtage 2 is a version of the standard Titan IIIC transtage, primarily modified to reduce weight by removing equipment not required for this mission and by redesigning portions of the structure. Guidance for this stage is provided by inputs from the Gemini capsule after docking. Therefore, the transtage inertial guidance system and supporting structure are removed. A new forward cylindrical skirt that incorporates a transition structure at the forward end to interface with a target docking adapter (TDA) is provided. The TDA is based on the Agena target vehicle TDA from the Gemini program. The structural configuration of transtage 2 is shown in Figure II-4.

a. Skirt Structure

Since the bending, axial, and shear loads are relatively low, we propose an adhesively bonded aluminum construction to obtain the lowest weight for the cylindrical forward skirt structure. The outer skin will be subjected to a maximum aerodynamic heating temperature of approximately 450°F for a short time and, therefore, HT 424 epoxy-phenolic adhesive will be used for bonding. Since the basic structure of the standard transtage includes eight longerons on the 120-inch-diameter periphery, the new honeycomb skirt will also provide for eight longerons to make the splice attachments conform to the existing structure.

b. Propulsion Module

The propulsion module of transtage 2 is identical to the standard Titan IIIC transtage except that part of the environmental control provisions are removed. This is possible because the powered flight time of the second transtage is very short and because thermal control of the standard transtage is provided for 6½ hours in orbit. The on-orbit time limitation for transtage 2 is affected by the low-temperature limits of the main propellant lines and possibly by the electric batteries. However, as long as transtage 1 is attached, the normally exposed propulsion module tank lines, engines, etc, of transtage 2 are covered by transtage 1. This added thermal protection will be more than adequate for 6½ hours, and may extend the thermal life of the system to approximately 10 hours.

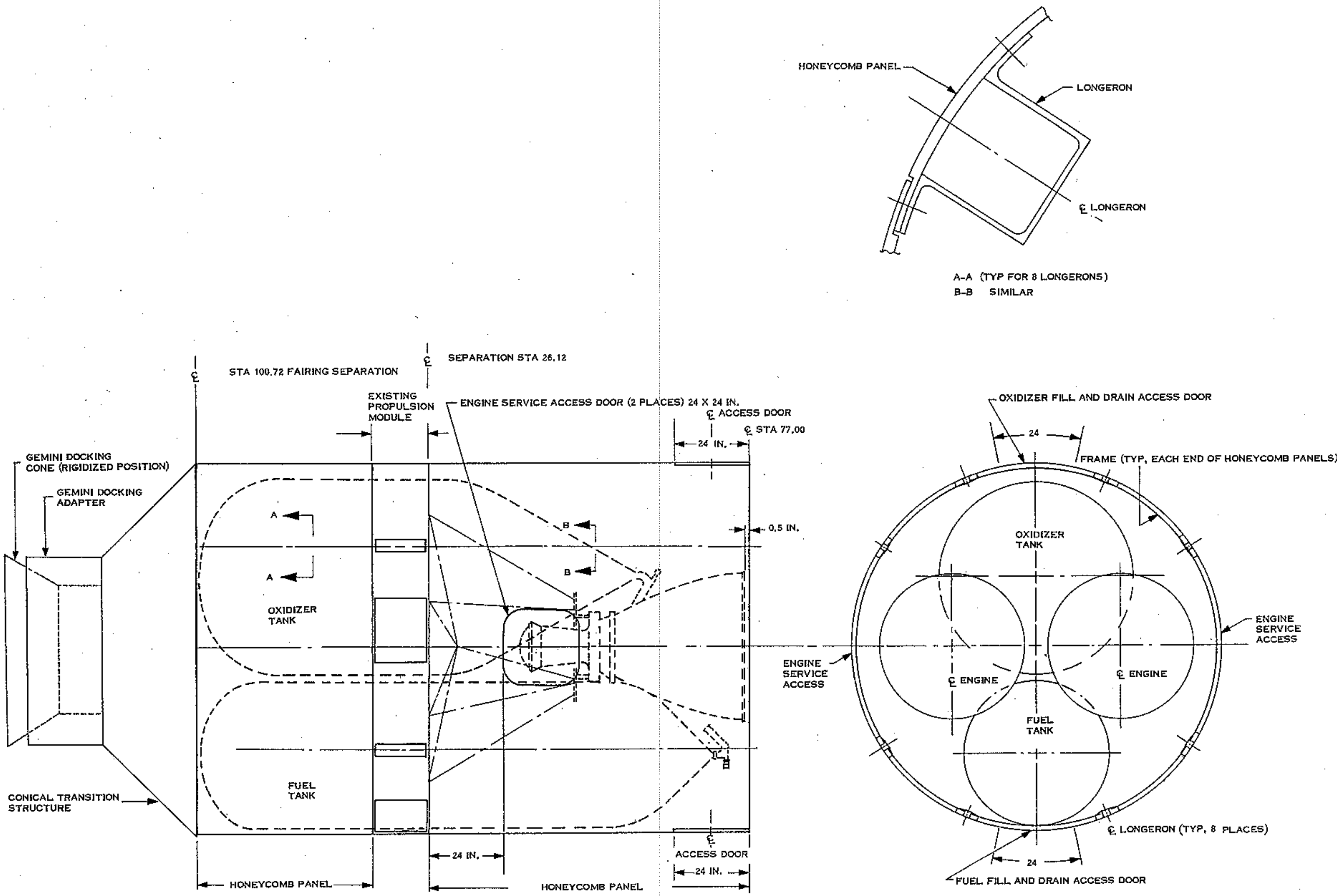


FIGURE II-4 TRANSTAGE 2

c. Electrical

The electrical system is shown in Figure II-5. This system consists of two power supplies: the auxiliary power supply (APS) and the transient power supply (TPS). The present TPS 25 amp-hr battery can be used in the transtage 2. It has heaters to prevent freezing. The APS 4 amp-hr battery is environmental-approved for transtage application under load. It is planned not to use the transtage 2 APS batteries until separation from transtage 1. Therefore, a transtage 2 APS battery freezing problem may occur.

