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LAUNCH VEHICLE HANDBOOK

Compiled by
MELVYN SAVAGE

ANALYSIS AND ...

OFFICE OF LAUNCH VEHICLE PROGRAMS

NASA

SEP 25 1961

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
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LAUNCH VEHICLE HANDBOOK COMPILATION OF LAUNCH PERFORMANCE AND WEIGHT DATA FOR PRELIMINARY PLANNING PURPOSES

National Aeronautics and Space Administration
Washington D C

Aug 61

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N-101307

COMPILATION OF LAUNCH VEHICLE PERFORMANCE AND WEIGHT DATA FOR PRELIMINARY PLANNING PURPOSES

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ANALYSIS & REQUIREMENTS OFFICE

11 AUGUST 1961

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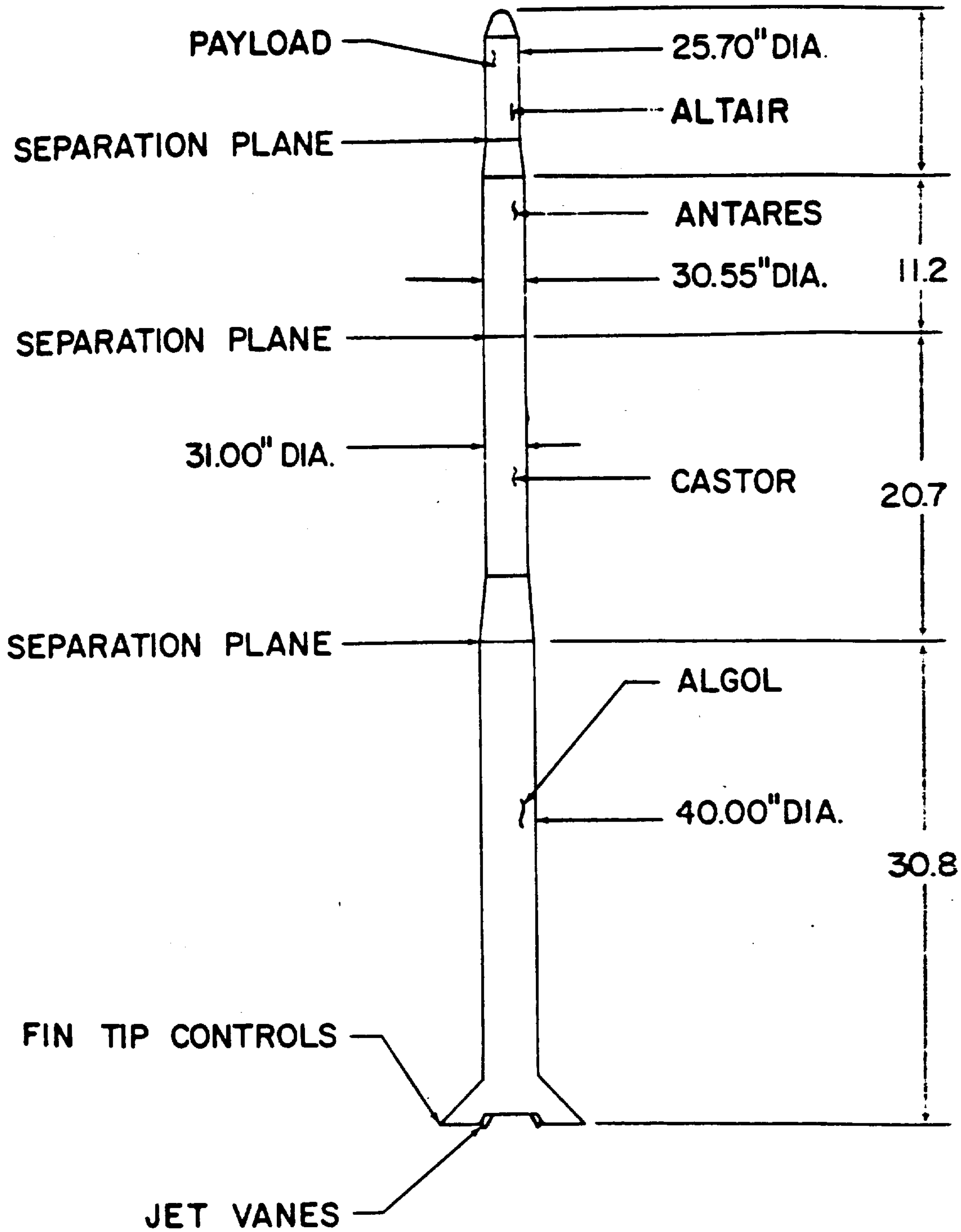
INTRODUCTION

The weight and payload data presented was obtained from the various vehicle system contractors. As such, it represents a variety of computational methods. Therefore direct comparisons of payload capabilities from one vehicle to another could be misleading as the degree of conservatism could vary.

The prime purpose in this compilation is to provide a source of information for preliminary planning purposes. Direct contact with the appropriate NASA vehicle program manager is essential before any firm spacecraft weight and trajectory is established.


PAYLOAD
25.70" DIA.
ALTAIR
SEPARATION PLANE
ANTARES
30.55" DIA.
11.2
SEPARATION PLANE
31.00" DIA.
CASTOR
20.7
SEPARATION PLANE
ALGOL
40.00" DIA.
30.8
FIN TIP CONTROLS
JET VANES

SCOUT B

Flight Programs

For nominal flights, the pitch gyro of the Scout will be torqued at rates which will produce a zero lift; that is, gravity turn, trajectory. This is done by first approximating as accurately as possible the pitch rate history associated with a desired controls-locked, no disturbance, zero-lift trajectory and then modifying this slightly to account for an inherent system lag. The basic pitch program is in reality a series of step functions of such magnitude and duration that the total area formed by them is equal to the total area under the pitch attitude rate curve, $\dot{\theta}$, of the expected trajectory. The magnitude and length of the step attitude rate functions are determined from a series of straight line slope approximations to the desired pitch attitude curve. Accuracy of the program is affected by winds, thrust misalignment, and inherent control system lags during the first stage and droop associated with the characteristic deadband of the second and third stage "on-off" control systems. Generally, however, the difference between the programmed attitude and the desired flight-path angle are small except at launch with these differences tending to diminish asymptotically with time as the velocity vector gradually tends to aline itself with the thrust vector. No attempt is made to adjust the pitch program for winds and thrust misalignment at this time.

In programming a Scout trajectory, certain restrictions must be adhered to. First, the Scout must be launched at elevation angles of 78° or greater if the aerodynamic heating encountered is not to become too severe. Secondly, the vehicle cannot be programmed to fly a trajectory much different from a nominal zero-lift trajectory due to structural limitations during first-stage flight and to the fact that the maximum available control power imposes certain limitations on the permissible deviations from a zero- lift flight path. At present no provision is made for roll or yaw maneuvers although such maneuvers do seem feasible with only nominal changes to the guidance system.

During actual flights, Scouts A, B, and C rocket motors will be fired according to the following sequences. The first-stage rocket motor will be fired from the ground level. After its burnout, the first-stage motor will remain attached to the vehicle until an altitude of 130,000 feet is reached. The coast to 130,000 feet is done to effectively cancel the aerodynamic instability of the remaining stages and to relieve heating loads that could be incurred by igniting the second stage at lower altitudes. The second stage is ignited by means of a programmer and the first stage is immediately blast separated. Following second-stage burnout, the vehicle coasts for a minimum period of 5 seconds after which time the programmer ignites the third stage and the second-stage burned out motor is blast separated. At this time the drag and heat fairings on third and fourth stages are ejected. Thus far the ascent, as previously described, is identical for both probe and orbital missions; however, after third-stage burnout, either of two procedures are followed, depending on whether the payload is an orbiter, or a probe. For probe trajectories, the fourth stage is spun up, ignited, and blast separated from the third stage 5 seconds after third-stage burnout. If there is a fifth stage, it is ignited immediately after fourth-stage nominal burnout time. For orbiters, the entire third stage with its hydrogen peroxide control system still operating coasts to the apogee of the ascent orbit, aligning itself to a programmed altitude for fourth-stage ignition. At this time the final stage is spun up, the fourth stage ignites, and the third-stage motor is blast separated. If there is a fifth stage, it is ignited immediately after fourth-stage burnout. A typical four-stage nominal ascent trajectory is presented in the table below.

SCOUT ORBITAL-ASCENT TRAJECTORY

PAYLOAD = 100 POUNDS

	1 BO.	2 IGN.	2 BO.	3 IGN.	3 BO.	4 IGN.	4BO.
t sec	41.3	75.3	115.0	120.0	159.4	479.2	520.2
V fps	4,589	3,932	10,371	10,312	17,648	15,974	27,036
h N.M.	9.02	21.40	41.10	44.30	74.10	230.40	230.40
R N.M.	6.42	19.60	56.40	63.20	136.40	886.75	1006.85
γ (Earth relative) deg	48.9	35.3	25.1	25.5	20.4	1.0	0

Guidance and Control System

Guidance of the Scout vehicle is obtained by a conventional three-axis "strapped down" gyro system combined with a three-axis control system. In this system, guidance is confined to the pitch plane only, with azimuth and roll orientation maintained during flight at essentially the initial reference attitudes established at the time of launch. The guidance system is based on three body mounted miniature integrating gyros (MIG's), three rate gyros (GNATS), and the pitch axis programmer.

In the lift-off configuration, the vehicle is aerodynamically stable. A proportional control system featuring a combination of jet vanes and aerodynamic tip control surfaces operated by hydraulic servo actuators is used to control the vehicle throughout the entire first-stage burning period. These controls operate in pairs for pitch and yaw control. The jet vanes provide the majority of the control force during the thrusting phase and the aerodynamic tip controls provide all the control force during the coasting phase following burnout of the first stage.

After separation of the first stage all succeeding stages are aerodynamically unstable. Because of this, the second stage is not separated until 130,000 feet altitude is reached. This reduces the effect of the unstable aerodynamic air loads on the control system. Control during second-stage burning is provided by hydrogen peroxide reaction jets which are operated as an on-off system within a small deadband. Second-stage nominal deadband values are as follows:

	Position	Rate
Pitch	$\pm 0.8^{\circ}$	$\pm 2^{\circ}$/sec
Yaw	$\pm 0.9^{\circ}$	$\pm 2.25^{\circ}$/sec
Roll	$\pm 0.2^{\circ}$	$\pm 5^{\circ}$/sec

Control of the third stage is also provided by hydrogen peroxide reaction jets. Two modes of control operation are provided. The thrusting phase controls consist of four 44-pound reaction jets for pitch and yaw control and four 14-pound jets for roll control. After burnout, when possible thrust-induced upsetting moments are zero, a switch is made to the coasting phase controls by a programed signal from the timer which also turns off the pitch and yaw jets. Yaw and roll control during coast is combined in the four roll jets that have been reduced from 14 pounds to a level of approximately 3 pounds by means of a restrictor orifice. Pitch control is maintained with a switch in pair of 2-pound jets. The same roll and yaw deadbands are utilized for the second- and third-stage coast periods; however, the pitch deadband is reduced during third-stage coast to 0.3° position and 0.75° per second.

Guidance and Control System (Contd)

The fourth stage which includes the payload package does not have an active guidance and control system. It receives the proper spatial orientation from the control exerted by the first three stages after which it is spin stabilized by a combination of two or four 40 pound-sec impulse spin motors, as required, which are mounted tangentially in the skirt at the base of the fourth stage. Scout A, B, and C guidance systems are identical.

Scout A, B, C

The following figures and charts portray information for Scout B only. Scout A will be dropped and Scout B will take its place in 1962. Scout C consists of Scout B plus a fifth stage.

SCOUT B

STAGE CHARACTERISTICS

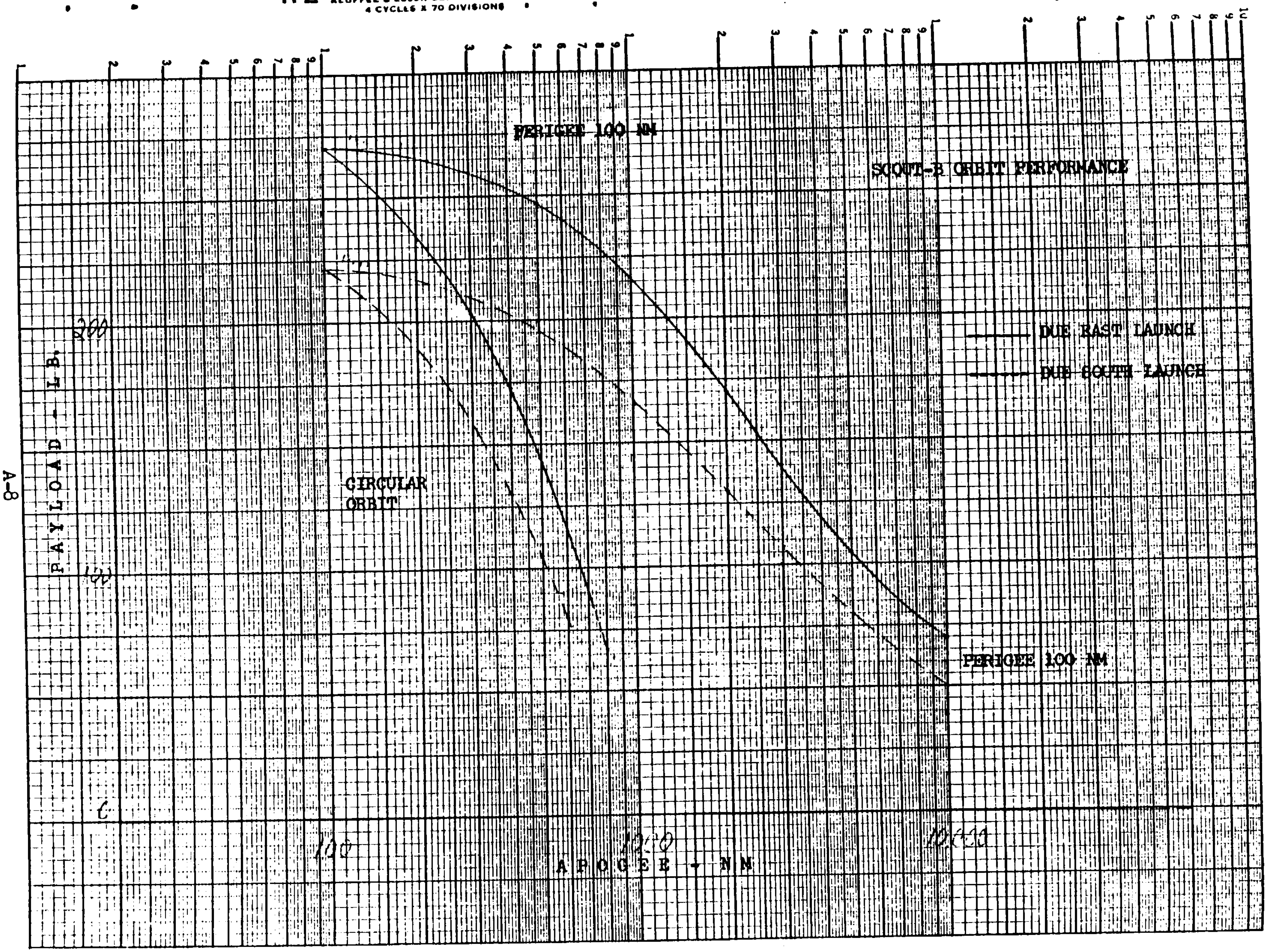
	1	2	3	4
1. Stage Designation	1	2	3	4
2. Lightoff weight (less payload)	36,838	12,993.5	3,410.2	643.2
3. Loaded stage weight	23,844.5	9,583.3	2767	643.2
4. Propellant weight (usable)				
a. Fuel weight	18,998	7320	2084	456
b. Oxidizer weight	N/A	N/A	N/A	N/A
5. Propellant weight (residual)				
a. Usable contingency	N/A	N/A	N/A	N/A
Unusable contingency	N/A	N/A	N/A	N/A
6. Stage burnout weight	4484.2	2044.7	660	178.2
7. SBW + jettisonable weight	4484.2	2044.7	698	293.2
8. Stage propellant fraction $\frac{\text{Propellant weight}}{\text{Propellant + structure wt.}}$	0.802	0 .771	0.753	0.709