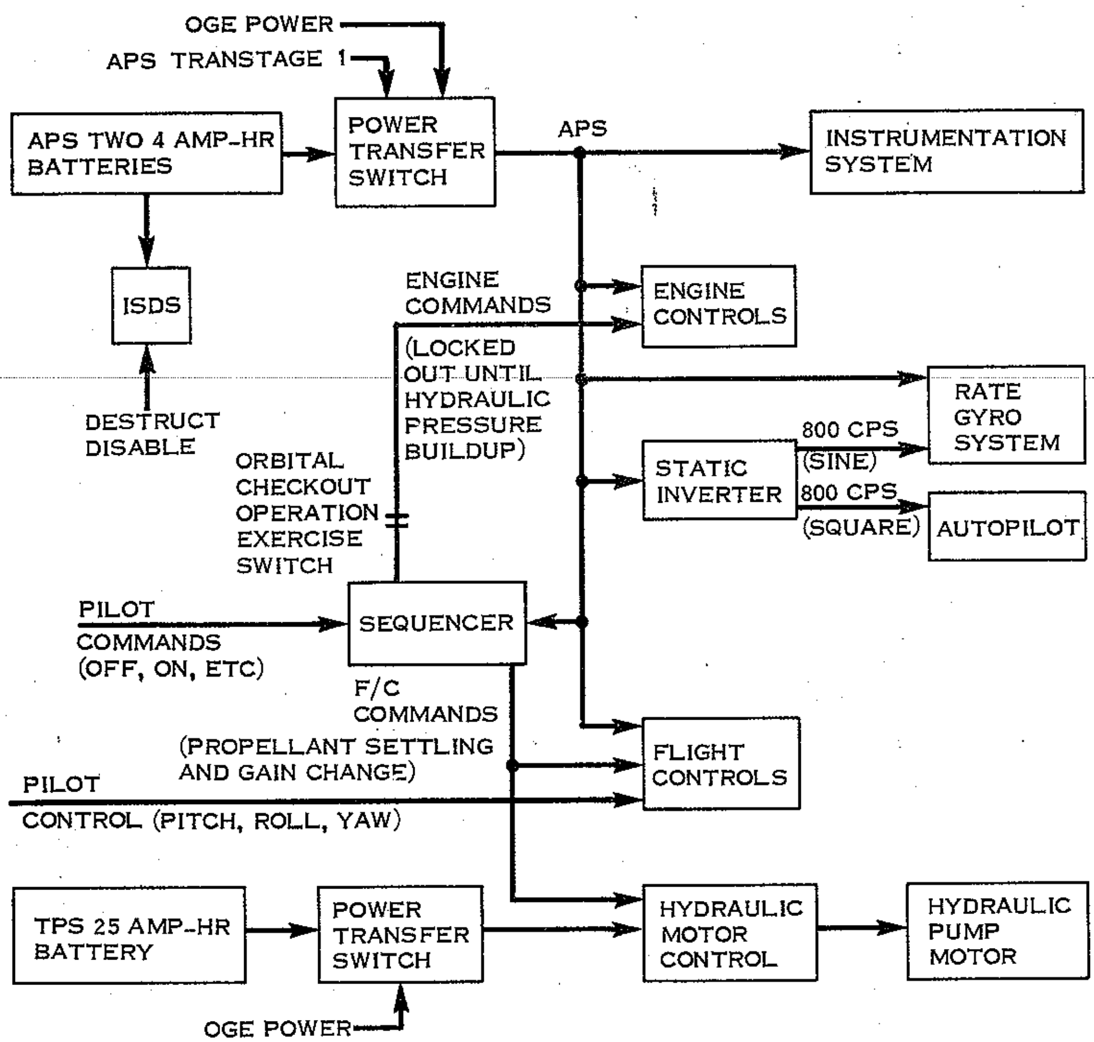


Figure II-5 Transtage 2 Electrical System

The APS uses two 4 amp-hr batteries to power the engine control, the static inverter, the rate gyro, the flight controls, the sequencer, the inadvertent destruct system (before Stage II engine shutdown), and the instrumentation systems. To reduce transtage 2 weight, the above functions will use the transtage 1 APS as long as feasible.

This requires transtage 1 and transtage 2 to remain attached until approximately 5 minutes before the transfer to the lunar trajectory.

The TPS drives the hydraulic pump. The pump is started before transtage 2 engine firing, and hydraulic pressure is a prerequisite to this firing. The flight plan uses the Gemini OAMS for propellant settling. A simultaneous or the same Gemini propellant settling signal is required to start the hydraulic pump motor.

During operations and checkout before launch, power will be supplied by the OGE.

The transtage 2 and Gemini power supplies are isolated. This isolation is accomplished by relays in the sequencer. The pilot commands close relays in the sequencer, which completes the transtage 2 circuits.

The other pilot inputs and steering commands are accomplished in the same manner. These steering commands close relays completing the flight control circuits.

d. Instrumentation and Telemetry

Scope - A telemetry system is provided in transtage 2 to transmit data during the period the spacecraft is docked to the transtage. Certain operational measurements are wired through the transtage 1/transtage 2 interface and transmitted via the transtage 1 telemetry system during launch and the early portion of orbit.

Measurement Requirements - Table II-1 is a tentative list of telemetry measurements for transtage 2. Of these measurements, tank pressures, destruct disable, and battery voltage will be wired in parallel to transtage 1 telemetry for ground monitoring prior to spacecraft docking.

Table II-1 Transtage 2 Measurements List

Item	Location
Pressure (9)	Oxidizer Tank; Fuel Tank; Helium Tank; Thrust Chamber S/A 4; Thrust Chamber S/A 5; Oxidizer Suction S/A 4; Fuel Suction S/A 5; Thrust Chamber Pressure Switch 1; Thrust Chamber Pressure Switch 2.
Position (2)	Thrust Chamber Valve S/A 5; Thrust Chamber Valve S/A 4.
Actuator Travel (4)	Pitch S/A 4; Yaw S/A 4; Pitch S/A 5; Yaw S/A 5.
Hydraulic Pressure (1)	Transtage
Voltage (9)	Transtage 1/Transtage 2 Separation; X Stage Fire Engine (FS 1), Engine Shutdown (FS 2) Signal; Gemini Docked/Undocked; Hydraulic Battery; APS-IPS Battery; Flight Control Static Inverter; Rate Gyro Out Pitch; Rate Gyro Out Yaw; Rate Gyro Out Roll.
Gain Change (1)	Autopilot
Destruct Disable Signal (1)	Destruct System
Voltage (5-vdc Power) (1)	Telemetry System
Temperature (2)	Oxidizer Suction S/A 4; Fuel Suction S/A 5.
Total Functions Measured - 30	

RF Link Analysis - The results of an RF link analysis to determine required transmitter power are presented in Table II-2. These results are based on the following assumptions:

1) Frequency - 250 mc;

2) Bandwidth - 100 kc;

3) Type of modulation - PCM;

4) Ground antenna - 30-foot dish (shipboard);

5) Vehicle attitude - elliptical orbit 80 to 160 miles, target down, pitchup of approximately 5 deg during burn;

6) Maximum short range - 1500 miles;

7) Receiver plus antenna temperature - 750 F.

Table II-2 Analysis of Transmitter Power Requirements

Item	Gain (db)	Loss (db)
Antenna gain (G_R) - 30-ft dish, 250 mc	24.9	
Receiver sensitivity = 10 log KTB	120.0 (milliwatts)	
Signal-to-noise		13.0
Airborne antenna gain (G_T)		
Space loss (α = 37 + 20 log f + 20 log R)		148.5
Line Losses		1.0
Polarization		3.0
Safety factor		6.0
Total	144.9	171.5
Required transmitter power P_T = 171.5 - 144.9 = 26.6 dbm = 0.457 watt.		

<u>Telemetry System Configuration</u> - Figure II-6 is a block diagram of the telemetry system for transtage 2. The transmitter will be a 2-watt solid-state unit drawing approximately 0.7 ampere of 28-vdc power. A high-level, 32-channel PCM encoder will be used to encode the data. This encoder will be a reduced version of the encoder used on the orbiting solar observatory (OSO) program. The pressure transducer will be pot-type transducers with 3-percent accuracy to provide a high-level input directly to the encoder. Signal conditioning is required for the static inverter voltage measurement. This will be identical to that used on Titan III with the exception of a high-level output. The antenna will be a modified Titan I operational antenna. Foaming will not be required for this antenna due to the low transmitter power. Transducer power will be supplied from a Martin-built 5-vdc power supply. Table II-3 lists the power and weight requirements for the telemetry equipment. 28-vdc power will be obtained from one 4 amp-hr battery. This will provide in excess of 4 hours of telemetry operation after spacecraft docking.

<u>Ground Station Limitations</u> - Similar limitations apply to tracking, and it would be advantageous to have rendezvous of the Gemini and the two transtages over a tracking station and TM station.

The transfer from 100 nautical-mile orbit to lunar trajectory will have to occur over a TM station with real-time playback.

Table II-3 Power and Weight of Telemetry Components

Component	Weight (lb)	Power (watts)
Encoder	4.0	0.5
Transmitter	2.0	20.0
5 vdc Power	1.0	1.0
Transducers	5.0	--
Antenna	3.5	--
Signal Conditioner	3.0	--
Brackets	10.0	--
Wire and Connectors	15.0	--
Totals	43.5	21.5

e. Engine Control System

The control system that will be used for powered flight after the Gemini has docked to the transtage will consist of a flight control computer and a rate gyro package located in transtage 2, plus attitude rate and attitude error signals from the Gemini capsule. It is possible to eliminate the need for Gemini rate data by passing the Gemini attitude error signal through a lead-lag filter in the flight control computer as is presently done in the Titan III transtage.

A more detailed discussion of this system is given in Subsection B.2.

f. Target Docking Adapter

The Gemini capsule TDA developed by McDonnell for the Agena structure weighs 360 pounds. Since the Agena is 5 feet in diameter and the Titan transtage cylindrical skirt is 10 feet in diameter, a conical transition structure is needed. Two structural approaches are possible and will be investigated. First, a skin-stringer conical structure frame can be considered and, secondly, a welded tubular truss structure could be designed. At this time, 40 pounds has been added to the weight estimate to provide for this transition structure.

The present Agena target vehicle includes the capability to command attitude and velocity change maneuvers of the Agena from the ground. This feature is considered unnecessary, so it is not included. A tape recorder is provided to record telemetry information for later transmission to the ground. This feature is also not required and is not included. The radar transponder, radar transponder antenna and erector, and the radar transponder power regulater installation normally in the TDA are moved to the interstage skirt of transtage 1 to save approximately 48 pounds of circumlunar payload weight.

g. Docking

The problem of transtage motion before Gemini docking was investigated to see whether command-control of the attitude and motion of the two transtages from the Gemini capsule before docking is necessary. The docking face motions before docking are summarized in Table II-4. This indicates that the attitude control system in transtage 1 can keep the motion of the docking face to such low frequencies and travels that commands from the Gemini capsule to the transtage are not necessary.

Table II-4 Lunar Flyby Gemini/Titan IIIC Docking Face Motions Prior to Docking (Pitch or Yaw Effect)

Engine On-Time (millisec)	Docking Face Travel (in.)	Angular Velocity (rad/sec)	Incremental Docking Face Velocity (fps)	Closing Velocity ÷ Face Velocity (No. Units)	Period (sec)	Center of Gravity Linear Velocity (fps)
100	±0.77	0.86×10^{-3}	0.009	88	42.8	0.0048
200	±0.77	1.72×10^{-3}	0.018	44	21.4	0.0096
400	±0.77	3.44×10^{-3}	0.036	22	10.7	0.0193

Conditions:

One 45-lb engine firing aft at a time;
Low-level mode (±½ deg);
Limit cycle amplitude, ±2 quanta or ±0.35 deg (one quantum is the IGS digital difference in signal output);
Angular rate imbalance ratio, 2 to 1 (ratio of magnitudes of positive-to-negative rates);
Center of gravity to docking face distance, 10.5 ft;
Closing velocity, 0.8 fps.

During docking, transtage 2 is controlled by the transtage 1 attitude control system (ACS). The ACS engines, four aft-pointing 45-pound thrust engines and two pairs of tangential 25-pound thrust engines, are pulsed to provide the attitude limits shown in Table II-1. The range of engine on-time shown, 100 to 400 milliseconds, is considered reasonable based on Titan III analysis and tests. A 100-millisecond on-time is probably the most valid for the subject mission.

h. Transtage 2 Weight Estimate

Table II-5 is a summary weight statement for transtage 2. The table indicates that considerable weight can be saved by removing equipment not required for this mission and by redesigning structural items to match the new mission requirements. Transtage 1 weights are summarized in Table II-6 for reference, and the total weight to be placed on-orbit by the Titan IIIC (weight above Station 77.0) is shown in Table II-7.

Table II-5 Transtage 2 Summary Weight Statement

	Equipment Module (lb)	Propulsion Module (lb)
Body	(151)	(161)
Honeycomb Panels (8)	100	
Longerons (8)	31	
Station -47 Frame	10	
Station -107 Frame	10	
Separation and Destruct	(29)	(56)
Wire	10	
Initiators	2	
Wafers	8	
Resistors and Primacord	1	
ISDS Kit	5	
Bracketry	3	
Propulsion	(--)	(1864)
Power Generating	(163)	(62)
4-amp-hr Batteries (2)	32	
25-amp-hr Battery	37	
200-amp Motor-Driven Switch	3	
20-amp Motor-Driven Switch	3	
Sequence System	9	
Wire and Connectors	40	
Diode Package	1	
Staging Connector Halves	3	
Installation Hardware	8	
Terminal Boards and Splices	2	
Truss and Supports	25	

Table II-5 (concl)

	Equipment Module (lb)	Propulsion Module (lb)
Orientation Control (Engine Actuator System)	(--)	(78)
Guidance	(55)	(--)
Autopilot (includes static inverter)	15	
Rate Gyro	15	
Wiring	15	
Support Structure	10	
Environmental Control	(5)	(25)
Insulation	5	
Telemetry	(31.5)	(18.5)
Wire and Connectors	9	6
Transducers	2	3
5-v Power Supply	1	0
Encoder	4.0	
Transmitter	2.0	
Antenna	3.5	
Signal Conditioner	3.0	
Bracket	7.0	3.0
Miscellaneous	0	6.5
DRY WEIGHT	434.5	2264.5
Payload Fairing Frame		(70)
Docking Mechanism and Installation		(352)
McDonnell Docking Mechanism		360
Remove Radar Transponder System		-48
Add Transition Structure		40
Residuals		(121)
Mean Outage		69
Pressure Gas		47
Vapor Retained		5
Ablative Material		(-30)
BURNOUT WEIGHT	434.5	2777.5
TOTAL TRANSTAGE BURNOUT WEIGHT	3212	

Note: Propulsion module data, excluding environmental control and telemetry, taken directly from Monthly Weight and Balance Status Report, Issue No. 20, SSD-CR-65-64. Environmental control reduced due to short flight time.

Table II-6 Weight Estimate, Transtage 1

Subsystem	Weight (lb)
Equipment Module	(2,129)
SSD-CR-65-64 Equipment Module	2,340
Remove Telemetry System	-488
Add Operational Telemetry	277
Propulsion Module	(2,408)
SSD-CR-65-64 Propulsion Module	2,451
Remove Telemetry System	-79
Add Operational Telemetry	36
Interstage Skirt (Above Station 77.0)	(350)*
Honeycomb Panels (8)	215
Longerons (8)	80
Station 77.0 Frame	15
Station -29.0 Frame	20
Access Doors and Miscellaneous Structure	20
Radar Transponder System	(48)
Transtage 1 Burnout Weight	(4,935)
Usable Propellant (Propulsion Module)	16,000
Attitude Control System Propellants	115
JGS Coolant	25
Plus Ablative Material	30
Transtage 1 Loaded Weight	21,105

*Not part of the existing transtage system weight.

Table II-7 Total Weight above Transtage 1, Station 77.0

Subsystem	Weight (lb)
Interstage Skirt and Radar Transponder	398
Transtage 2 Burnout Weight	3,212
Transtage 2 Usable Propellant	22,565
Ablative Material	30
Total Weight above Transtage 1, Station 77.0 (Loaded)	26,205

4. Destruct System

The destruct system designed for the Titan III standard space launch vehicle is compatible with the requirements of manned flight and with the range safety criteria imposed at ETR.

The only modification to the existing transtage (TS-1) would be the addition of two pairs of destruct wires across the interface. Each pair would contain a destruct positive and a negative (return) wire. These wires would provide the necessary signal to destruct the top transtage (TS-2) if a destruct command were received from the ground transmitters.

TS-2 would require a destruct system and an inadvertent separation destruct system (ISDS) as shown in Figure II-7. The ISDS circuitry will operate by sensing interruption of a voltage to TS-2. Power is then provided to start a timer that controls the squib fire circuit. When the timer runs out, the initiators are fired.

The normal destruct system for TS-2 requires current-limiting resistors, an initiator, and a discrete signal from the guidance package to disable the destruct system before the docking procedure. This discrete will be issued shortly after achieving the rendezvous orbit. The discrete will lock out the ISDS system via the enable/disable switch and simultaneously safe the initiator. Safing the initiator opens the positive and negative inputs to the bridgewires, shorts each bridgewire, and rotates a segment of the explosive train 90 degrees. With the explosive train interrupted, the destruct wafers would not detonate even if the destruct cartridge fired.

The destruct system of both transtages will be safed before rendezvousing with a manned vehicle. This is done to prevent a destruct function from occurring in the event of a potential difference between the rendezvousing vehicles. When the destruct system is safed, the only way of firing the wafer charges is to pass an arc directly on them. This will not occur since the arc (if any) will be located at the mating surfaces of the two rendezvousing vehicles.

B. CIRCUMLUNAR TRANSFER VEHICLE CONFIGURATION

The circumlunar transfer vehicle configuration is shown in Figure II-8. This vehicle consists of transtage 2, the target docking adapter, and the Gemini capsule. As previously described, transtage 2 has been stripped of all unnecessary inert weight and a maximum propellant load has been provided to stay within the maximum Titan IIIC payload capability of 26,205 pounds for a 100-nautical-mile orbit injection (as discussed in Chapter II, Section C).

Either as soon as the Gemini capsule is rigidly docked to Titan transtage 2, or just before injection into the circumlunar flight path, transtage 1 is separated from the second transtage propulsion module, taking the interstage skirt with it. The two transtages are separated by backing the Gemini away using the Gemini reverse thrust jets. This also provides the acceleration needed to settle the propellants in the tanks so the transtage 2 engines can be started. On engine ignition, two possible problems have been investigated. One is the engine starting transient loads (thrust peaking, misalignments, unsymmetrical thrust of the two engines, etc) that produce bending moments on the structure, particularly at the docking mechanism attachment and the Gemini nose structure. The second possible problem is the stability of the flight control system.

The interface between the Gemini capsule and transtage 2 is designed for utmost simplicity. Hardwire connections between the two bodies are provided by the standard Gemini umbilical connector, which provides for a maximum of nine wires. Also, a panel, similar to that proposed for the Agena system, is located on the target docking adapter to provide the astronauts with visual data on transtage 2 systems during checkout and subsequent engine firing of this stage. These data are also transmitted to the ground station after transtage 1 separation by a small VHF telemetry transmitter and antenna mounted on the transtage 2 structure. A detailed discussion of the interface definition is given in the following paragraphs of this section.