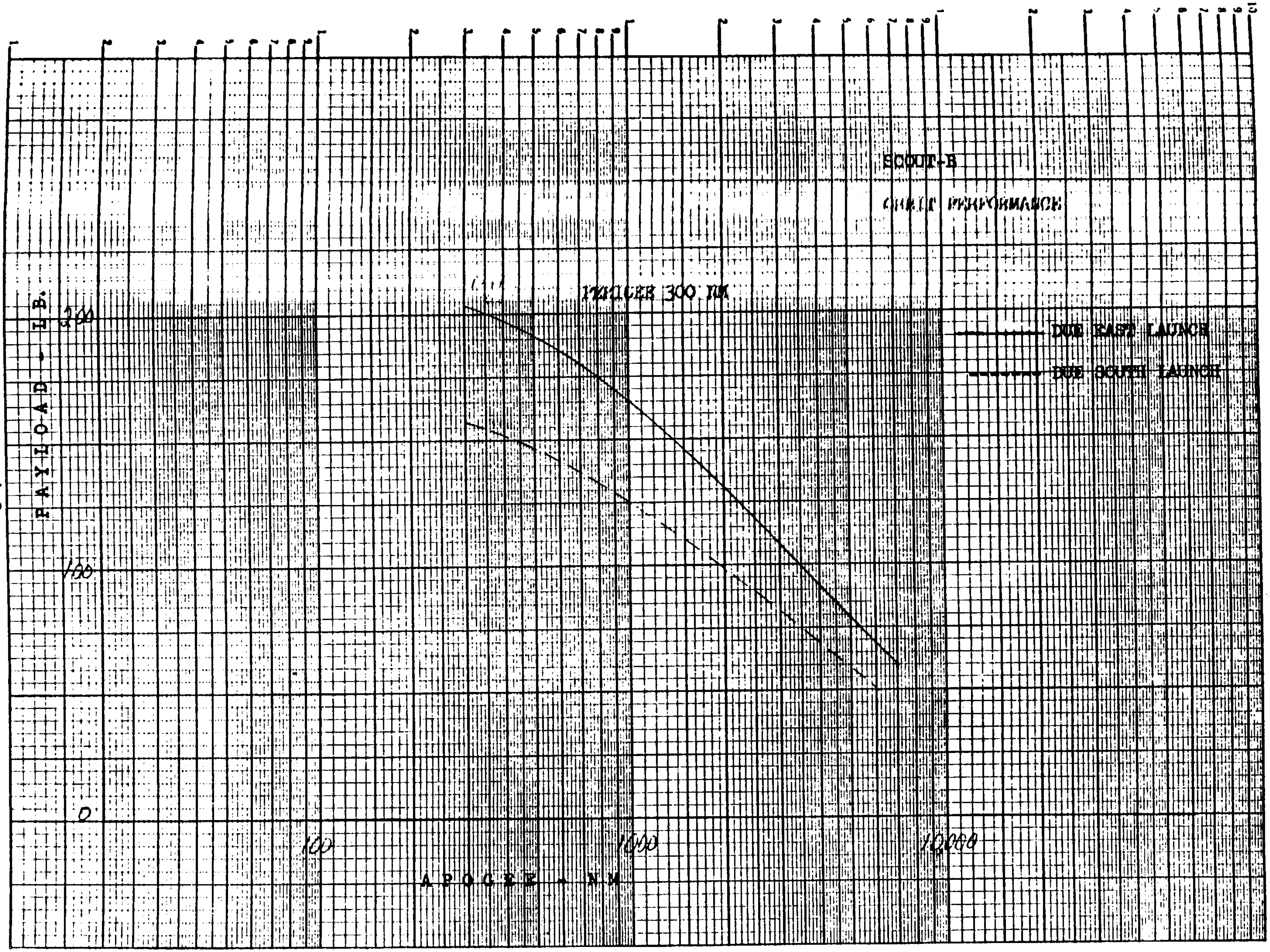
SCOUT B

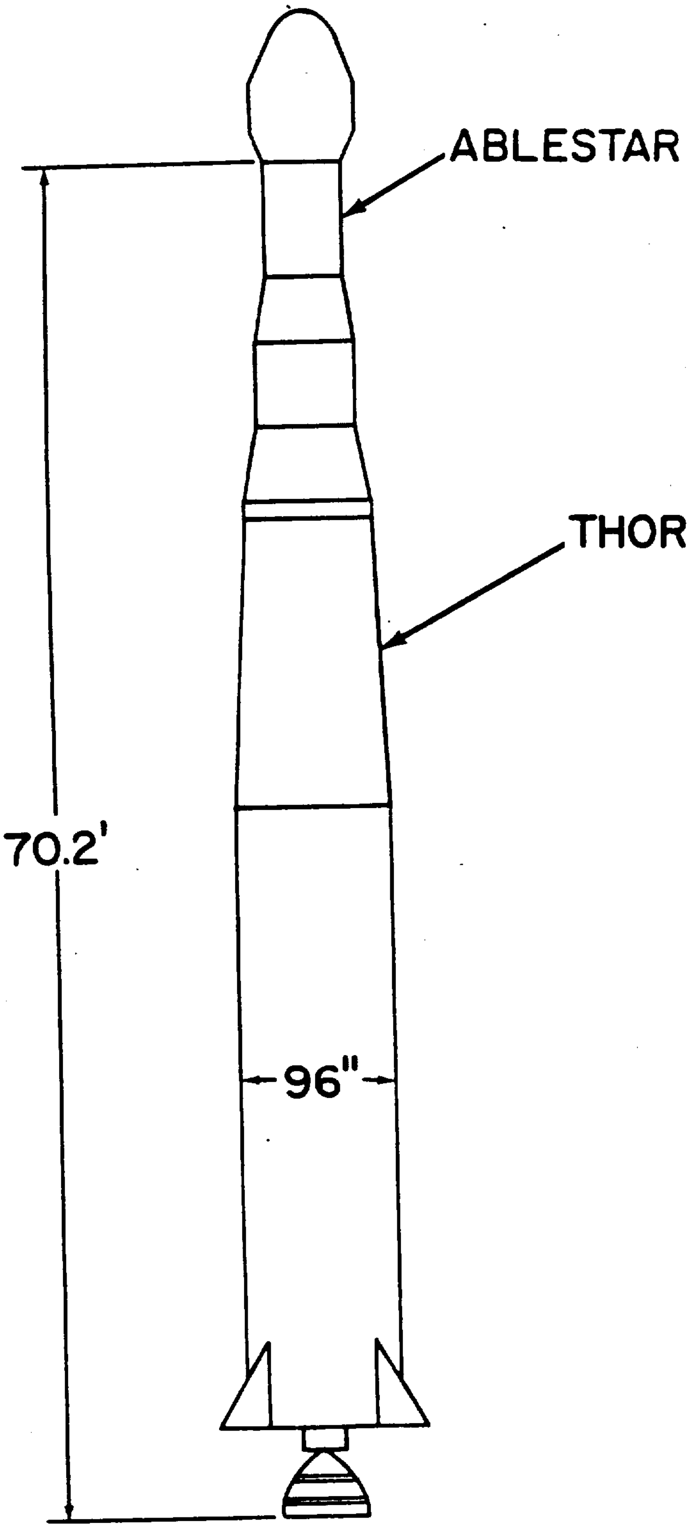
PROPULSION

Stage Number

	1	2	3	4
1. Engine Designation	ALGOL	CASTOR XM-33-E5	ANTARES X254	ALTAIR X248
2. Contractor	Aerojet	Thiokol	ABL	ABL
3. Propellants	Polyurethane Solid	PBAA Solid	Double Base Solid	Double Base Solid
4. No. of chambers	1	1	1	1
5. Thrust (SLS) per chamber	98,760	---	---	---
6. Thrust (vac) per chamber	---	48,430	13,450	2,820
7. Total Impulse (SLS)	4,077,800	---	---	---
8. Total Impulse (vac)	---	1,932,400	534,080	116,840
9. Nozzle expansion ratio	4.64	15.61	25.15	25.8
10. Chamber pressure	294.4	398.7	294.4	216.8
11. Nozzle exit area	813.6	1164	668.5	195.3
12. Relite capabilities	N/A	N/A	N/A	N/A
13. Propulsion Feed system	N/A	N/A	N/A	N/A


SCOUT-B ORBIT PERFORMANCE
DUE EAST LAUNCH
DUE SOUTH LAUNCH
PERIGEE 100 NM
PERIGEE 100 NM
CIRCULAR ORBIT
PAYLOAD - LB.
APOGEE - NM
100
1000
10000
200
100


SCOUT-B
PERIGEE 300 NM
DUE EAST LAUNCH
DUE SOUTH LAUNCH
PAYLOAD - LB.
200
100
0
APOGEE - N.M.
100
1000
10000



THOR ABLESTAR

DM-21A + Able Star

The Thor Able Star is not currently being used in the NASA program. It is currently in use in the Transit Project. The nominal weights and performance are presented in the following tables and figures.

Axial Loading and Wind Shear Limitations (See similar sections of Thor Delta.)

TABLE I

THOR (DM219)/ABLE STAR

STAGE CHARACTERISTICS

	First Stage	Second Stage
1. Stage Designation	DM 21-A	Able-Star
2. Lift-off Weight (less P.L.)	119,026 lbs	10,116 lbs.
3. Loaded Stage Weight	108,747 lbs	9,846 lbs
4. Propellant Weight (usable)		
a. Fuel Weight (99.5% PU)	31,856 lbs	8,331 lbs*
b. Oxidizer Weight (99.5% PU)	68,494 lbs	
c. Other Expendables	232 lbs	
5. Propellant Weight (residual)		
a. Usable contingency	505 lbs	127 lbs
b. Unusable contingency	416 lbs	20 lbs
c. Other Fluids	456 lbs	28 lbs
6. Stage Burnout Weight	8,331 lbs (1)	1,514 lbs
7. Jettisonable Weight		270 lbs
8. Stage Propellant Fraction	0.921	0.846

* Total propellants based on 98.5% P.U.

(1) Includes 163 lbs. adapter for 2nd stage

TABLE II THOR-ABLE STAR

PROPULSION CHARACTERISTICS	FIRST STAGE Main Propulsion	FIRST STAGE Vernier	SECOND STAGE
1. Engine Designation	XLR-79-NA-9	XLR-101-NA-9	AJ10-104
2. Contractor	Rocketdyne	Rocketdyne	Aerojet General Corp.
3. Propellants	RP-1 & LOX	RP-1 & LOX	UDMH-IRFNA
4. No. of Chambers	One	Two	One
5. Thrust (SLS) Per Chamber	150,000 lbs.	1,000 lbs.	-----
6. Thrust (vacuum) per chamber	176,000 lbs.	1150 (pump fed) 1000 (tank fed)	7890 lbs.
7. I_{sp} (SLS)	245 sec	207.7 sec	
8. I_{sp} (vacuum)(1)	284 sec	240 (pump fed) 235 (tank fed)	278 sec
9. Nozzle Expansion Ratio	8	5	40:1
10. Chamber Pressure	530 psia	360 psia	206±6 psia
11. Nozzle Exit Area, A_c	1640 in^2	10.5 in^2	865.60 sq in
12. Relite Capabilities	None	None	Yes
13. Propulsion Feed System	Turbo-pump	Turbo-pump & tank fed (Post MECO)	Pressure Feed

Note: (1) I_{sp_v} At MECO (main engine cut-off)

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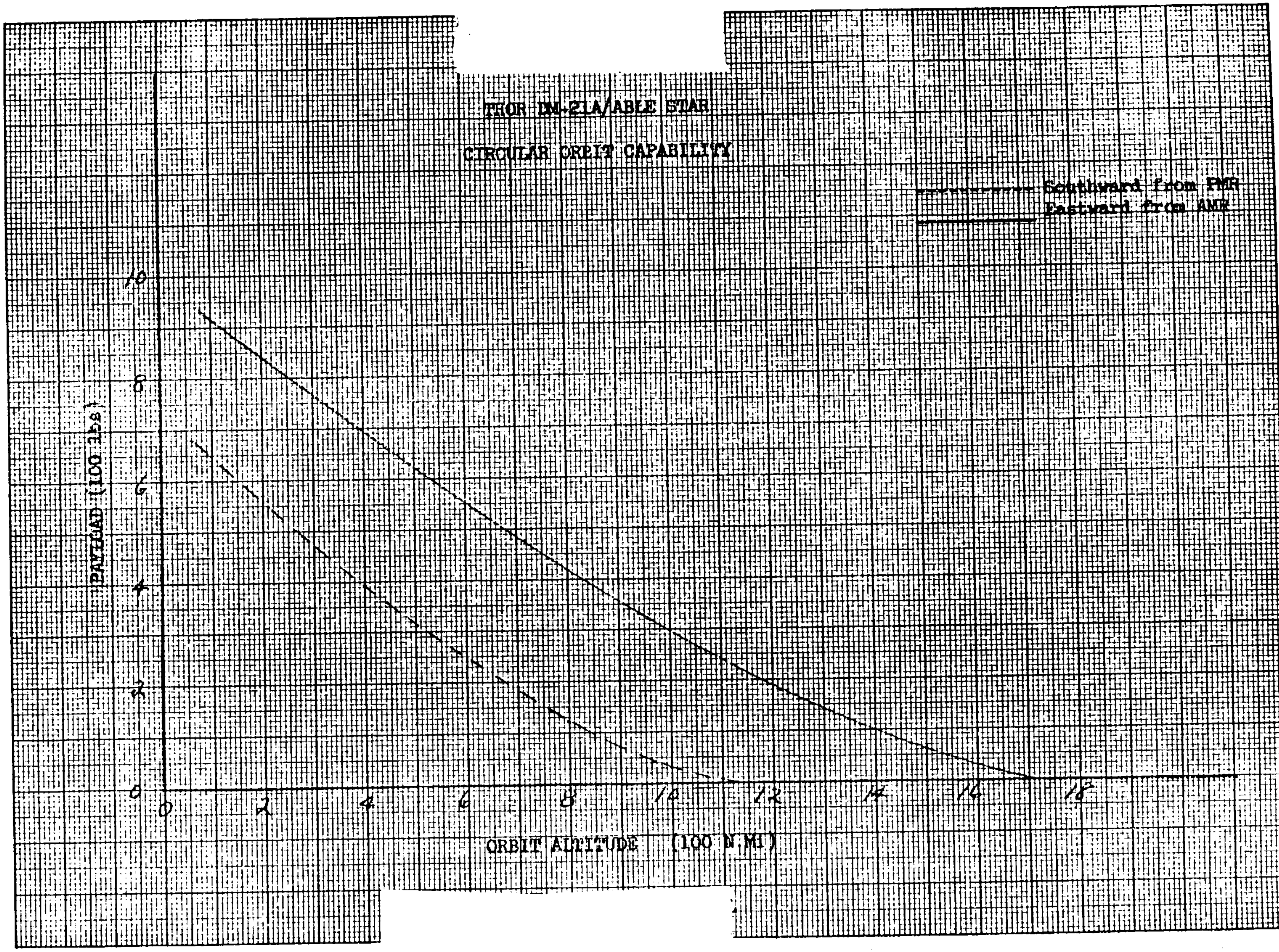
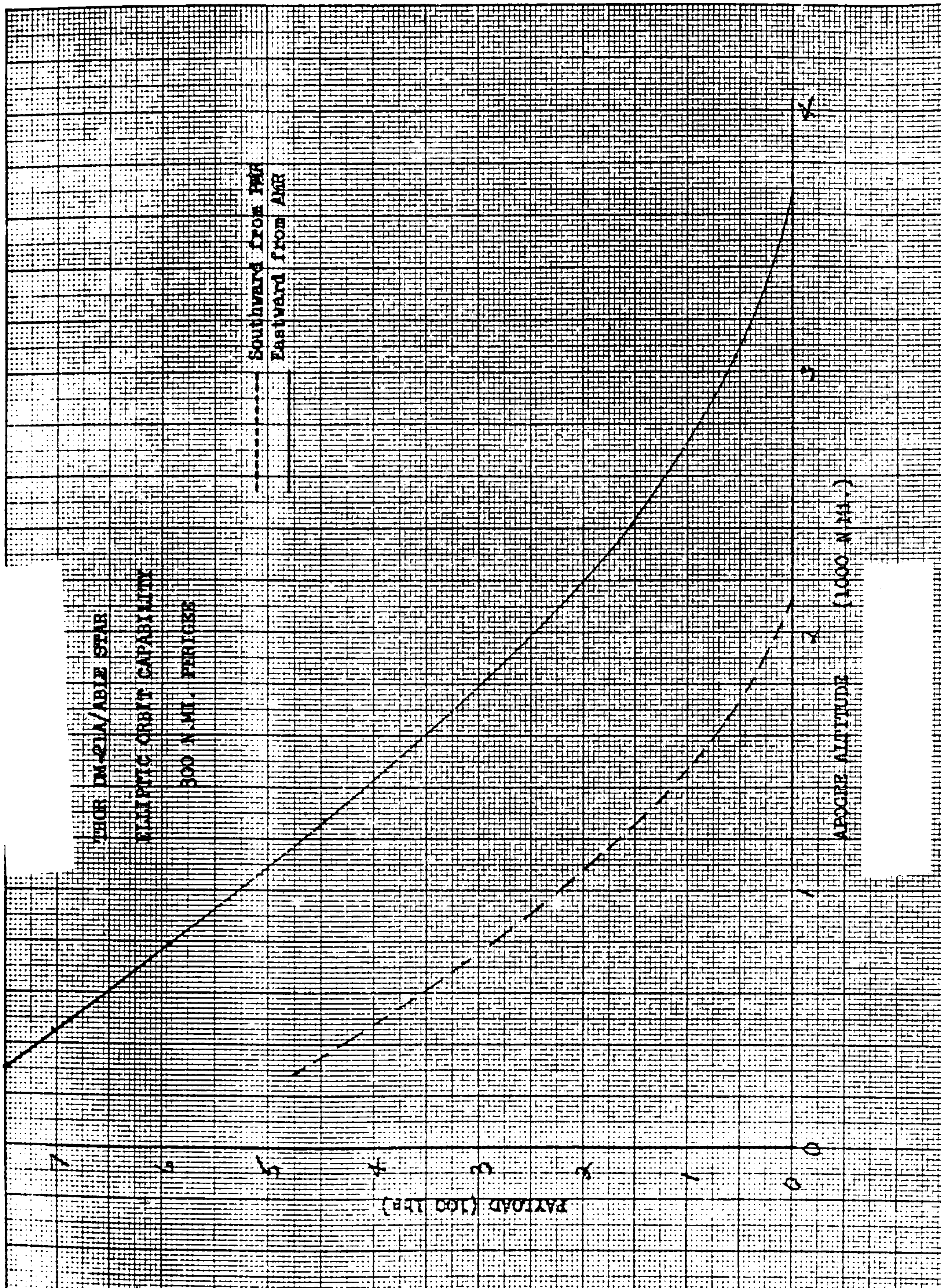
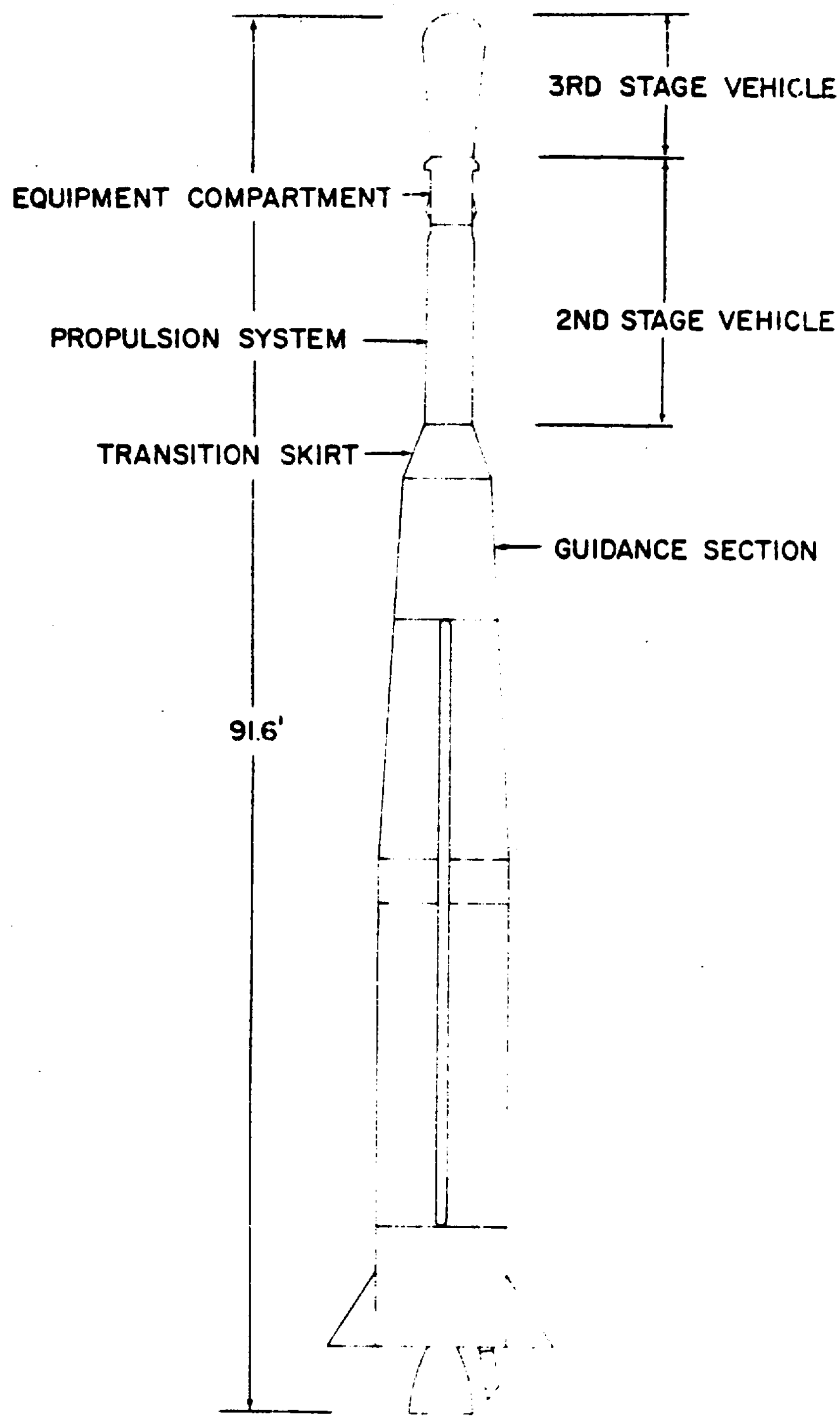

THOR DM-21A/ABLE STAR
CIRCULAR ORBIT CAPABILITY
Southward from PMR
Eastward from AMR
PAYLOAD (100 lbs)
0
2
4
6
8
10
ORBIT ALTITUDE (100 N.MI)
0
2
4
6
8
10
12
14
16
18


THOR DM-21A/ABLE STAR
ELLIPTIC ORBIT CAPABILITY
300 N.MI. PERIGEE
Southward from PMR
Eastward from AMR
APOGEE ALTITUDE (1000 N.MI.)
PAYLOAD (100 LBS)
0
1
2
3
4
5
6
7



DELTA MODEL DM-19

15

DELTA, DM 19

The trajectory and payload data presented are based on nominal inputs. The payload is defined as the weight available for the scientific payload and its integration structure as well as any guidance required beyond that of the conventional vehicle.

Third Stage/Payload Fairing Configurations

A thermal and aerodynamic shield is provided to decrease aerodynamic drag and adequately protect the payload and the third stage rocket during flight through the sensible atmosphere. This fairing is in two halves which is affixed to and separated from, the second stage vehicle by means of explosive thrusters. The fairing is ejected at an altitude where protection from aerodynamic heating is no longer required.

Figure 1 presents the so-called "high drag" and "low drag" fairings. The high drag fairing is used for payloads of larger volume. Circular and elliptical orbit payload data is presented for both fairings. The detailed trajectory data is presented only for the low drag fairing.

Characteristics of Delta Guidance System

The Delta utilizes a flight controller mounted in the first stage guidance section for control of the vehicle during first stage powered flight. Three rate gyros mounted on the first stage are used in conjunction with the flight controller. At the end of the first 90 seconds of powered flight, the modified 300 BTL radio guidance system, located in the second stage vehicle modifies the first stage flight control programmer signals.

The BTL system specifies that the desired minimum elevation angle prior to guidance termination be 13°. An elevation angle of 5° is the absolute minimum that can be tolerated, and only under extreme circumstances should this limit be approached. A satisfactory trajectory can be obtained if the BTL radar line-of-sight limit angle is held to 10°.

After second stage burnout, the remaining second/third stage vehicle attitude shall be controlled by the coast control system consisting of the second stage flight controller and four helium gas jet nozzles working on an on-off type of operation.

The third stage is spin stabilized at 150 ± 30 rpm. Spin-up is accomplished just prior to third stage ignition by rockets (the number varying dependent on the moment of inertia of the payload) tangentially mounted on the spin table.

The guidance and control accuracies are as follows:

Accuracy at Guidance Termination

The systems accuracy is such that the three standard deviation values of vehicle dispersion due to guidance at termination of guidance at second stage shutdown are:

Body Attitude	5.4 mils
Velocity Error in Range	9.0 ft/sec
Crossing Velocity Error	12.0 ft/sec
Altitude	0.24 n. mi.

Terminal Accuracy (Probe)

The dispersion errors (3σ) of the final velocity vector at third stage burnout when the vehicle is used as a probe with no coast control are:

Velocity	125.0 ft/sec
Flight Path	6.0 mils

Terminal Accuracy (Satellite)

The dispersion error (3σ) of the final velocity shall be as specified in Terminal Accuracy (Probe) with degradation of angular accuracy due to the coast control system. The coast control system shall be capable of being programmed with an accuracy of 0.1% in both time and angle. The random drift of the MIG reference shall be no greater than 1.0°/hr. during the coast phase.

Axial Loading Limitations

The critical structural criteria for the Thor booster is the maximum axial load at main engine cut-off. This limitation places an upper limit on booster payload. This limit is 16,000 pounds for unmodified vehicles and 17,000 pounds for a maximum allowable increase in fuel tank pressure.

Wind Shear

The vehicle is capable of withstanding a 95% probable wind shear. Structurally the critical wind shear shall be considerably greater than this value, however, the control system is designed for the 95% probable value of wind shear.

TABLE I

DELTA VEHICLE

STAGE CHARACTERISTICS

	STAGE NUMBER 1	2	3	
1. Stage Designation	XLR-79-NA-9	AJ10-118	ABL X-248-A5	
2. Light-off Weight (Less payload) (Lbs)	111,125	5197.0	514.8	
3. Loaded Stage Weight (Lbs)	105,928	4682.2	514.8	
4. Propellant Weight (Useable) (Lbs)			455.3	
(a) Fuel	29,400	837.0		
(b) Oxidizer	67,914	2341.0		
(c) Other fluids overboard				
During burning	220	16.0	7.6	
Start & stop losses		20.0	0.6	
5. Propellant weight (residual) (Lbs)				
(a) Useable contingency	491	132.0	0	
(b) Unusable contingency	444	52.0		
(c) Other fluids, trapped & unused	450	14.0		
6. Stage burnout weight (SBW) (Lbs)	8,394	1334.2	51.3	
7. SBW + Jettisonable weight (Lbs)	8,394	1469.0	51.3	
8. Stage Propellant Fraction				
$\frac{\text{Useable propellant weight}}{\text{Stage loaded weight}}$	.919	.678	.885	

TABLE II

DELTA VEHICLE

STAGE CHARACTERISTICS

PROPULSION (SEPARATE LISTINGS FOR MAIN ENGINE OR ENGINES AND VERNIER ENGINES)	STAGE NUMBER			
	1	Vernier	2	3
1. Engine Designation	XLR-79-NA-9	XLR-101-NA-9	AJ10-118	ABL X-248-A5
2. Contractor	Rocketdyne	Rocketdyne	Aerojet Gen.	Allegany
3. Propellants	RP-1 & LOX	RP-1 & LOX	UDMH & WIFNA	SOLID
4. No. of Chambers	1	2	1	1
5. Thrust (SLS) Per Chamber (Lbs)	150,000	1,000	---	---
6. Thrust (Vacuum) Per Chamber (Lbs)	~174,000	1150(Pump Fed) 1000(Tank Fed)	7443	2751.3
7. I_{sp} (SLS) (Sec)	245	207.7	---	---
8. I_{sp} (Vacuum)$^{(1)}$ (Sec)	284	240(Pump Fed) 235(Tank Fed)	263	254
9. Nozzle Expansion Ratio	8:1	5	20:1	20:1
10. Chamber Pressure (Psi)	530	360	206	225
11. Nozzle Exit Area (In^2)	1640	10.5	432.8	249.6
12. Relite Capabilities	None	None	None	None
13. Propulsion Feed System	Turbo-Pump	Turbo-Pump and Tank Fed (Post Meco)	Press.	Temp.

Note: (1) I_{sp} at MECO (Main Engine cut-off)

TABLE III

TRAJECTORY AND PAYLOAD DATA

Vehicle: Delta DM-19
(low drag fairing)

Mission: 300 N. Miles Circular

Launch: AMR East Launch

Payload Weight in Orbit = 640 lbs.