When the transtage 2 engines are ignited, the Gemini spacecraft will be accelerated to the proper velocity for injection into the circumlunar trajectory. At start of transtage 2 engine burning, the longitudinal acceleration will be approximately 0.6 g and at engine burnout it will be approximately 5 g. The acceleration will tend to pull the astronauts away from their seats and they will be restrained by their seat belt harness.

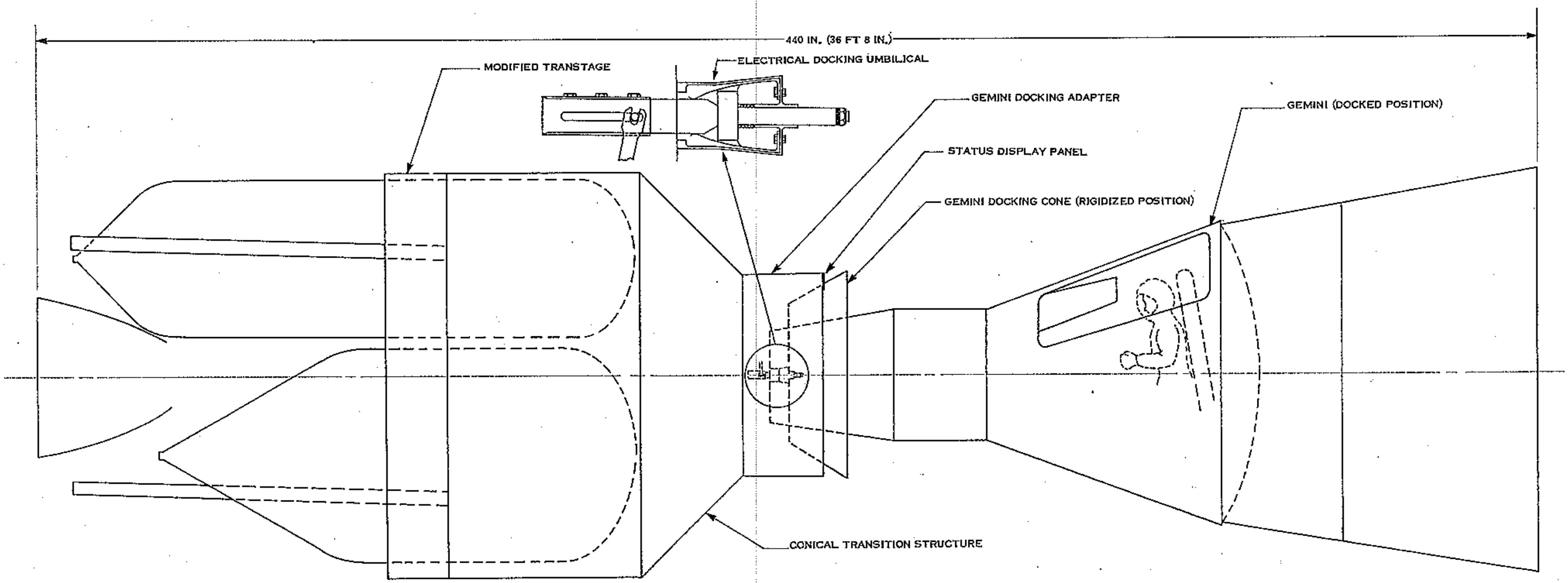


FIGURE II-8 CIRCUMLUNAR TRANSFER VEHICLE

The transtage 2 engine burn time is approximately 400 seconds, and, after burnout, the Gemini spacecraft will be separated from the transtage by the Gemini latch release mechanism in the capsule's nose. The astronauts will then turn the spacecraft around and use the OAMS package for midcourse velocity corrections.

1. Structural Dynamic Loads Analysis

To size the engine ignition loads on the McDonnell docking adapter and to determine the effects of the nonlinearities of the adapter structure on the stability of the Circumlunar Transfer Vehicle, the transient elastic response of the coupled structure-autopilot system to the transtage thrust buildup was calculated.

The mathematical model used for the analysis consists of two rigid-body masses, representing the Gemini and transtage, coupled laterally by the nonlinear bending stiffness of the adapter as determined by McDonnell test data. These data define the bending stiffness as a function of both applied bending moment and axial load. To account for this situation, the model also includes a representation of the configuration coupled by the axial adapter stiffness. Although axial stiffness is also load-dependent, it is approximated as being a constant equal in value to the axial spring rate associated with steady-state acceleration after ignition. This structural representation is coupled to the autopilot through the lateral component of thrust as determined by the engine angle demanded by the autopilot in response to angular displacement feedback from the Gemini and angular rate feedback from both Gemini and the transtage.

Excitation of the structure-autopilot system is provided by an initial attitude error of half a degree and two transient thrust buildup definitions for each transtage engine (Fig. II-9). The total thrust is applied to the axial model. The lateral component of total thrust, as determined by engine angle, and the thrust differential apply a moment about the center of gravity of the circumlunar vehicle. The transient response is calculated by a numerical integration scheme in which the load dependency of the bending spring and the quantizer as well as the sample, hold, and delay functions of the autopilot are accounted for.

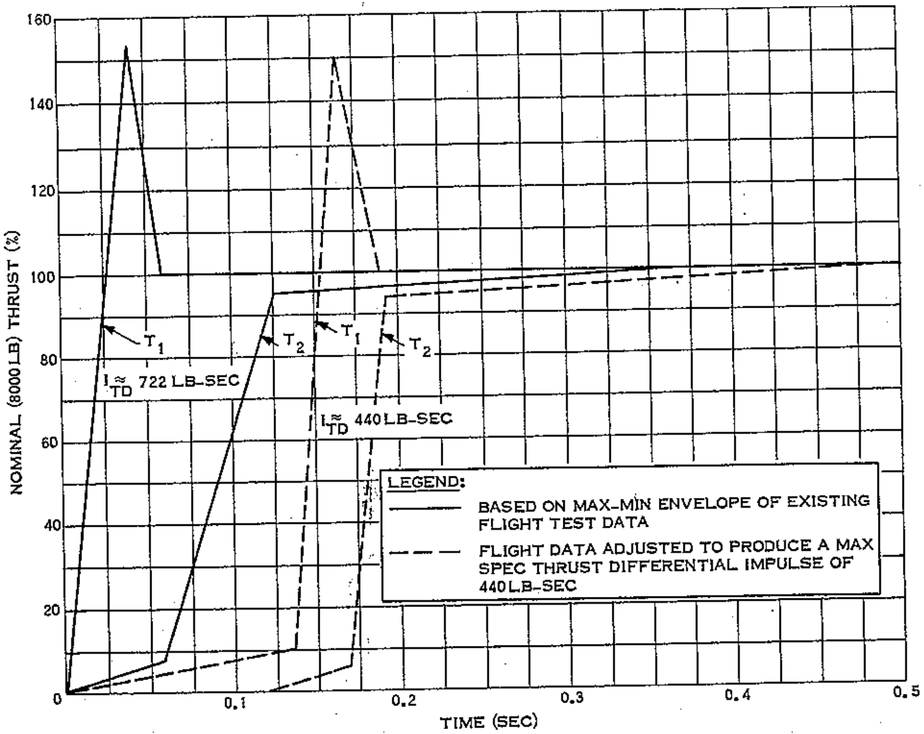


Figure II-9 Transient Thrust Buildup

The results of the analysis (Fig. II-10 and II-11) show the system stable in the rigid-body and fundamental elastic modes for both thrust definitions. The thrust buildup transient producing a 722 lb-sec differential impulse slightly exceeds the 97,500 in.-lb load capability of the adapter. This differential impulse is extremely conservative because the maximum and minimum thrust rise rates are based on a compilation of all the transtage engine flight thrust buildup data. These data include variations in engine inlet conditions, pressure and propellant temperature, that would affect the thrust transient during buildup. In the actual case, both engines would have the same inlet conditions, which would then show a smaller differential impulse than the 722 lb-sec. The 722 lb-sec case is therefore included in this report for reference only. The 440 lb-sec thrust differential generated by the second buildup considered produces an adapter bending moment of only 68,000 in-lb (Fig. II-11). Aerojet General Corporation and Martin Company have agreed (Spec No. IFS-T-3-13004) that the 440 lb-sec impulse is a realistic upper limit to be used as a specification value for the maximum design thrust differential for the transtage engines. Therefore, it is concluded that the target docking adapter is structurally adequate for this mission.