POSITION CONDITIONS	TIME (sec)	ALTITUDE (NM)	DISTANCE (NM)	DYNAMIC PRESSURE (lb/ft^2)	INERTIAL VELOCITY (ft/sec)	ACC. (g's)	TILT ANGLE(1) (deg)	IMPACT POINTS (NM)	
1. Launch	0	0	0	0	1340	1.36	90.0	0	
2. Stage I Burnout	158.7	44.6	82.0	2.3	14,376	12.20	72.1	1189.6	
3. Stage II Ignition	162.7	47.7	90.8	1.1	14,352	1.34	72.4	1193.0	
4. Stage II Burnout	260.6	122.6	340.6	0	19,555	2.82	76.4	3261.6	
5. Stage III Ignition	720.3	301.1	1629.0	0	17,906	3.01	90.4	3260.2	
6. Stage III Burnout	762.3	300.0	1758.6	0	23,499.	0	90.0	0	

(1) Referred to local horizontal

TRAJECTORY AND PAYLOAD DATA

TABLE IV

Vehicle: Delta DM-19 (low drag fairing)

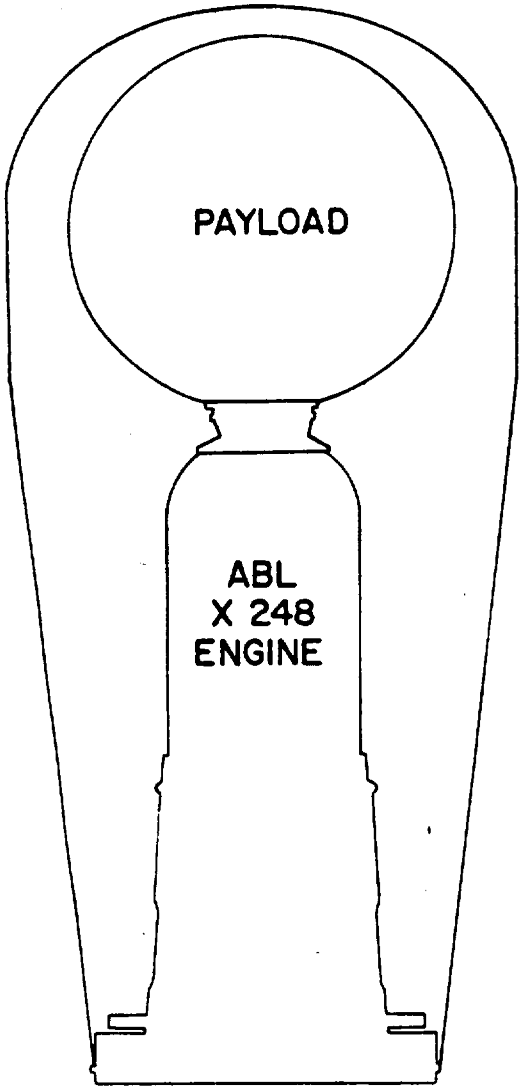
Mission: 900 N Miles, Circular

Launch: AMR East Launch

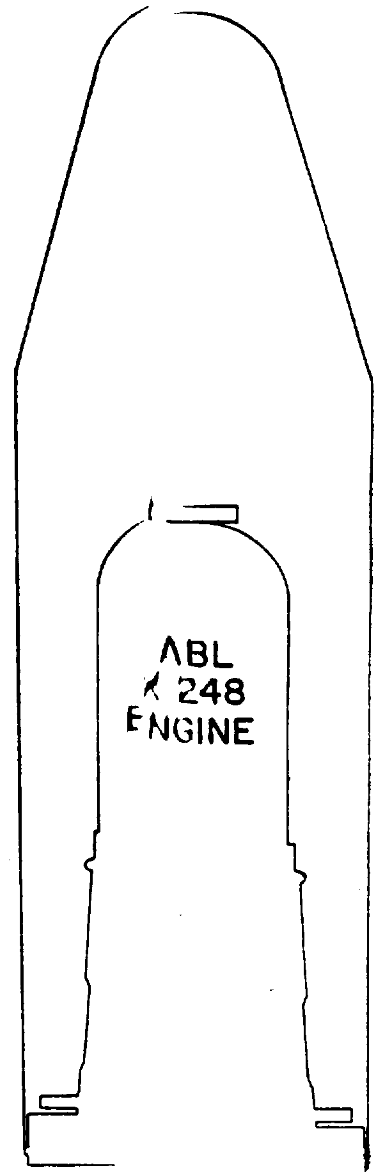
Payload Weight in Orbit = 305 lbs.

POSITION CONDITIONS	TIME (sec)	ALTITUDE (NM)	DISTANCE (NM)	DYNAMIC PRESSURE (lb/ft^2)	INERTIAL VELOCITY (ft/sec)	ACC. (g's)	TILT ANGLE[(1)] (deg)	IMPACT POINTS (NM)
1. Launch	0	0	0	0	1340	1.36	90	0
2. Stage I Burnout	159.9	62.0	74.9	0.02	14,532	12.45	60.7	1504.6
3. Stage II Ignition	163.9	67.0	82.8	1.44	14,486	1.26	61.0	1509.0
4. Stage II Burnout	275.7	216.2	344.5	0	20,580	3.43	64.6	4784
5. Stage III Ignition	1185.	900.4	2362.3	0	15,002	4.11	90.0	4784
6. Stage III Burnout	1225.	900.0	2452.0	0	21,830	0	90.0	0

(1) Referred to local horizontal

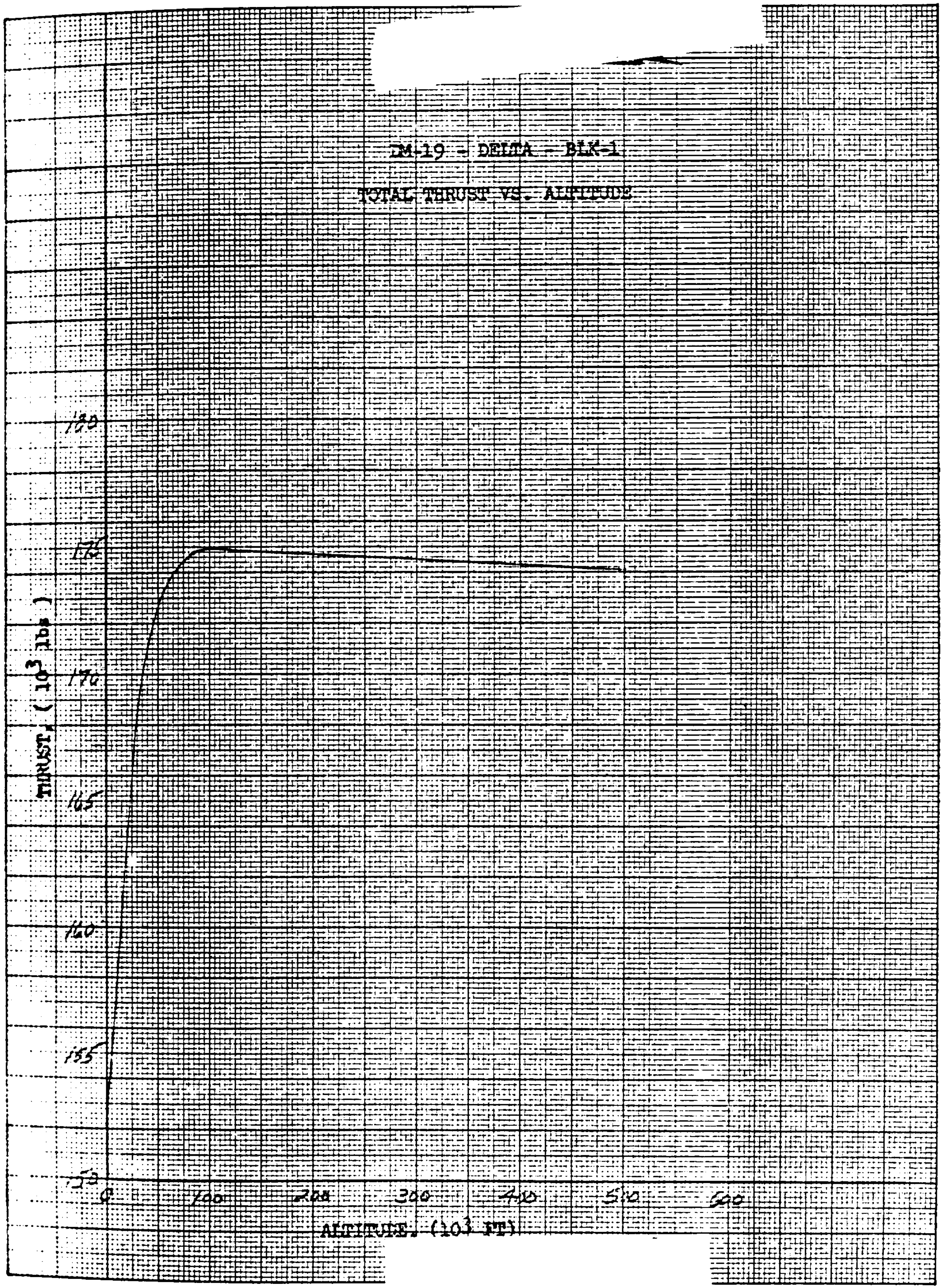


STANDARD FAIRING
(HIGHER DRAG)

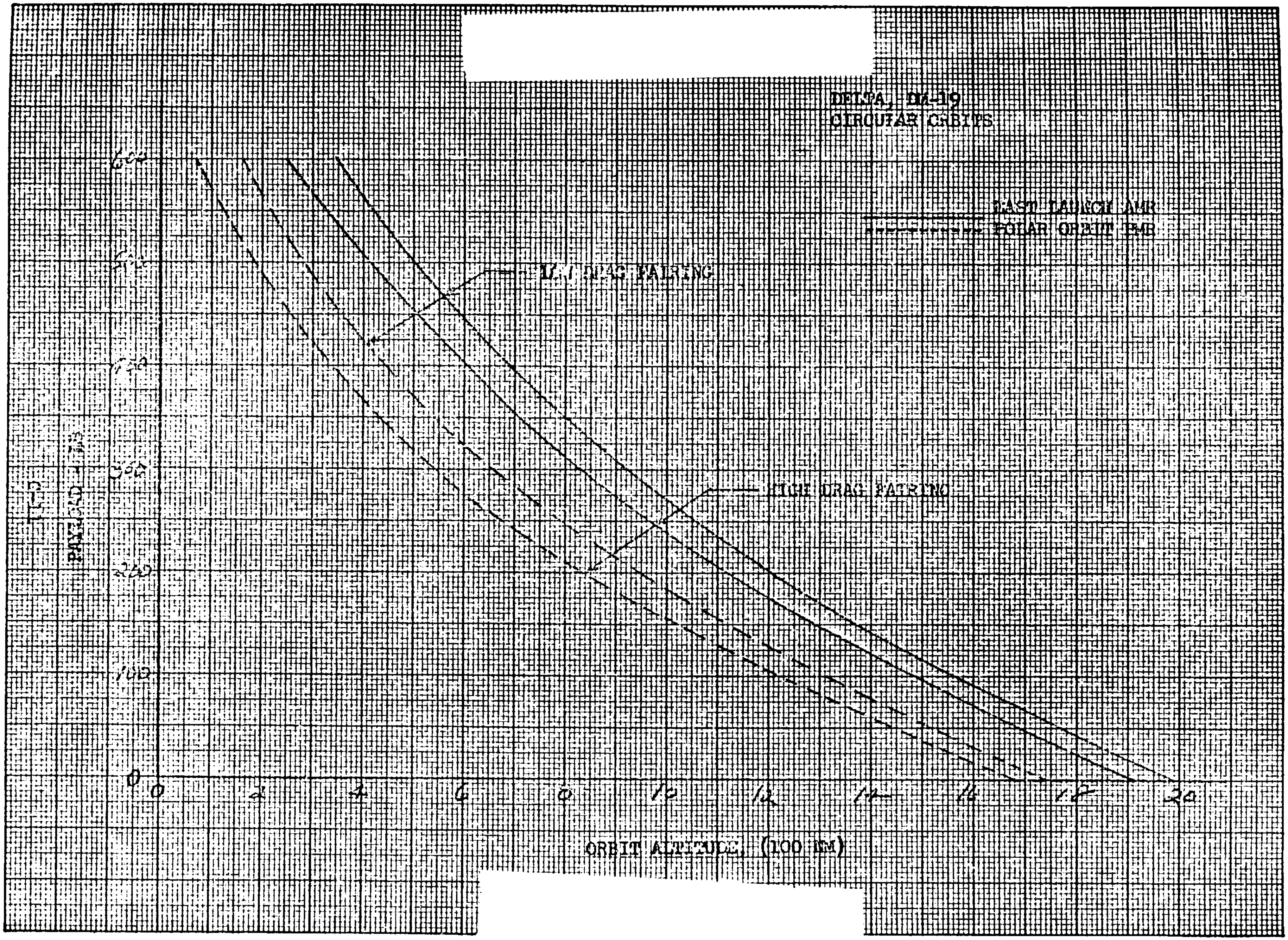


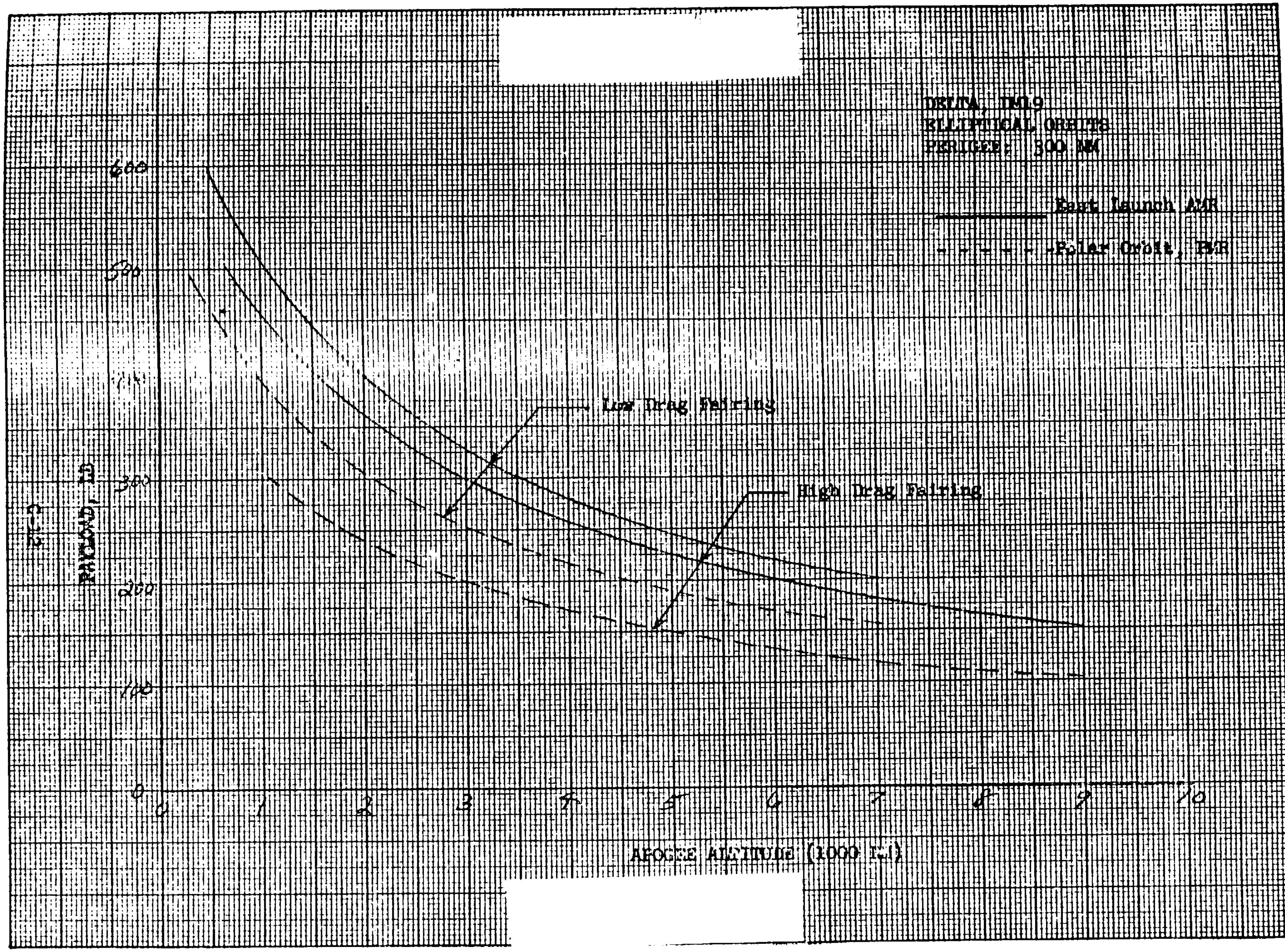
STREAMLINED FAIRING
(LOWER DRAG)

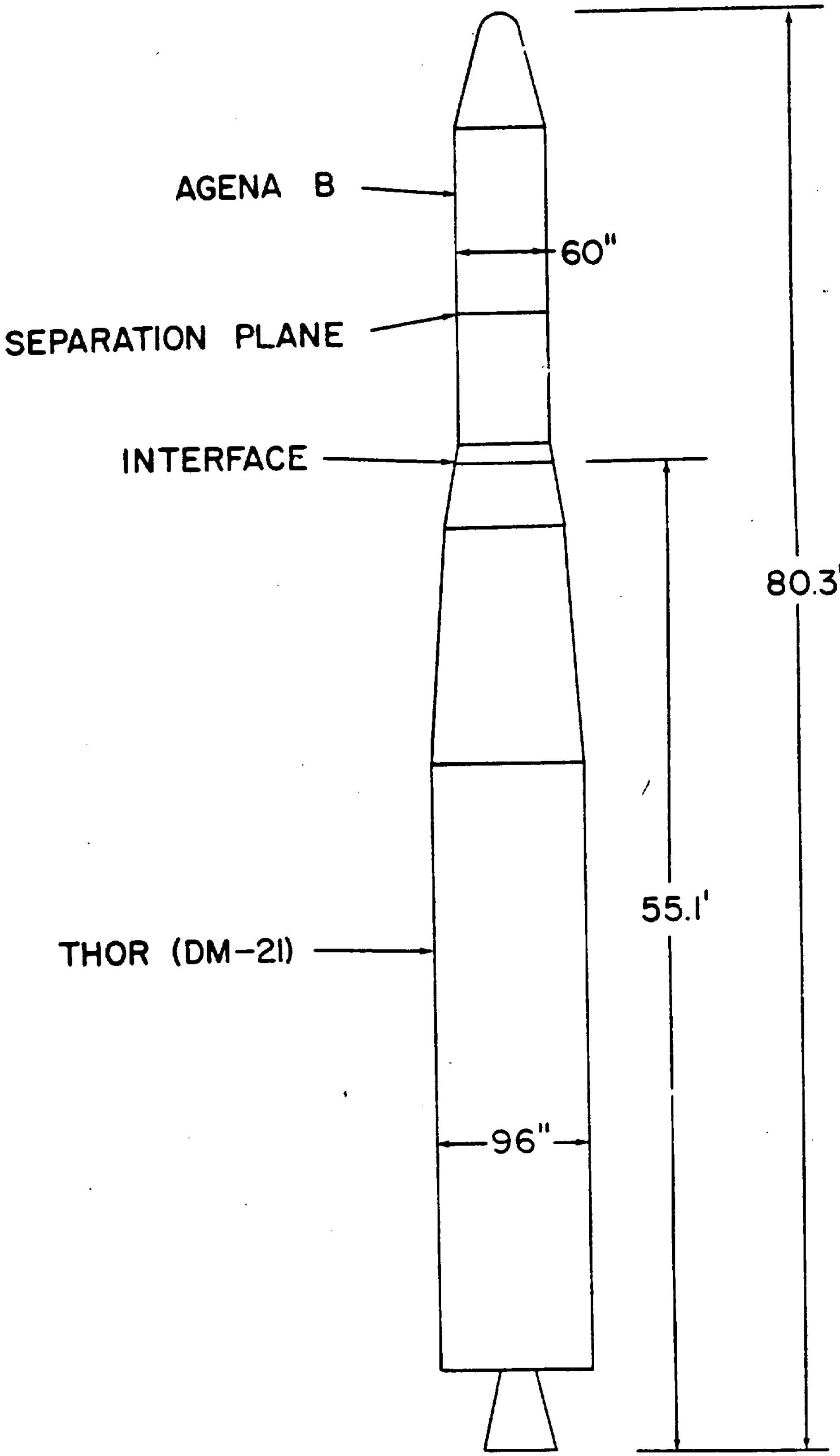
DELTA PAYLOAD FAIRINGS



DM-19 - DELTA - BLK-1
TOTAL THRUST VS. ALTITUDE
180
175
170
165
160
155
150
THRUST, (10^3 lbs)
0
100
200
300
400
500
600
ALTITUDE, (10^3 FT)


DELTA, DM-19
CIRCULAR ORBITS
EAST LAUNCH AMR
POLAR ORBIT PMR
LOW DRAG FAIRING
HIGH DRAG FAIRING
PAYLOAD - LBS
600
100
0
0
2
4
6
8
10
12
14
16
18
20
ORBIT ALTITUDE, (100 NM)


DELTA, DM19
ELLIPTICAL ORBITS
PERIGEE: 300 NM
East Launch AMR
Polar Orbit, PMR
Low Drag Fairing
High Drag Fairing
PAYLOAD, LB
600
500
300
200
100
0
0
1
2
3
4
5
6
7
8
9
10
APOGEE ALTITUDE (1000 NM)



THOR—AGENA B

THOR DM 21 - AGENA B

TRAJECTORY AND PAYLOAD DATA

The trajectory data presented is based on nominal inputs. The payload data, however, is minimum guaranteed (-3σ) obtained by taking the root-sum-square of the degradations in the booster and algebraically adding similar degradations in the Agena to obtain the -3σ total degradation. In practice this means that there is a quantity of nominally unused propellant contingency in both the booster and the Agena to compensate for trajectory dispersions.

The Thor booster trajectory is similar to that of the Atlas boosted trajectory described in the Atlas-Agena B section except that the gravity turn trajectory segment is continued until vernier engine cutoff.

Characteristics of Thor/Agena B Guidance System

The Thor/Agena B utilizes a radio command guidance system for the first stage. This first stage guidance function is accomplished by the Bell Telephone Laboratories (BTL) system which tracks the launch vehicle to obtain positional information at the ground station. Positional rate-of-change information is used to obtain vehicle velocity in a ground-based computer. Comparison of the measured position and computed velocity with desired parameters permits correction command to be radioed to the vehicle. A radio command receiver and Autopilot (employing HIG and rate gyro loops) enables the Thor vehicle to make corrections based upon these commanded inputs. The minimum flight path angle is limited to approximately ten degrees to maintain radio line-of-sight. The vehicle-borne Thor guidance equipment weighs approximately 150 lbs.

The second stage (Agena B) airborne guidance and control system constrains the trajectory and vehicle attitude as required during the ascent interval between separation from the first stage booster and injection into orbit. Initial attitude references are established by orienting the first stage booster during vernier to the desired attitude via radio steering commands and uncaging the Agena B gyros at vernier shut-off command. Attitude reference during second stage flight is provided by open-loop torquing of the body-mounted high-performance Agena gyros. Vehicle pitch and yaw attitude is controlled to the gyro reference by Agena engine swiveling during thrust intervals and by gas reaction jets during coast. Roll is controlled by gas reaction jets during both powered and unpowered flight. The Agena engine burns twice, separated by a coast period to produce the velocity and altitude gains required. Thrust duration for both burns, and hence velocity gained, is controlled through acceleration integration. Initiation of second burn is determined by grou nd command.

TABLE I

THOR DM21/AGENA B

STAGE CHARACTERISTICS	1	2
1. Stage Designation	Thor Booster and Vernier	Agena B
2. Light-off Weight (less payload)[a], lb	123,349	N/A
2'. Separation Weight (less payload), lb	N/A	14,697(4)
Jettisoned Weight	N/A	56
3. Loaded Stage Weight[b] (= ignition weight for Agena)	108,395	14,641
4. Propellant Weight Usable, lb	99,380	13,077(5)
(a) Fuel		3,668
(b) Oxidizer		9,409
5. Propellant Weight Residual, lb	1,859	203
(a) Usable Contingency	630	60(6)
(b) Unusable Contingency	1,229	143
5'. Non-Impulse Expendables, lb	646	79
(a) Propellants	---(1)	25(7)
(b) Others	646	54(8)
6. Stage Burnout Weight (SBW)[c], lb	8,160(2)	1,425(9)
7. SBW + Jettisoned Weight[d], lb	8,417(3)	N/A
8. Stage Propellant Fraction, $\frac{\text{Usable propellant wt}}{\text{Stage loaded wt}}$	99,380/108,395 = .917	13,077/14,641 = .893

Footnotes:

(a) "Light-off Weight (less Payload)" is the sum of all remaining stage weights including the stage proper, interstage adapters but excluding payload

(b) "Loaded Stage Weight" consists of structure, equipment and propellant (usable and residual) weights of an individual stage excluding the next stage adapter

(c) "Stage Burnout Weight (SBW)" equals the "Loaded Stage Weight" less both the usable propellant and the non-impulse expendables

(d) "SBW plus Jettisonable Weight" equals "SBW" plus jettisonable weight of the next stage adapter, miscellaneous jettisonable weight and, if applicable, nose cone weight.

(1) Includes the following items:

101 lb	lube oil in boost phase
124 lb	pressure gas " "
26 lb	lube oil in vernier phase
395 lb	pressure gas in vernier phase
646 lb	

(2) Includes the following items:

1217 lb	unusable contingency boost phase
12 lb	unusable contingency vernier phase
26 lb	lube oil in vernier phase
395 lb	pressure gas in vernier phase
6510 lb	booster inert weight
8160 lb	

(3) Includes the following items:

8160 lb	as above
232 lb	adapter
18 lb	retro-rockets
7 lb	destruct system
8417 lb	

(4) Includes the following items:

6 lb	horizon sensor fairings
6 lb	control gas
38 lb	ullage rockets 1st burn
5 lb	propellant pre-flow 1st burn
1 lb	engine start charge 1st burn
56 lb	

(5) Does not include item 5a

(6) Not included in Stage Burnout Weight.
It is assumed used in -3σ case.

(7) Consists of 1st burn postflow + 2nd burn preflow.

(8) Includes the following items:

38 lb	2nd burn ullage rockets
15 lb	control gas
1 lb	2nd engine start charge
54 lb	

(9) Includes the following items:

515 lb	structure (including tanks)
316 lb	propulsion
86 lb	controls (includes 3 lb of residual control gas)
114 lb	guidance
162 lb	APU
28 lb	communications
16 lb	miscellaneous
143 lb	unusable contingency (item 5b)
1425 lb	

TABLE II

THOR DM21/AGENA B

STAGE NO.

STAGE CHARACTERISTICS - PROPULSION		1 Booster	2 Vernier	3 Agena B
1. Engine designation		MB-3 Blk II XLR-79-NA-11		Bell 8096
2. Contractor		Rocketdyne	Rocketdyne	Bell Aerosystems Co.
3. Propellants		O_2, RP-1	O_2, RP-1	IRFNA, UDMH
4. No. of chambers		1	2	1
5. Thrust (S.L.S.) per chamber, lb		165,453	N/A	N/A
6. Thrust (vacuum) per chamber, lb		190,998	1000	16,027
7. I_{sp} (S.L.S.), sec		248.3	N/A	N/A
8. I_{sp} (vacuum), sec		286.2	250	289.5 (-3σ)
9. Nozzle expansion ratio		8:1		45:1
10. Chamber pressure, psia		536		508
11. Nozzle exit area, A_c, in^2		1644		770
12. Relite capabilities		none	none	one
13. Propellant feed system		turbopump	turbopump	turbopump

TABLE III

TRAJECTORY AND PAYLOAD DATA

Vehicle: Thor DM21/Agena B

Mission: 300 N. Miles, Circular

Launch: AMR at 109° East of North

Payload Weight in Orbit = 1,640 lbs.

POSITION CONDITIONS	TIME (sec)	ALTITUDE (NM)	DISTANCE NM	DYNAMIC PRESSURE (lb/ft^2)	INERTIAL VELOCITY (ft/sec)	ACC. (g's)	TILT ANGLE (deg)	IMPACT POINTS (NM)
1. Launch	0	0	0	0	1,340	1.3	0	0
2. Main engine cutoff	147	37	60	11	10,590	7.5	70	600
3. Vernier engine cutoff	156	43	75	---	10,520	0.08	71	600
4. Agena first Ignition	187	58	125	---	10,250	1.02	75	600
5. Agena first cutoff	420	80	710	---	26,022	5.8	90	---
6. Agena second Ignition	3,157	300	---	---	24,480	5.9	90	---
7. Agena final cutoff	3,159	300	---	---	24,840	6.5	90	---