2. Flight Control System Analysis

The flight control problem involved when the transtage propulsion module and Gemini vehicle are attached through the docking adapter has been preliminarily investigated. Bending data were obtained by assuming two rigid bodies coupled by a flexible hinge (the docking adapter). Two slosh masses were attached to the propulsion module member. Data on the docking adapter stiffness and damping were obtained from McDonnell Aircraft Corporation. A linear analysis about the zero-moment hinge condition was conducted at full, one-half full, and empty conditions for the propulsion module.

On the basis of this simplified analysis, we recommend that the flight control system contain a rate gyro package aft of the docking adapter that will provide one gain change, and attitude plus attitude rate information from the Gemini, with filters on each sensor output. This system is similar to that in Stage II of the Titan III.

The McDonnell docking adapter data show a nonlinear spring rate versus moment loading. Under load the spring rate decreases, creating a less stable situation. The linear analysis described above and elaborated on in Appendix A did not include the nonlinear effects.

As part of the loads analysis, a transient response simulation of the transtage 2/Gemini configuration was developed. This simulation included the nonlinear effects of moment and axial loading on the docking adapter, transtage differential thrust buildup, actuator saturation effects, and guidance computer sample-hold-delay and quantizer effects. This simulation showed that the system is stable under worst case startup conditions. It can be concluded that stability margins with the transtage 2 rate gyro package are sufficient to allow for the nonlinear effects.

3. Transtage 2/Gemini Interface

In general, transtage 2 (TS-2) will interface with the Gemini spacecraft via the McDonnell Aircraft Corporation target docking adapter (TDA). This interface will comprise the electrical, mechanical, structural, and a portion of the functional interfaces between the two vehicles. Additional functional interfaces will exist through the TS-2 status display panel, telemetry system, and Gemini/TS-2 hardwire command link.

Electrical Interface - TS-2 will supply unregulated 28-vdc power to the TDA for operation of the TDA rigidize mechanism and rendezvous lights. In addition, 28-vdc unregulated power will be supplied by TS-1 to the rendezvous radar transponder, which is located in the TS-1 forward skirt.

Structural/Mechanical Interface - The TS-2/TDA structural/ mechanical interface will be consistent with the present TDA design. Geometry will be maintained by the use of master tools furnished by McDonnell.

Functional Interface - Functional interfaces will exist in the flight control system, TDA rigidize system, TS-1 and TS-2 separation systems, TS-2 telemetry subsystem, and the TS-2 status display panel.

Flight Control System - Demodulated pitch, yaw, and roll error signals will be supplied by the Gemini IGS to the TS-2 flight control system autopilot for processing and for engine control. In addition, these signals will be differentiated and mixed with TS-2 rate gyro signals for vehicle stabilization.

TS-2 engine command to fire will be furnished by Gemini as a 28-vdc signal to the engine ignition relays. Engine shutdown will be commanded by removal of this signal (either guidance or manual shutdown).

TDA Rigidize System (Undock) - A 28-vdc signal (manual) from the Gemini spacecraft will activate a TS-2 relay to supply power to the TDA rigidize system for releasing the spacecraft. In addition, TDA status will be displayed on the TS-2 status display panel.

TS-1/TS-2 Separation System - TS-1 and TS-2 separation will be commanded by a 28-vdc signal from the spacecraft to activate separation system relays.

TS-2 Telemetry - The TS-2 telemetry system will form a functional interface with the spacecraft via the ground communications link. TS-2 critical systems functions will be real-time monitored during boost, rendezvous, and docking and during TS-2 burn. Functions to be monitored will provide detailed backup to the status display panel and furnish time-integrated data relative to TS-2 power, propulsion, and flight control systems. The detailed measurement list is given in Table II-1.

<u>TS-2 Status Display Panel</u> - The TS-2 status display panel will furnish the Gemini astronauts with a continuous visual display of TS-2 systems readiness and critical parameters monitoring. Physical mounting of the status display panel will be provided on the transtage forward skirt and will comply with all requirements for visibility and line of sight of the Gemini command astronaut. The following status lights will be included in the display:

1) TS-2 ready to dock - TDA unrigidized, TS-1 and 2 destruct system safed, etc;

2) Docking complete - TDA rigidized, continuity verified, first-phase engine ignition relays enabled, etc;

3) TS-2 checkout complete - Electric power, flight control and propulsion systems within tolerance;

4) TS-1 separation verify - Final-phase engine ignition relays enabled;

5) Warning light - Critical parameter out of tolerance, abort indicated.

The following analog displays will be provided:

1) Helium storage pressure;

2) Fuel tank pressure;

3) Oxidizer tank pressure;

4) Flight control hydraulic pressure.

C. LAUNCH VEHICLE PERFORMANCE

Preliminary evaluations of performance capabilities and requirements have been conducted for the rendezvous technique of accomplishing a circumlunar mission. The parking orbit was nominally circular at an altitude of 100 nautical miles and an inclination of 28.9 degrees. Launch azimuths will be 84 degrees for both vehicles on a simultaneous launch and 96 degrees for the second launch vehicle if launched after the first orbit. The payload capability of the Gemini launch vehicle is assumed to be as high as 8500 pounds on the prescribed orbit. Primary emphasis in this study has been evaluation of the Titan IIIC launch vehicle to ensure performance compatibility with GLV under the most demanding conditions.

1. Titan IIIC Performance

Table II-8 presents the performance of the standard Titan IIIC for a specific set of conditions appropriate to the unmanned portion of the circumlunar mission. Also shown are exchange ratios for estimating the effects of alternative orbit altitudes and inclinations.

Table II-8 Titan IIIC Booster Performance

Conditions:	1) Due east launch
	2) 100-n-mi circular orbit (direct injection)
	3) Optimum transtage propellant load (approx 16,000 lb)
	4) 2-sigma minimum performance (3-sigma velocity margin estimated as RSS of 3% of individual stage velocity increments)
	5) Limit nominal weights with instrumentation removed
	6) Latest approved engine performance data
	7) Standard payload fairing jettisoned at approx 300,000 ft
Available Payload:	26,205 lb (including 70-lb retained payload fairing frame)
Exchange Ratios:	1) -25-lb payload/n-mi circular orbit altitude (average to 200 n mi)
	2) 2.5-lb payload/fps earth velocity component

The payload of the Titan IIIC will consist of a modified transtage with partially loaded propellant tanks. Table II-9 shows the correlation of booster performance with propellant load of the vehicle. The indicated value corresponds to about 306 pounds less than tank capacity.

Table II-9 Transtage 2 Performance

Available Gross Weight lb		26,205
Less Inert Weight lb		
Interstage Skirt	398	
Transtage 2 Burnout	3,212	
Engine Ablative Material	30	3,640
Available Transtage 2 Propellant lb		22,565

<u>Note</u>: Payload on lunar transfer trajectory = 8590 lb;

3-sigma minimum performance above park orbit. Velocity increment = 10,307 fps including 50-fps powered flight losses;

Exchange ratios:
1) -1-lb payload/lb transtage 2 burnout weight,
2) +0.52-lb payload/lb transtage 2 propellant load (up to tank capacity of 22,871 lb),
3) -1.82-lb payload/fps transfer velocity requirement.

Payload capability for the lunar transfer maneuver is presented parametrically in Figure II-12. For a specific set of assumptions, a value of 8590 pounds has been calculated. This performance is included in Table II-9 with the pertinent exchange ratios to facilitate interpretation.

The performance capabilities of Titan IIIC can be further improved, if necessary, by implementing special techniques and procedures. A practical maximum is represented by achieving fully loaded propellant tanks for transtage 2. This condition will correspond to a maximum lunar transfer payload of 8755 pounds. Booster performance improvement in excess of 306 pounds on park orbit (to fully load transtage 2 tanks) will produce small lunar payload gains if park orbit altitude can be raised. A more practical use of improved performance would be to provide launch azimuth flexibility and/or orbit turn capability to decrease launch window restrictions.