1.) Referred to local vertical

TABLE IV

TRAJECTORY AND PAYLOAD DATA

Vehicle: Thor DM21/Agena B

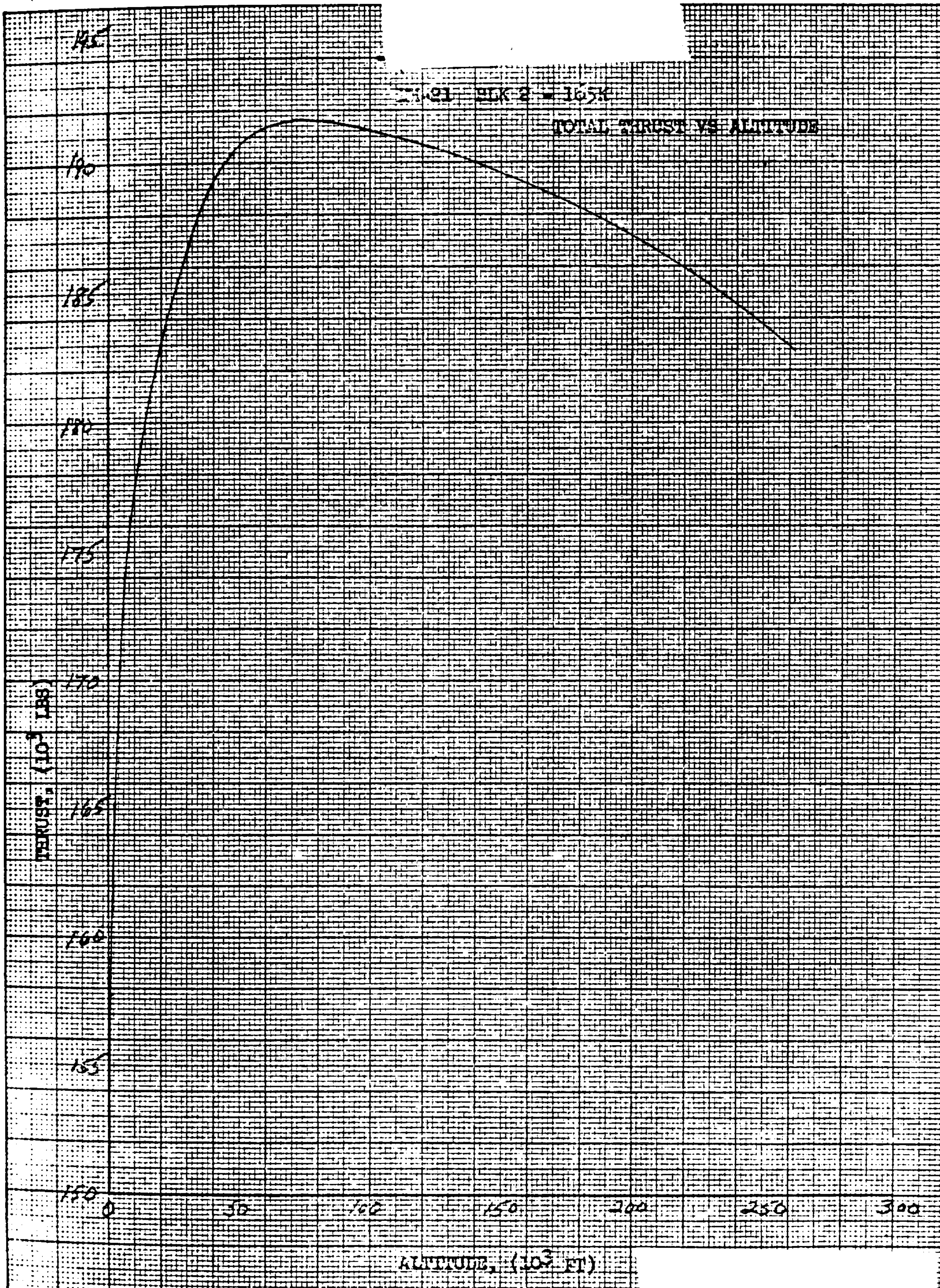
Mission: 900 N. Miles, Circular

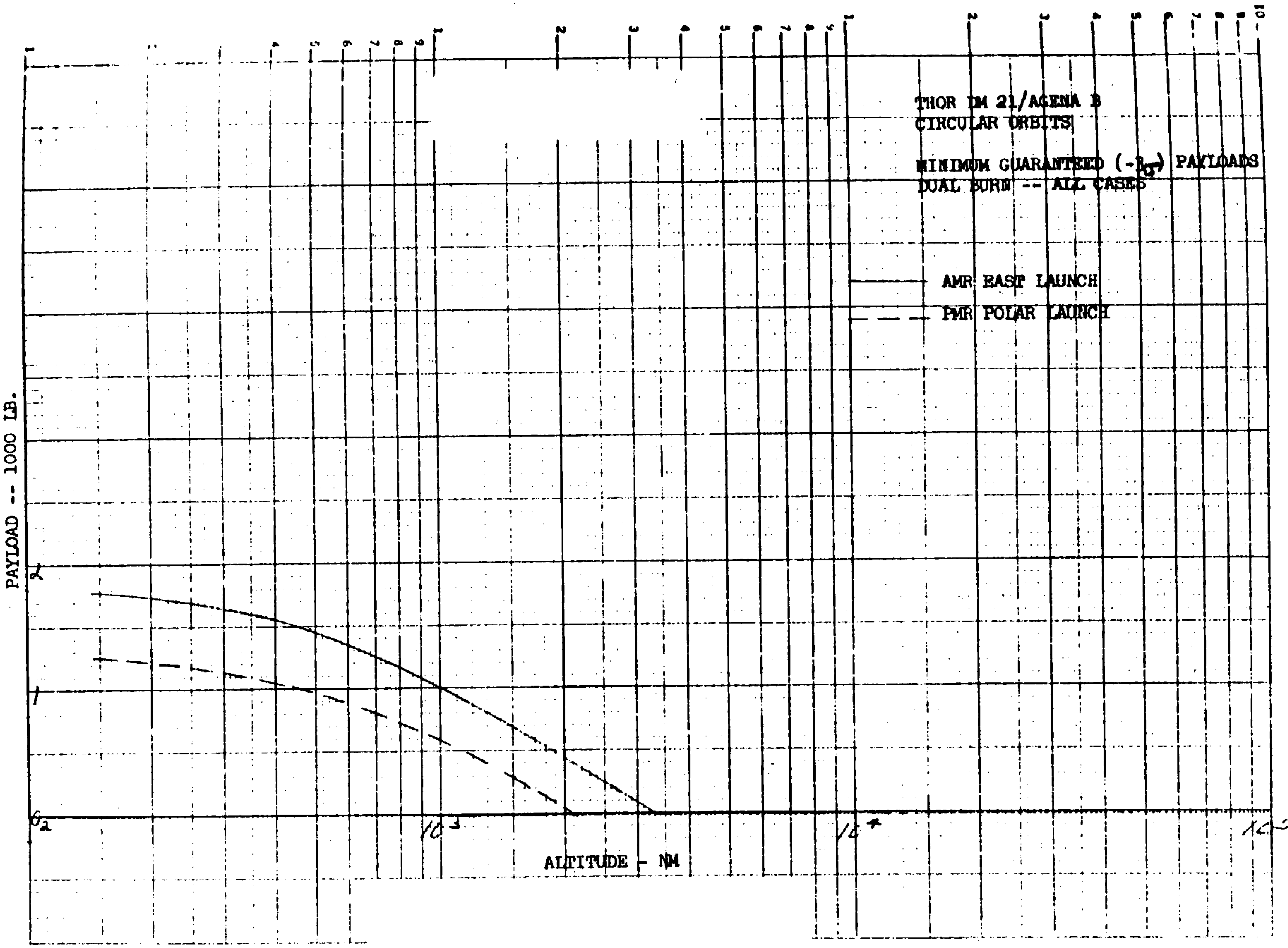
Launch: AMR at 109° East of North

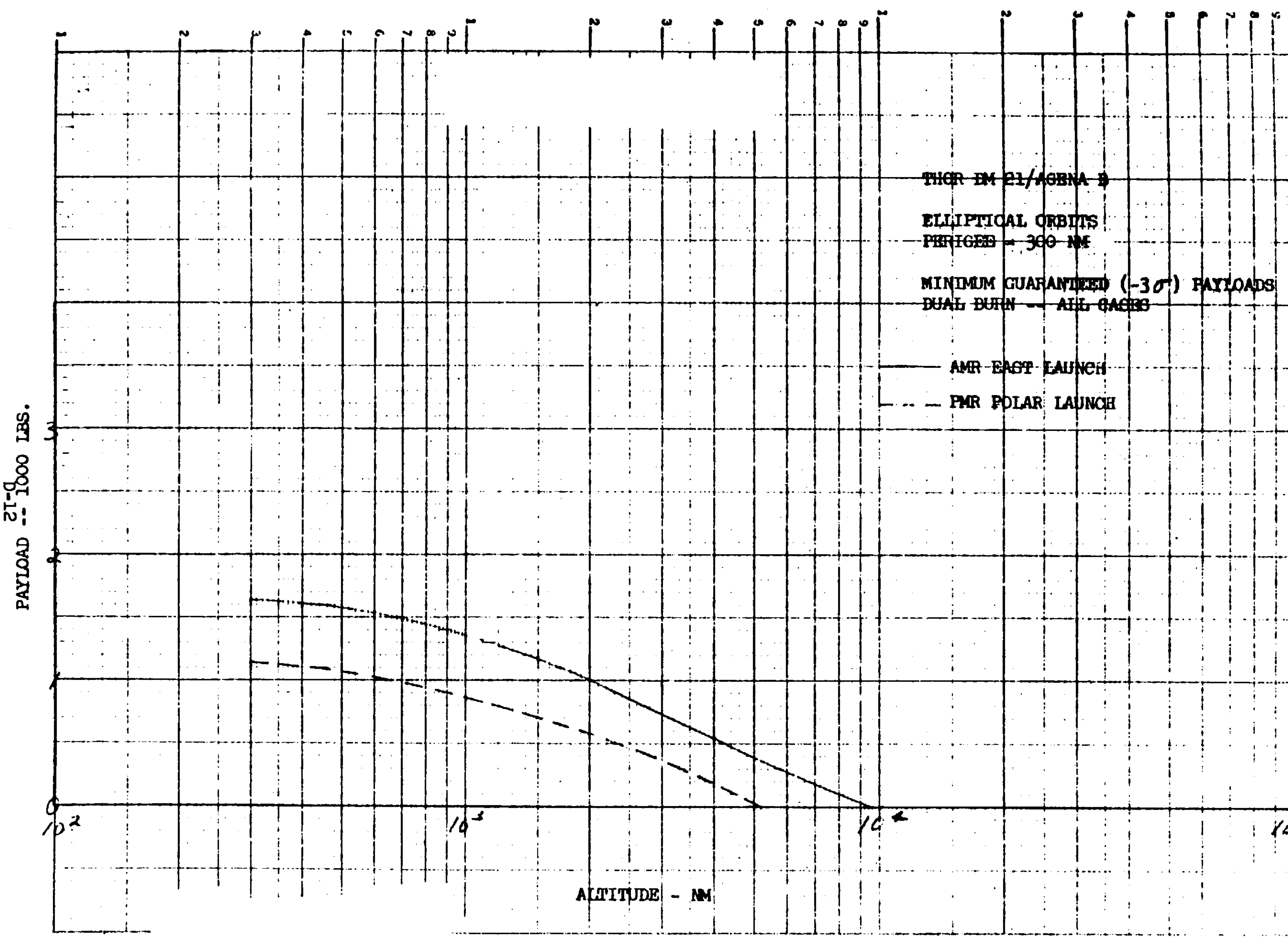
Payload Weight in Orbit: 1,195 lb

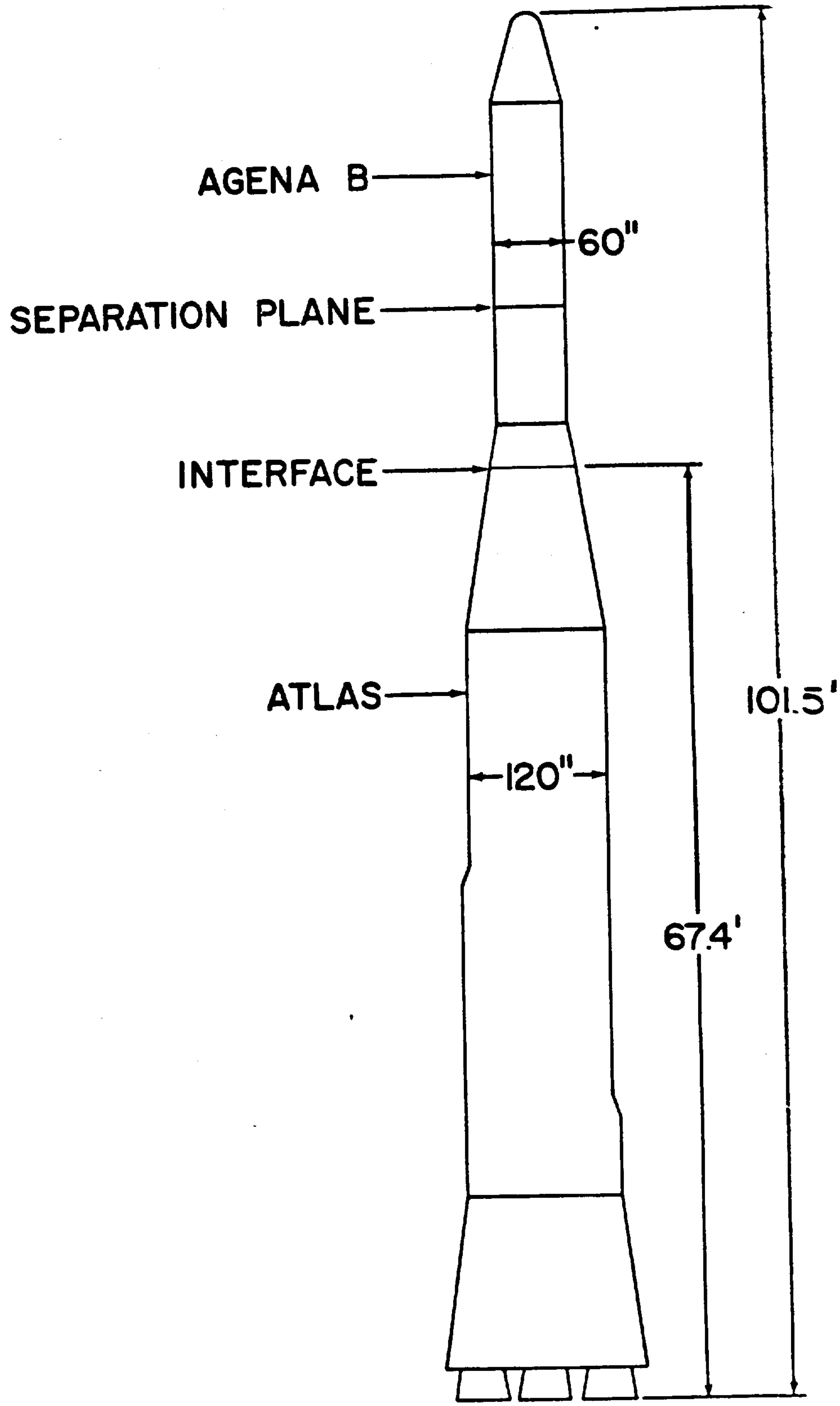
POSITION CONDITIONS	TIME (sec)	ALTITUDE (NM)	DISTANCE (NM)	DYNAMIC PRESSURE (lb/ft^2)	INERTIAL VELOCITY (ft/sec)	ACC. (g's)	TILT[1)] ANGLE (deg)	IMPACT POINTS (NM)
1. Launch	0	0	0	0	1,340	1.3	0	0
2. Main engine cutoff	147	38	60	11	10,410	7.5	70	600
3. Vernier engine cutoff	156	43	75	--	10,340	0.08	71	600
4. Agena first ignition	187	58	125	--	10,070	1.02	75	600
5. Agena first cutoff	425	80	730	--	26,941	5.3	90	---
6. Agena second Ignition	3,509	900	---	--	21,841	5.4	90	----
7. Agena final cutoff	3,515	900	---	--	23,056	5.6	90	---

1.) Referred to local vertical


21 BLK 2 - 165K
TOTAL THRUST VS ALTITUDE
195
190
185
180
175
170
165
160
155
150
0
50
100
150
200
250
300
THRUST, (10^3 LBS)
ALTITUDE, (10^3 FT)


THOR DM 21/AGENA B
CIRCULAR ORBITS
MINIMUM GUARANTEED (-3σ) PAYLOADS
DUAL BURN -- ALL CASES
AMR EAST LAUNCH
PMR POLAR LAUNCH
PAYLOAD -- 1000 LB.
ALTITUDE - NM
2
1
0
10^2
10^3
10^4
10^5


THOR DM 21/AGENA D
ELLIPTICAL ORBITS
PERIGEE - 300 NM
MINIMUM GUARANTEED (-3σ) PAYLOADS
DUAL BURN -- ALL CASES
AMR EAST LAUNCH
PMR POLAR LAUNCH
PAYLOAD -- 1000 LBS.
ALTITUDE - NM
0
1
2
3
10^2
10^3
10^4
10^5



ATLAS-AGENA B

E-1

ATLAS SM68D/AGENA B

TRAJECTORY AND PAYLOAD DATA

The trajectory data presented is based on nominal inputs. The payload data, however, is minimum guaranteed (-3σ) obtained by taking the root-sum-square of the degradations in the booster and algebraically adding the root-sum-square of the degradations in the Agena to obtain the -3σ total degradation. In practice this means that there is a quantity of nominally unused propellant contingency in both the booster and the Agena to compensate for trajectory dispersions.

The Atlas-boosted trajectory begins with a vertical take-off followed by a roll maneuver to the desired azimuth heading. At the proper heading, the vehicle is pitched downward and has a small angle of attack until approximately 60 seconds after the take off, when a gravity turn (zero lift) trajectory is initiated.

At booster first stage separation, the vehicle is again pitched downward and follows a constant altitude flight path until sustainer cutoff and vernier engine cutoff. At this time, the Agena separates and coasts for approximately 35 seconds. During the coast period, the Agena is pitched downward to a small positive angle above the local horizontal. Upon Agena ignition, powered flight at constant altitude with respect to the local horizontal proceeds until the velocity vector is aligned with the local horizontal for injection at transfer orbit perigee.

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Characteristics of Atlas-Agena B Guidance System

Atlas guidance is performed by a General Electric-Burroughs radio track and command system. The vehicle is tracked to provide positional information. Doppler data provides instantaneous velocity indication. Based on these measured parameters, commands from the ground computer are transmitted to a vehicle-borne receiver to permit the vehicle to perform desired maneuvers. This closed-loop operation constrains the Atlas-Agena B to the desired trajectory. The vehicle-borne G.E. guidance equipment weighs approximately 150 lbs. Minimum flight path angle is limited to about 10 degrees in order to maintain the requisite line-of-sight.

The Agena B guidance system functions the same for the Atlas-Agena B as that described for the Thor-Agena B. Initial attitude references for the Agena B are established by orientation of the Atlas during vernier.