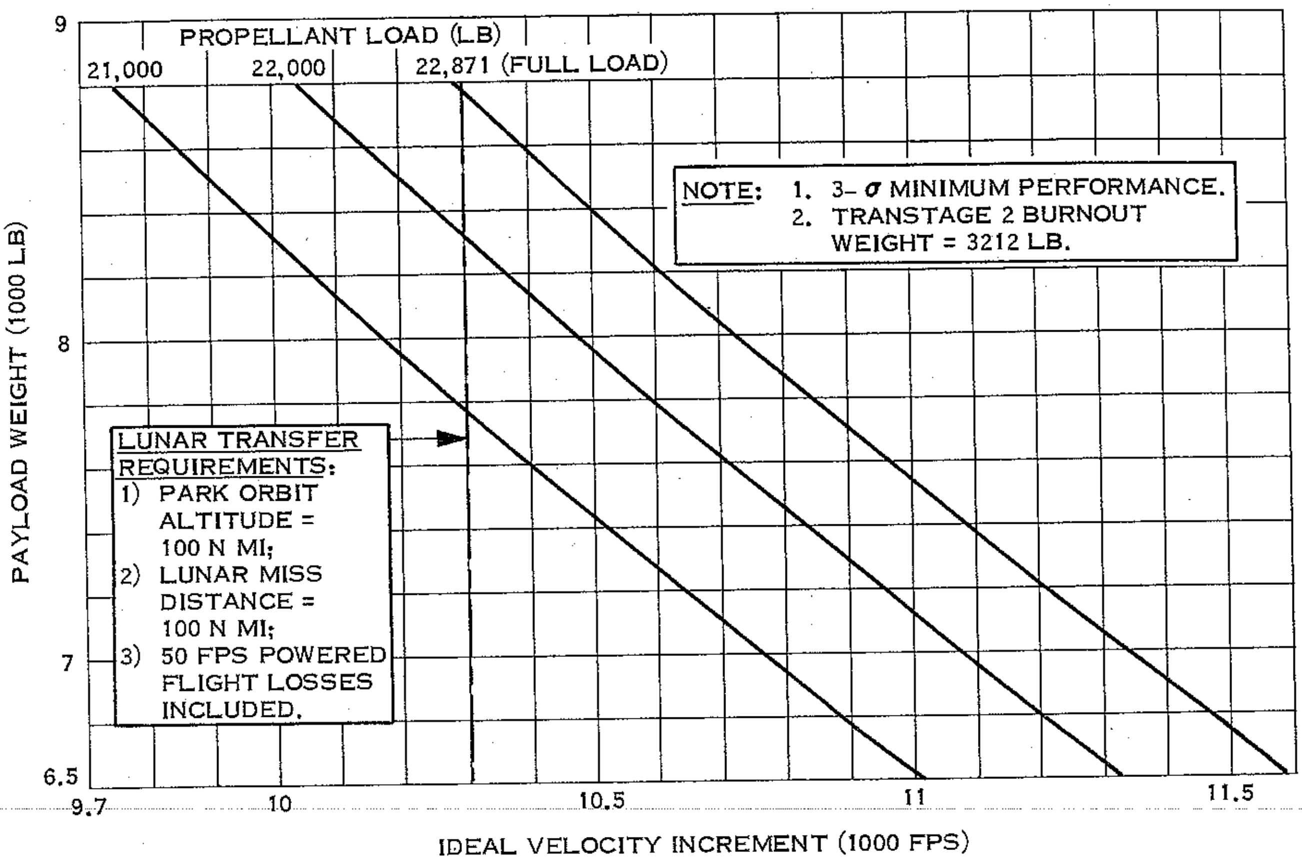


Figure II-12 Transtage 2 Performance

Methods of increasing the performance potential of Titan IIIC beyond that presented in Table II-8 include:

1) Optimized trajectory shaping (including elliptic ascent to park orbit);

2) Chilled propellants (about 45°F) in liquid stages;

3) Reduced ullage (increased propellant loads);

4) Selected vehicles with high-performance engines (based on engine acceptance test data).

2. Heat Shield Qua'ification Test

A heat shield qualification flight will be conducted with a standard instrumented Titan IIIC early in the circumlunar program. Targeting assumptions will be:

1) Reentry in the vicinity of Ascension Island;

2) Velocity = 36,000 fps (400,000-ft altitude);

3) Flight path angle = 6½ deg nominal (400,000-ft altitude);

4) Weight = 5000 lb (full-scale geometry).

The flight profile (Fig. II-13) will involve coast and restart of the transtage with the second powered phase directed downward. Burnout altitude will be restricted to above 700,000 feet to permit time for reorientation of the Gemini capsule.

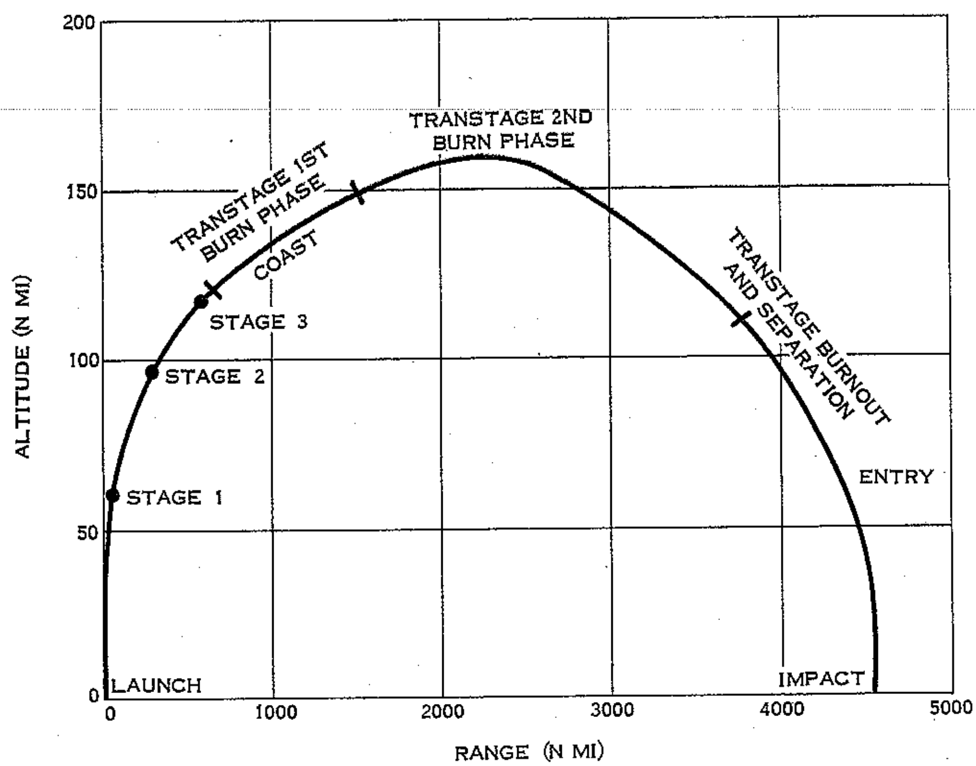


Figure II-13 Heat Shield Qualification Trajectory

Preliminary performance data indicate a test trajectory profile similar to that shown in Figure II-13 can be achieved. The payload weights associated with the profile are estimated to be around 4500 pounds.

Another approach to the test trajectory would be to inject into orbit and then to bend the path downward as the final velocity increment (~10,500 fps) is applied, thus achieving the desired reentry conditions. Impact in this case would be determined by the time in orbit. Payload weights would be greater than those associated with a path shaped to impact at Ascension.

3. Direct-Flight Technique

Some preliminary investigations have been conducted for the direct-flight technique of accomplishing the circumlunar mission with Titan IIIC. The two-sigma payload capability is estimated at 7450 pounds with the following qualifications:

1) Transtage control module guidance and associated systems removed and guidance functions provided by Gemini capsule;

2) Trajectory shaping optimized (no park orbit, one launch opportunity per month);

3) No chilled propellants, selected engines, etc.

4. Titan IIIA Countdown Data

The following data indicate the capability of the SSLV to perform a scheduled countdown satisfactorily.