Agena B Structural Limitations

There are two salient periods of flight where the Agena and its booster receive the severest loading. These are during the atmospheric portion of the trajectory near maximum q (aerodynamic pressure) and at booster burnout. The maximum q condition gives rise to the highest bending moments on the missile while booster burnout gives the highest axial inertia force. The Agena B has been primarily designed by booster burnout conditions, but in general the booster becomes critical due to this flight condition before the Agena does.

a) Maximum Q Condition

The bending moments occurring during the atmospheric portion of flight are due primarily to angle of attack. Since the trajectories flown in the past have always been programmed for zero angle attack with respect to stationary atmosphere, the angles of attack experienced are due to wind shear and gusts aloft. Relieving inertia forces are always present during such transient pitch or yaw maneuvers which makes the accurate determination of the load distribution dependent upon the weight distribution as well as the aerodynamic characteristics.

At maximum q in previous programs the booster has limited angles of attack to 8° or less, and for current configurations proposed for NASA the limitation is as low as 3°.

b) Load and Acceleration Limit

As has been mentioned the basic structure of the Agena is designed for booster burnout. Allowing for a moderate amount of bending moment due to thrust misalignment and c.g. offset, a payload force of 45,000 lb-g's is permissible on the Agena B.

TABLE I

ATLAS SM65D/AGENA B

STAGE CHARACTERISTICS

STAGE NO.	1	2	3
1. Stage Designation	Booster	Sustainer plus Vernier	Agena B
2. Light-off Weight (less payload)[(a)], lb.	276,381	N/A	N/A
2.' Separation Weight (less payload), lb. Jettisoned Weight	N/A N/A	N/A	14,699 58[3)]
3. Loaded Stage Weight[(b)] (= ignition weight for Agena) lb.	223,393	N/A	14,641
4. Propellant Weight Usable, lb.	216,383	29,526	(SEE THOR DM 21/AGENA B DATA SHEET)
(a) Fuel	66,231	9,052	
(b) Oxidizer	150,152	20,474	
5. Propellant Weight Residual, lb.	N/A	2,180	
(a) Usable Contingency		1,496	
(b) Unusable Contingency		684	
5.' Non-Impulse Expendables, lb.	N/A	54	
(a) Propellants		--	
(b) Others		54	
6. Stage Burnout Weight (SBW)[(c)], lb.	N/A	6,828[1)]	
7. SBW + Jettisoned Weight[(d)], lb.	7,010	7,192[2)]	
8. Stage Propellant Fraction	216,383/223,393 = .968	--	

Footnotes:

(a) (b) (c) (d) (See definition in Thor DM21/Agena B footnote section)

(1) Includes the following items:

6144 lbs sustainer inert weight
672 lbs unusable contingency sustainer
12 lbs unusable contingency vernier

6828 lbs

(2) Includes the following items:

6828 lbs as above
340 lbs adapter
9 lbs retro rockets
15 lbs destruct system

7192 lbs

(3) Includes the following items:

8 lbs horizon sensor fairing
6 lbs control gas
38 lbs ullage rockets 1st burn
5 lbs propellant pre-flow 1st burn
1 lb engine start charge 1st burn

58 lbs

TABLE II

ATLAS SM65D/AGENA B

STAGE CHARACTERISTICS - PROPULSION	1 Booster	2 Sustainer	3 Vernier	4 Agena B
1. Engine designation	MA-3 XLR-89-NA-3	MA-3 XLR-105-NA-3		Bell 8096
2. Contractor	Rocketdyne	Rocketdyne	Rocketdyne	Bell Aerosystems Co.
3. Propellants	O_2, RP-1	O_2, RP-1	O_2, RP-1	IRFNA, UDMH
4. No. of chambers	2	1	2	1
5. Thrust (S.L.S.) per chamber, lb.	154,192	56,448	863	N/A
6. Thrust (vacuum) per chamber, lb.	178,300	78,513	1000	16,027
7. I_{sp} (S.L.S.), sec.	251	217	180	N/A
8. I_{sp} (vacuum), sec.	290	311	203	289.5 (-3σ)
9. Nozzle expansion ratio	8:1	25:1		45:1
10. Chamber pressure, psia	502	655		508
11. Nozzle exit area, A_c, in^2	1,644	1,675		770
12. Relite capabilities	none	none	none	one
13. Propellant feed system	turbopump	turbopump	turbopump	turbopump

TABLE III

TRAJECTORY AND PAYLOAD DATA

Vehicle: Atlas SM65D/Agena B

Mission: 300 NM Circular

Launch: AMR at 90° East of North

Payload Weight in Orbit: 5,800 lb

POSITION CONDITIONS	TIME (sec)	ALTITUDE (NM)	DISTANCE (NM)	DYNAMIC PRESSURE (lb/ft^2)	INERTIAL VELOCITY (ft/sec)	ACC. (g's)	FLIGHT PATH ANGLE[1)] (deg)	VEHICLE ATTITUDE (deg)	IMPACT POINTS (NM)
1. Launch	0	0	0	0	1,396	1.30	0	0	0
2. Booster cutoff	144.7	30.71	56.27	45	11,070	6.60	71.75	71.17	470
3. Sustainer ignition	144.7	30.71	56.27	45	11,070	1.37	71.75	71.17	470
4. Sustainer cutoff	261.5	74.10	286.60	0	17,527	2.79	84.10	80.61	1,100
5. Vernier ignition	261.5	74.10	286.60	0	17,527	0.04	84.10	80.61	1,100
6. Vernier cutoff	281.5	78.60	338.32	0	17,499	0.04	85.27	80.23	1,108
7. Agena first ignition	316.5	84.63	428.41	0	17,431	0.80	87.34	81.10	1,108
8. Agena first cutoff	545.5	82.30	1,128.1	0	26,111	2.10	90.00	81.10	--
9. Agena second ignition	3,290	300	10,976	0	24,463	2.11	90.00	90.00	--
10. Agena final cutoff	3,295.44	300	10,992	0	24,840	2.24	90.00	90.00	--

1.) Referred to local vertical

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TABLE IV

TRAJECTORY AND PAYLOAD DATA

Vehicle: Atlas SM65D/Agena B

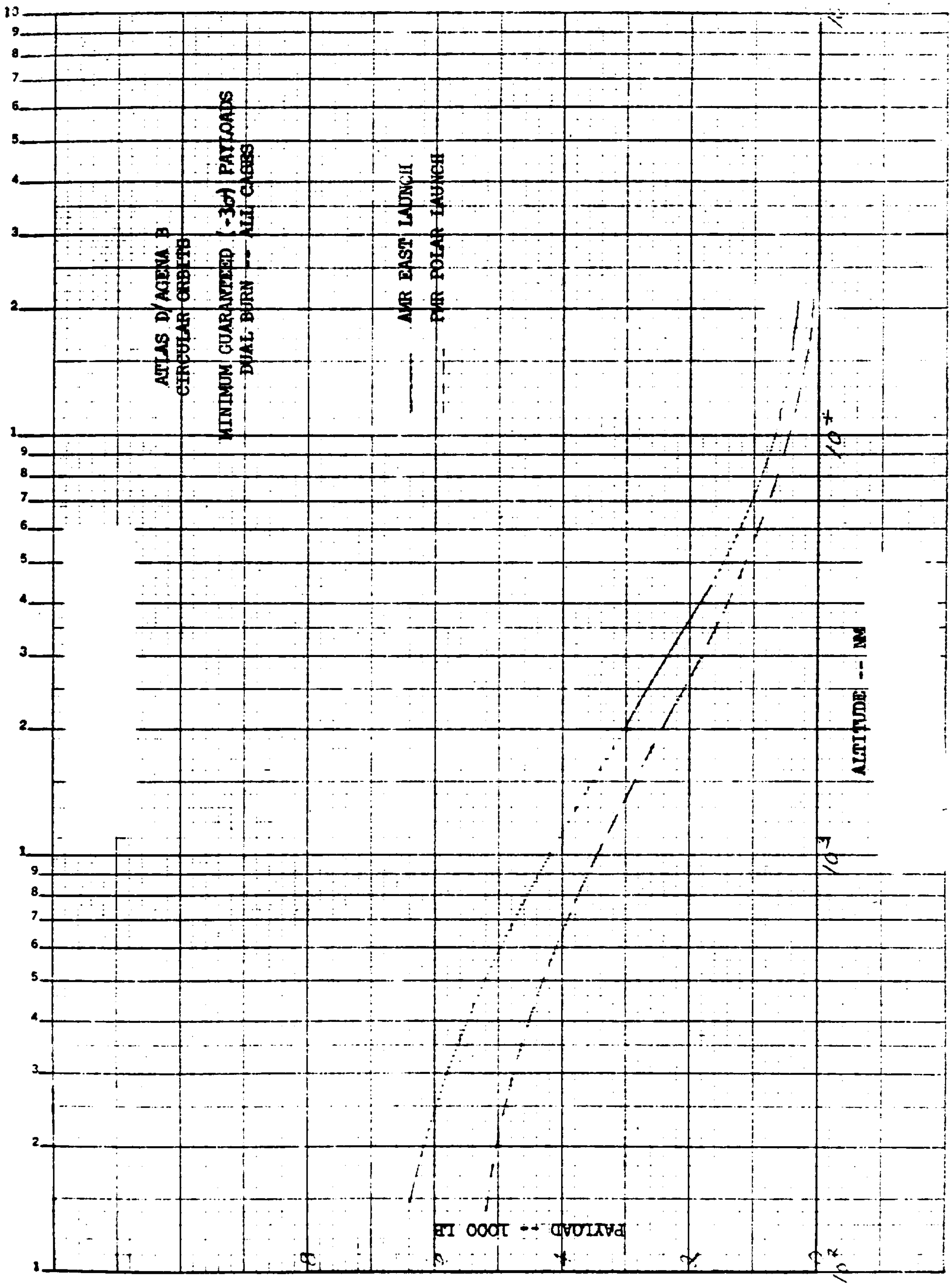
Mission: Escape

Launch: AMR 90 East

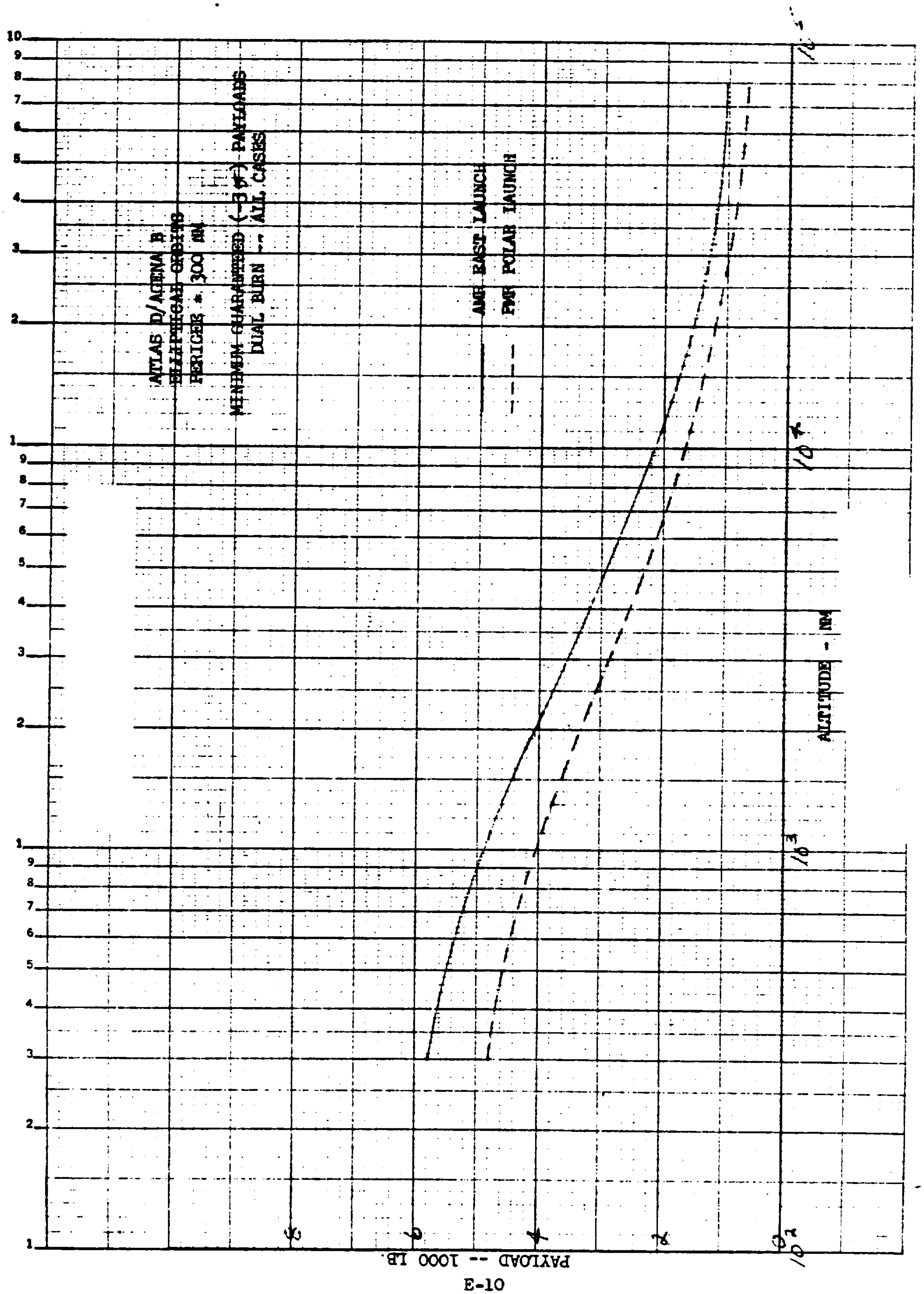
Payload Weight in Orbit: 890 lbs. -30

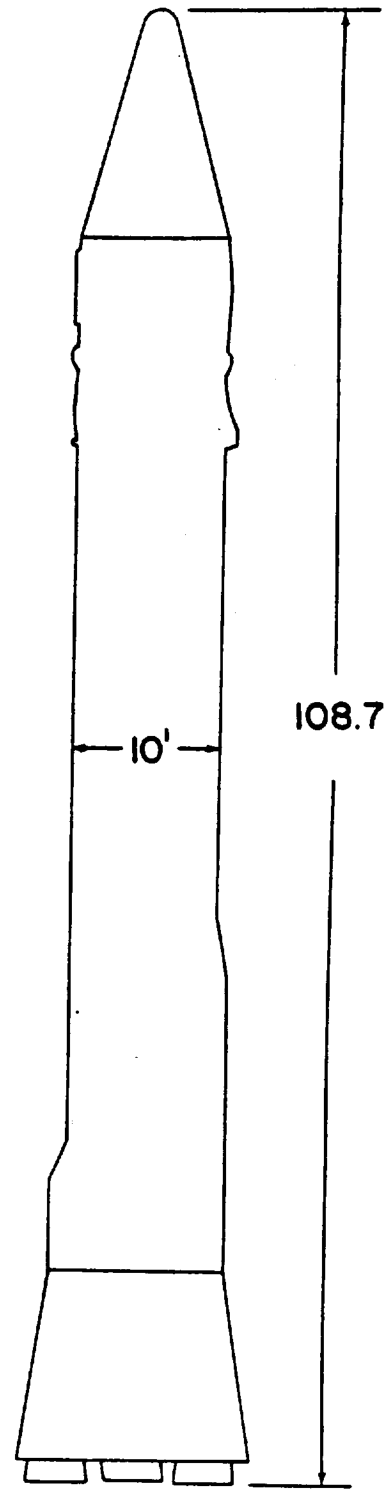
POSITION CONDITIONS	TIME (sec)	ALTITUDE (NM)	DISTANCE (NM)	DYNAMIC PRESSURE (lb/ft^2)	INERTIAL VELOCITY (ft/sec)	ACC. (g's)	FLIGHT[1)] PATH ANGLE (deg)	VEHICLE ATTITUDE (deg)	IMPACT POINTS (NM)
1. Launch	0	0	0	0	1,396	1.33	0	0	0
2. Booster cutoff	144.8	32.87	58.64	32	11,586	7.11	71.07	70.96	540
3. Sustainer ignition	144.8	32.87	58.64	32	11,586	1.48	71.07	70.96	540
4. Sustainer cutoff	261.3	84.00	302.20	0	18,875	3.32	82.37	79.71	1,600
5. Vernier ignition	261.3	84.00	302.20	0	18,875	0.05	82.37	79.72	1,600
6. Vernier cutoff	279.35	90.50	352.25	0	18,837	0.05	83.20	78.82	1,610
7. Agena first ignition	356	109.30	563.73	0	18,635	1.04	86.78	87.82	1,610
8. Agena final cutoff	591	121.2	1,414.1	0	36,153	7.0	88.00	87.82	--

1.) Referred to local vertical


ATLAS D/AGENA B
CIRCULAR ORBITS
MINIMUM GUARANTEED (-3σ) PAYLOADS
DUAL BURN -- ALL CASES
AMR EAST LAUNCH
PMR POLAR LAUNCH
ALTITUDE -- NM
PAYLOAD -- 1000 LB

48


ATLAS D/AGENA B
ELLIPTICAL ORBITS
PERIGEE = 300 NM
MINIMUM GUARANTEED (-3σ) PAYLOADS
DUAL BURN -- ALL CASES
AMR EAST LAUNCH
PMR POLAR LAUNCH
ALTITUDE - NM
PAYLOAD -- 1000 LB.