First Flight Test, SSLV 2

First Launch Attempt - 31 Aug 64		Est
Countdown start	T - 215	0725
Manual hold to allow completion of other work in process	T - 30	1030
Count resumed	T - 30	1100
Hold to adjust CMG reset circuitry	T - 0:31	1129
Count resumed	T - 0:31	1131
Range Azusa no-go	T - 0:18	
Hold for Azusa no-go and reset to T - 3	T - 0:03	1132
Count resumed	T - 3	1202

Range Azusa no-go	T - 0:27	
Automatic hold due to propellant pressure	T - 0:09	
Recycle to T - 3 and extend hold	T - 3	1210
Hardware replaced and leaks repaired		1735
Launch cancelled due to unfavorable atmospheric conditions		1750

Launch - 1 Sep 64

Countdown start	T - 215	0625
Liftoff	T - 0	1000

Second Flight Test, SSLV 1

First Launch Attempt - 20 Nov 64

Countdown start	T - 215	0626:36
Launch scrubbed*	T - 21	1610

*1) Tracking and flight safety
2) Encoder indicating erratic ambients
3) Low Stage II oxidizer pressure
4) Decaying Stage III fuel tank pressure

Launch - 10 Dec 64

Countdown start	T - 215	0810
Manual hold for check - no problem	T - 0:31	1144
Count resumed (reset)	T - 3	1149
Liftoff	T - 0	1152 (+7 min)

Third Flight Test, SSLV 3

Launch - 11 Feb 65

		Est
Countdown start	T - 215	0625
Hold for command control receiver malfunction	T - 155	0725
Resume count	T - 155	0944
Liftoff	T - 0	1219 (+2 hr 19 min)

Fourth Flight Test, SSLV 6

Launch - 6 May 65

Countdown start	T - 215	0625
Liftoff	T - 0	1000

Summary

Four flights to date.
Two scrubbed on first launch attempt.

On second launch attempt of these two:

1) One countdown went straight through to liftoff;
2) One countdown was held manually for a check that revealed no problem. The count was reset to T - 3 and proceeded normally through liftoff. Total lost time was seven minutes.

The third launch went through countdown with a delay of two hours and 19 minutes resulting from a command control receiver (GFP) malfunction.

The fourth launch went straight through to liftoff.

5. Gemini Launch Vehicle (GLV) Countdown Data

The following data indicate the capability of the GLV to perform a scheduled countdown satisfactorily.

First Flight

Launch - 8 Apr 64

Scheduled 5-hour countdown - no holds.

Second Flight

First Launch Attempt - 9 Dec 64

Scheduled 7-hour countdown - no holds.

Launch killed just after engine ignition due to hydraulic servovalve housing rupture.

Launch - 19 Feb 65

Scheduled 7-hour countdown - one 3-minute hold for spacecraft check.

Third Flight

Launch - 23 Mar 65

Scheduled 7-hour countdown - one 24-minute hold to rectify pressure instrumentation leak in oxidizer discharge line.

Fourth Flight

Launch - 3 Jun 65

Scheduled 7-hour countdown - 1 hour 16 minute hold due to erector cable slack.

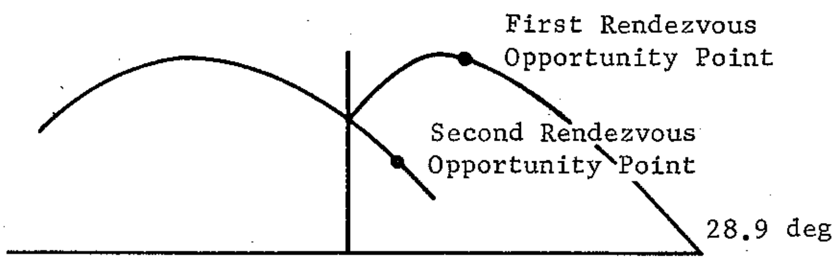
D. EARTH-ORBIT RENDEZVOUS OPERATIONS

The ascent-to-rendezvous technique will be selected to provide multiple launch opportunities for rendezvous and to maximize the performance capability of the Titan IIIC/GLV for the circumlunar mission. The altitude and inclination of the rendezvous orbit will be established during the mission studies and will be based on integration and tradeoff analysis of all Gemini mission requirements, i.e., ground tracking for guidance system updating, recall, recovery, etc. Preliminary investigations of the mission requirements have indicated that combining the advantages of the simultaneous launch and establishing a rendezvous-compatible orbit provides the necessary conditions for rendezvous and injection into the translunar orbit. Details of the method are presented in the following subsections to establish gross operational procedures and the associated system requirements.

1. Ascent to Rendezvous

Combining the techniques of launching both the Titan IIIC and GLV simultaneously and launching the Titan IIIC so a rendezvous-compatible orbit is established provides a relatively simple method of establishing multiple launch opportunities for rendezvous. With this technique, the Titan IIIC is launched into a 28.9-degree orbit (launch azimuth about 84 degrees) so the transtage will overfly the launch site at the start of the second orbit. Nominally, the GLV would be launched about four minutes after the Titan IIIC is launched, and rendezvous would occur at the Titan IIIC orbit injection point. If the GLV launch does not take place, then a second opportunity will occur about one orbit period later, when the transtage overflies the launch site. In the second case, the GLV is launched at an azimuth of 96 degrees and rendezvous occurs at the Gemini orbit injection point.

For this procedure, both the Titan IIIC and GLV are counted down to liftoff simultaneously. The Titan IIIC is counted down to T - 30 seconds and held. The GLV is counted down to T - six minutes and held. The optimum hold time, the T - 30 seconds to liftoff, will be established during the study. The GLV countdown is restarted two minutes before the Titan IIIC launch time, and the liftoff occurs four minutes after the Titan IIIC launch. In the event that both vehicles do not complete the countdown within the allowed time, the launch operations are rescheduled for the next day. If a malfunction is detected during the last four minutes of the GLV countdown (after the Titan IIIC has been launched), the GLV launch sequence is recycled to T - six minutes. The next launch opportunity occurs about 90 minutes later when the transtage overflies the launch site. This provides about 80 minutes for repairs and to reload the GLV computer. The launch azimuth for the second opportunity will be about 96 degrees and the rendezvous will occur at the Gemini orbit injection point. A schematic of the ascent technique is shown below.



2. Error Analysis

The launch trajectory dispersions of the Titan IIIC and GLV lead to a separation distance at the nominal rendezvous point. A brief analysis of the launch dispersions was completed to indicate the magnitude of the separation distances. The three-sigma values of the miss distances are shown in Table II-10 for both the simultaneous launch technique and for launching one orbit later after the Titan IIIC achieves orbit.

Table II-10 Intercept Errors

	Simultaneous Launch (± n mi)	Launch One Orbit Later (± n mi)
In-Track	22	32.0
Cross-Track	0.6	1.0
Altitude	0.4	5.0

3. Closure Maneuvers

Although the launch dispersions shown in Table II-10 result in significantly severe rendezvous closure maneuvers, the distances are within the radar range capability of Gemini so the search pattern and tracking should present no particular difficulties. To reduce the energy requirements for rendezvous to a reasonable level, a three-hour time period will be allowed for closure and docking. The major displacements will be obtained by injecting the Gemini into an elliptical phasing orbit. At the end of the phasing orbit, about 90 minutes, the Gemini proceeds to dock with the transtage. The velocity requirements to complete the maneuvers are shown in Table II-11. The data are shown for the worst-case conditions and represent, for the different launch techniques, a summation of the RSS of the injection velocity dispersion and the transfer maneuver velocity requirements. The data do not include the propellant required to execute the docking maneuvers.

Table II-11 Closure Maneuvers

Launch	ΔV (fps)	Perigee/Apogee Altitude Change (n mi)
Simultaneous	61	10
After One Orbit	75	14

4. Error Propagation to Translunar Injection

The launch dispersions also affect the position of the Gemini/transtage at the normal time for injecting the Gemini into the circumlunar trajectory. The nominal injection time is about six hours after the Titan IIIC is injected into orbit. Since the GLV must rendezvous with the Titan IIIC transtage, the position uncertainties at the injection time will be those due to Titan IIIC injection dispersions. The dispersions shown in Table II-12 are the dispersion in track RSS with the injection time dispersions.

Table II-12 Injection Dispersions

	Nautical Miles
In-Track	50 (0.83 deg of central arc)
Cross-Track	1
Altitude	5

During the rendezvous and docking maneuvers, the Gemini and transtage will complete at least three orbits within the ground tracking network. The use of these data could reduce the position uncertainty to about three miles in track. The problems of collecting, reducing, and transmitting the data to update the Gemini guidance system will be analyzed during the study period.

E. MISSION PROFILE CONSIDERATIONS

The principal phases of the circumlunar mission using earth-orbit rendezvous include launch and rendevous, injection along a translunar trajectory, translunar and transearth midcourse corrections, and earth atmosphere reentry. Each of these mission phases is discussed separately, and the conclusions reached in discussing the separate mission phases are integrated with the planning for a circumlunar mission subject to typical mission objectives and constraints. Finally, an abort rationale is discussed.

The use of a circumlunar ballistic trajectory is assumed for all missions considered. In addition, passing around the moon for any reason other than trajectory shaping or mission safety will be considered secondary. The only exception to this restriction is that lighting conditions at the moon will be considered in the mission planning. Finally, it is assumed that only two vehicles will be launched from earth and joined before injection into the translunar trajectory -- the manned capsule (modified Gemini) and a propulsion stage (transtage).

The factors to be considered in the mission planning comprise vehicle capability and operational mission constraints. The transtage must furnish out-of-atmosphere propulsive energy, and the spacecraft must incorporate in-atmosphere maneuverability and withstand entry heating. The operational mission constraints include tracking requirements (liftoff to landing), landing site selection, lighting conditions at departure, lunar encounter, earth landing, mission safety, and abort considerations. Although such factors as these will be discussed in detail in the mission planning section, their significance must be recognized during discussion of the separate mission phases.

1. Launch-and-Rendezvous Phase

The launch-and-rendezvous phase includes separate launches of two vehicles, their rendezvous, and preparation for translunar injection. Launch vehicle performance capabilities are presented in Section C and the rendezvous operations are discussed in detail in Section D. The ascent technique is to launch directly to the rendezvous point, acquire the rendezvous target, and maneuver to close. The orbital parameters are chosen to provide two launch opportunities one orbit period apart. With one orbit period allowed for a launch window, one orbit period for phasing, and three hours for closure and docking, a nominal six hours is needed between initial launch and translunar injection. The consequences of not arriving at the correct injection point at the correct time are discussed in Subsection 2, Translunar Injection Phase.

2. Translunar Injection Phase

The plane of symmetry for the lunar mission is the moon's orbital plane (MOP). For the most part, the basic data presented are referenced to the MOP. The symbols used to define the trajectory geometry near earth and near the moon are shown in Figures II-14 and II-15, respectively.

The lunar flyby program schedule allows 18 months from go-ahead to first launch of the configurations for rendezvous and escape into lunar orbit. Thus, the lunar flyby flights are readily correlated with the Gemini and Apollo programs, since the major rendezvous and docking missions for Gemini will have been concluded, and the Apollo rendezvous, docking, and maneuver missions and lunar missions will be programed later.

The design and build sequence for the lightweight transtage is the pacing schedule item. A total of seven months from go-ahead is allowed for engineering design. This time is ample on a normal working basis. On Titan III, major structures release for the total structures occurred 7½ months after Phase II go-ahead.

Procurement of material and long-lead items is scheduled to start one month after go-ahead, allowing four months for acquisition of materials and items required for initial build. The propulsion unit is required for final assembly 12 months after go-ahead. This is consistent with the 14-month lead time for delivery of a Titan III during an existing production program.

The controlling fabrication detail is the honeycomb skin. Within the scheduled procurement and fabrication spans there is a four-month allowance for procuring and tooling skin materials and another 1½ months for fabricating structural test items.

Of the 9½ months shown for final assembly/checkout/test/ship, the last two are for checkout/test and pack and ship. This leaves 7½ months for final assembly and installation of test items and the first flight article. This time is ample for establishing the final bonding techniques and process plans for fabricating the lightweight transtage structure.

By comparison, the build plan for a Titan III standard transtage allows two months for detail fabrication and another 3½ months for tank fabrication, final assembly, and general checkout and tests prior to the Titan III vertical tests. In summary, approximately double this fabrication and test span is available for the lightweight transtage initial build.

Delivery of all articles to ETR 1½ to two months prior to flight is comparable to present flight operations requirements.

For the Gemini circumlunar flyby program, we have made the following assumptions:

1) Gemini A hardware, modified as required, will be reallocated by NASA for the heat shield qualification flight and two manned flights;

2) Standard GLVs for the two manned flights will be provided by NASA under existing contracts;

3) GLV launch operations will be performed under existing NASA contracts;

4) Standard Titan IIIC boosters will be used for boosting,

   a) Lightweight transtage propulsion segments for Gemini rendezvous,

   b) A modified Gemini for heat shield qualification tests;

5) The present Gemini eject abort system will be retained for the GLV/Gemini lunar flyby launcher;

6) The Titan IIIC/Gemini interface adapter for the heat shield qualification flight will be provided by Martin;

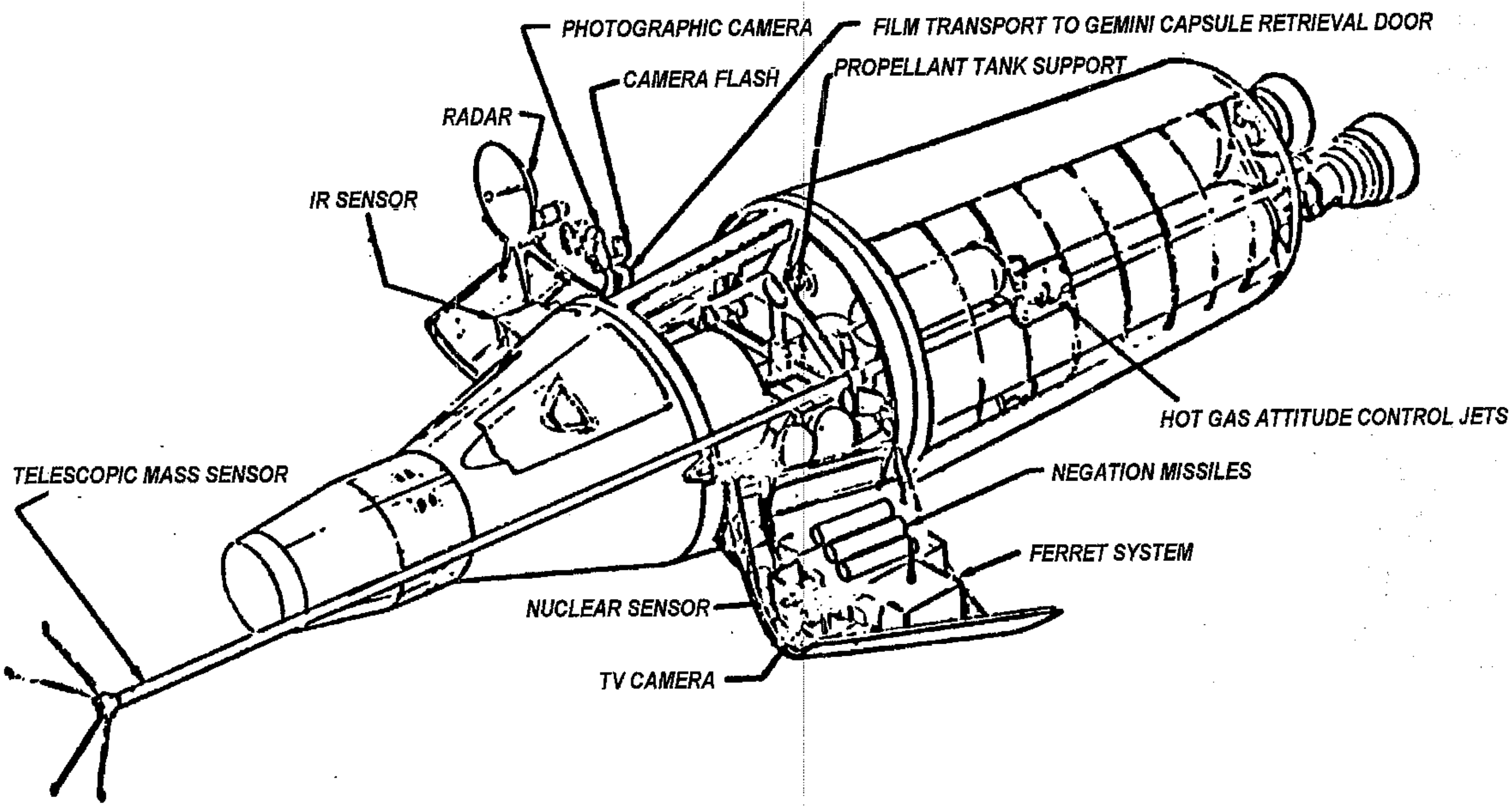
7) The lightweight transtage will utilize the existing Titan III transtage propulsion unit, with a new interstage skirt and forward section accomodating the Gemini docking mechanism;

8) McDonnell Aircraft Corporation will provide Gemini ground equipment required at Pad 40 for heat shield qualification flight (if finalized schedules allow, Gemini ground equipment may be retained at Pad 40 from MOL EFT program heat shield qualification flight);

9) The lunar flyby launch sequence will be the Titan IIIC with lightweight transtage payload first, and then the GLV/Gemini;

10) McDonnell Aircraft Corporation will supply Martin with the Gemini docking mechanism to be incorporated in the lightweight transtage structure.

Milestone activities related to the Gemini spectacular are presented in Figures III-1 and III-2.



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