ATLAS/α-CENTAUR

GENERAL DESCRIPTION

The first stage of ATLAS/α - CENTAUR is a modified version of the ATLAS ICBM. The stage is basically a Series D ATLAS, incorporating certain structural and equipment modifications as dictated by the α CENTAUR second stage. Major differences between a Series D ATLAS and the first stage of the ATLAS/α - CENTAUR are as follows:

(a) The forward conical section of the liquid oxygen tank has been enlarged to a straight 10-foot diameter cylindrical section.

(b) A new 10-foot diameter interstage adapter together with a new ATLAS/ CENTAUR separation system has been added.

(c) The vehicle equipment pads have been modified to reflect electronic changes and to reduce over-all weight.

(d) Some airborne systems have been removed. Others have been modified to adapt their use to the CENTAUR application

CENTAUR GUIDANCE SYSTEM

The Centaur guidance system is all inertial consisting primarily of a four-gimbal all attitude inertial platform and a general purpose serial digital computer with a magnetic drum memory. The airborne guidance program is written onto the drum memory from a punched paper tape along with a pre-flight calibration and alignment program for trimming and aligning the platform prior to launch.

For the 24 hr. equatorial orbit mission the Centaur guidance system performs the following functions. During the Atlas booster phase, the vehicle pitch program is generated by the Atlas autopilot; however, the guidance system monitors the vehicle position and velocity and generates the booster staging discrete as a function of vehicle acceleration. For the Atlas sustainer stage the guidance system generates vehicle steering signals which are used to orient the thrust vector so as to reduce the position and velocity dispersions generated during the open-loop booster stage. The sustainer engine cutoff command is also given by the guidance system. After separation of the Centaur stage from the Atlas booster, the Centaur guidance system controls the vehicle during each of the succeeding three phases of powered flight necessary to place the vehicle in its final orbit. The guidance system provides steering and cutoff signals to the Centaur autopilot during the powered phases of flight and also provides an attitude reference to the autopilot prior to the second and third firings of Centaur in order that the vehicle will assume the proper attitude prior to thrust initiation.

The following table presents a typical 3 start flight sequence:

TYPICAL FLIGHT SEQUENCE; ATLAS/α - CENTAUR
(3 Start Mission)

(1) 0 → 15 sec. vertical rise and roll to desired azimuth

(2) Time dependent pitch program to booster staging (Booster staging initiated by accelerometer; a=5.8 g's)

(3) At Beco (Booster engine cut-off) + 15 sec., tank insulation panels are jettisoned; sustainer phase is flown at const. inertial attitude

(4) At Beco + 63 sec., payload shroud is jettisoned

(5) Sustainer phase is terminated by propellant depletion (Constant inertial attitude)

(6) Atlas vernier solo phase; Seco (Sustainer engine cut-off) → Seco + 9.5 sec. Centaur main engine prestart (chilldown) initiated at Seco

(7) Centaur separation and ullage rocket firing initiated at Veco (Vernier engine cut-off)

(8) First ullage rocket firing period=14.5 sec.

(9) First main engine firing; constant pitch rate

(10) Main engine cutoff; Centaur orientated "tail to sun" in parking orbit for first coast period

(11) Approx. 300 sec. prior to second main engine start vehicle is re-oriented with firing direction

(12) Ullage rockets start 42 sec. prior to main engine

(13) Engine prestart init. 20 sec. prior to main engine start

(14) Second main engine burst

(15) Vehicle orientated "tail to sun" for second coast

(16) Vehicle re-oriented with third firing direction

(17) Ullage rockets start 54 sec. prior to M.E. (Main engine)

(18) Prestart initiated 20 sec. prior to M.E.

(19) Third main engine burst

(20) Payload separation

STRUCTURAL RESTRAINTS

Structural considerations of the configuration limit the product of the angle of attack, α, and dynamic pressure, q, to approximately 1400 deg. - lb/ft^2 with 2 - sigma - winds at AMR.

The maximum permissible longitudinal and lateral acceleration factors are 7.0 and 1.0 g, respectively.

PERFORMANCE

For trajectory simulation, the flight profile from launch to parking orbit injection is selected by the use of a time dependent pitch program until booster engine staging, and an initial attitude and constant pitch rate during Centaur firing to maximize burnout weight into the parking orbit. The factors which constrain this flight profile are:

(1) aerodynamic heating and loading

(2) booster engine staging acceleration = 5.8 g's

(3) injection into a circular parking orbit at 110 nautical miles

The payload estimate includes propellant reserve for 3-sigma dispersions affecting performance capability. The individual effects of dispersions on payload are root-sum-squared to determine the propellant reserve.

The payload capability is defined as that available for:

(1) The scientific experiment

(2) Payload separation equipment

(3) Interface payload adapter

(4) Any additional requirements dictated by vehicle-payload integration.

ATLAS/CENTAUR WEIGHT DATA

STAGE CHARACTERISTICS	Stage No. 1	2	3
1. Stage designation	Booster Phase	Sustainer Phase	Centaur Stage
2. Initial wt. (1)	297,501	(5)	31,862
3. Loaded stage wt. (2)	264,296	N.A.	33,205 (7)
4. Propellant wt. (usable)			
a. Fuel wt. (main)	74,800		4,584
b. Oxid. wt. (main)	174,546		22,920
c. Addit. Prop.	86 (4)		147 (8)
5. Propellant wt. (residual)			
a. Usable contingency	N.A.	456	(9)
b. Unusable	1,134	928	658
6. Stage burnout wt.	7,192	7412	3,688
7. SBW + Jett. wt.	7,473 (3)	8731 (6)	4,269 (9) (10)
8. Stage propellant fraction, $\frac{\text{Usable propellant wt.}}{\text{Loaded stage wt.}}$	.944		.862

GROUND RULES

1. Centaur tanked to min. LH_2 ullage volume (30 ft^3), oxid/fuel burning mixture ratio = 5.0.

2. Centaur inert wt. includes an assumed 150 lb. flight instrumentation & telemetry system

SUPERSCRIPT EXPLANATIONS

(1) Weight at start of stage; less payload

(2) Gross stage weight at launch

(3) Includes 58 lbs. Centaur vented propellants, 170 lbs. jet oil, and 53 lbs. propellants thru vernier vent system

(4) Vernier expended during vernier solo

(5) Boost phase staged when T-D/W = 5.8, therefore sustainer initial wt. depends on payload wt.

(6) Includes 37 lbs. jett. oil, 417 lbs. Centaur jett. insulation, 865 lbs. jett. payload shroud

(7) Includes insulation panels and payload shroud

(8) H_2O_2 for ullage rockets

(9) Depends on the mission

(10) For 24 hr. eqt. cir. orbit - includes 135 lbs. vented prop. 288 lbs. jet. chilldown prop., 44 lbs. leaked prop, 49 lbs. atit. control prop and 65 lbs. boost pump prop.

ATLAS/ox - CENTAUR

STAGE CHARACTERISTICS

PROPULSION

	1	2	3	Remarks
1. Engine designation	MA-5	MA-5	LR-115	
2. Contractor	NAA	NAA	P & W	
3. Propellants	RP-1 & LOX	RP-1 & LOX	LH & LOX	
4. No. of chambers	5 (2 booster) (1 sustainer) (2 vernier)	3 (1 main) (2 vernier)	2	Stg. 1 consists of 2 booster & 1 sustainer chambers
5. Thrust (S.L.S.) per chamber, lb.	154.5K 57K (1) 1K (2)	57K (1) 1K (2)	N.A.	
6. Thrust (vacuum) per chamber, lb.	178.5K (2) 80.7K (1) 1.0K (2)	80.732K (1) 1K (2)	15K	
7. I_{sp} (S.L.S.), sec.	251.19 (2) 214.7 (1) 207.5 (2)	214.7 (1) 207.5 (2)	N.A.	
8. I_{sp} (vacuum), sec.	291.00 (2) 308.1 (1) 238.8 (2)	308.1 (1) 238.8 (2)	425	
9. Nozzle expansion ratio	8:1 (2) 25:1 (1) 5:1 (2)	25:1 (1) 5:1 (2)	40:1	
10. Chamber pressure, psia	502.8 (2) 652 (1) 344 (2)	652 (1) 344 (2)	300	
11. Nozzle exit area, A_c, in^2	1640 (2) 1660 (1)	1660	1122	

ATLAS/~~X~~ - CENTAUR

STAGE CHARACTERISTICS

PROPULSION (CONTD)

STAGE NO.

	1	2	3
12. Relite capabilities	None	None	Three Starts
13. Propulsion feed system	Turbo Pump	Turbo Pump	Turbo Pump
TRAJECTORY LIMITATIONS			
1. Max. q ($\frac{1}{2}\rho v^2$) at separation	26 psf. nom.	0	0 lbs/sq.ft
2. Minimum separation time after burnout of stage	0	10 sec. vernier plus 14.5 sec.	Indefinite
3. Limitation (if any) on coast time	N.A.	N.A.	Indefinite (several hours)
4. Line of sight requirements for guidance	None	None	None

TYPICAL TRAJECTORY DATA

VEHICLE ATLAS/X - CENTAUR

MISSION 300 n. mi. ALT. CIR. ORBIT
PAYLOAD 9700 LBS.

CONDITION	TIME (sec)	ALT (n.mi.)	RANGE (n.mi.)	DYN.PRESS (P.S.F.)	SPEED(1) (F.P.S.)	ACC. (g's)	TILT ANGLE(1) (deg.)	IMPACT (n.mi.)
LAUNCH	0	0	0	0	1,342	1.19	90.0	-
MAX. DYN. PRESSURE	84	7.66	3.24	730.0	2,577	2.18	60.6	-
MAX. ACCELERATION	153.9	34.71	45.37	14.5	9,682	5.80	68.0	-
PHASE 1 BURNOUT	156.9	36.51	49.15	9.8	9,793	1.06	68.3	500
INSULATION JETT.	169.9	44.06	66.00	1.4	10,172	1.24	70.2	600
PAYLOAD SHROUD JETT.	202.0	61.10	112.00	0	11,265	1.46	74.5	700
STAGE 1 BURNOUT	237.4	77.27	171.00	0	12,330	.03	78.5	1000
STAGE 2 IGNITION (1st)	251.9	82.73	196.00	0	12,249	.75	80.0	-
STAGE 2 BURNOUT	619.0	110.00	1150.00	0	25,876	2.13	90.0	-
STAGE 2 IGNITION (2nd)	3378.0	300.00	-	0	24,562	2.13	90.0	-
STAGE 2 BURNOUT	3383.0	300.00	-	0	24,900	2.18	90.0	-

Remarks: (1) Inertial Reference from local vertical
Parking Orbit Alt. = 110 n.mi.
Launch Azimuth = 90°E
Launch Site = AMR

TYPICAL TRAJECTORY DATA

VEHICLE ATLAS/α-CENTAUR

MISSION EARTH ESCAPE

PAYLOAD 2400 LBS.

CONDITION	TIME (sec)	ALT (n.m.)	RANGE (n.m.)	DYN.PRESS. (p.s.f.)	SPEED(1) (f.p.s.)	ACC. (g's)	TILT ANGLE(1) (deg)	IMPACT (n.m.)
LAUNCH	0	0	0	0	1,342	1.21	90.0	-
MAX. DYN. PRESSURE	86	8.3	5.2	804.0	3,138	2.29	64.5	-
MAX. ACCELERATION	151	30.2	50.0	36.4	9,910	5.80	73.0	-
PHASE 1 BURNOUT	154	31.6	54.0	27.5	10,039	1.09	73.2	500
INSULATION JETT.	167	37.6	72.1	8.5	10,437	1.26	74.9	600
PAYLOAD SHROUD JETT.	215	57.2	147.8	0	12,280	1.60	79.3	700
STAGE 1 BURNOUT	255	71.6	222.8	0	13,755	.03	81.6	1000
STAGE 2 IGNITION	269	76.0	251.5	0	13,698	.88	83.0	-
STAGE 2 BURNOUT	672	110.0	-	0	36,111	4.79	90.0	-

Remarks: (1) Inertial Reference from local vertical

Parking Orbit Altitude = 110 n.m.

Launch Azimuth = 90° E

Launch Site = AMR (Pad 36)

TYPICAL TRAJECTORY DATA

VEHICLE ATLAS/α- CENTAUR

MISSION 24 HR. CIRC. EQT. ORBIT

PAYLOAD 618 LBS.

CONDITION	TIME (sec)	ALT (n.mi.)	RANGE (n.mi.)	DYN.PRESS (P.S.F.)	SPEED(1) (F.P.S.)	ACC. (g's)	TILT ANGLE(1) (deg)	IMPACT (n.mi.)
LAUNCH	0	0	0	0	1,342	0	90.0	—
MAX. DYN. PRESSURE	84	8.04	4.88	800.0	3,082	2.27	64.1	—
MAX. ACCELERATION (STG. 1)	149.1	30.29	48.85	35.0	9,853	5.80	72.4	—
PHASE 1 BURNOUT	152.1	31.76	52.84	26.2	9,981	1.09	72.8	500
INSULATION JETT.	165.1	37.88	70.69	7.8	10,378	1.27	74.5	600
PAYLOAD SHROUD JETT.	213.1	57.96	145.57	0	12,221	1.61	78.9	700
STAGE 1 BURNOUT	260.6	75.99	236.03	0	14,142	.03	81.3	1000
STAGE 2 IGNITION (1st)	275.1	80.74	265.46	0	14,082	.94	82.5	—
STAGE 2 BURNOUT	536.7	110.00	995.35	0	25,538	2.25	90.0	—
STAGE 2 IGNITION (2nd)	3955.6	103.84	14,215.24	0	25,596	2.29	90.0	—
STAGE 2 BURNOUT	4038.2	109.32	14,581.99	0	33,594	4.12	87.9	—
STAGE 2 IGNITION (3rd)	22903.0	19,321.34	19,349.47	0	5,289	4.29	90.0	—
STAGE 2 BURNOUT	22938.9	19,321.31	19,346.84	0	10,088	6.74	90.0	—

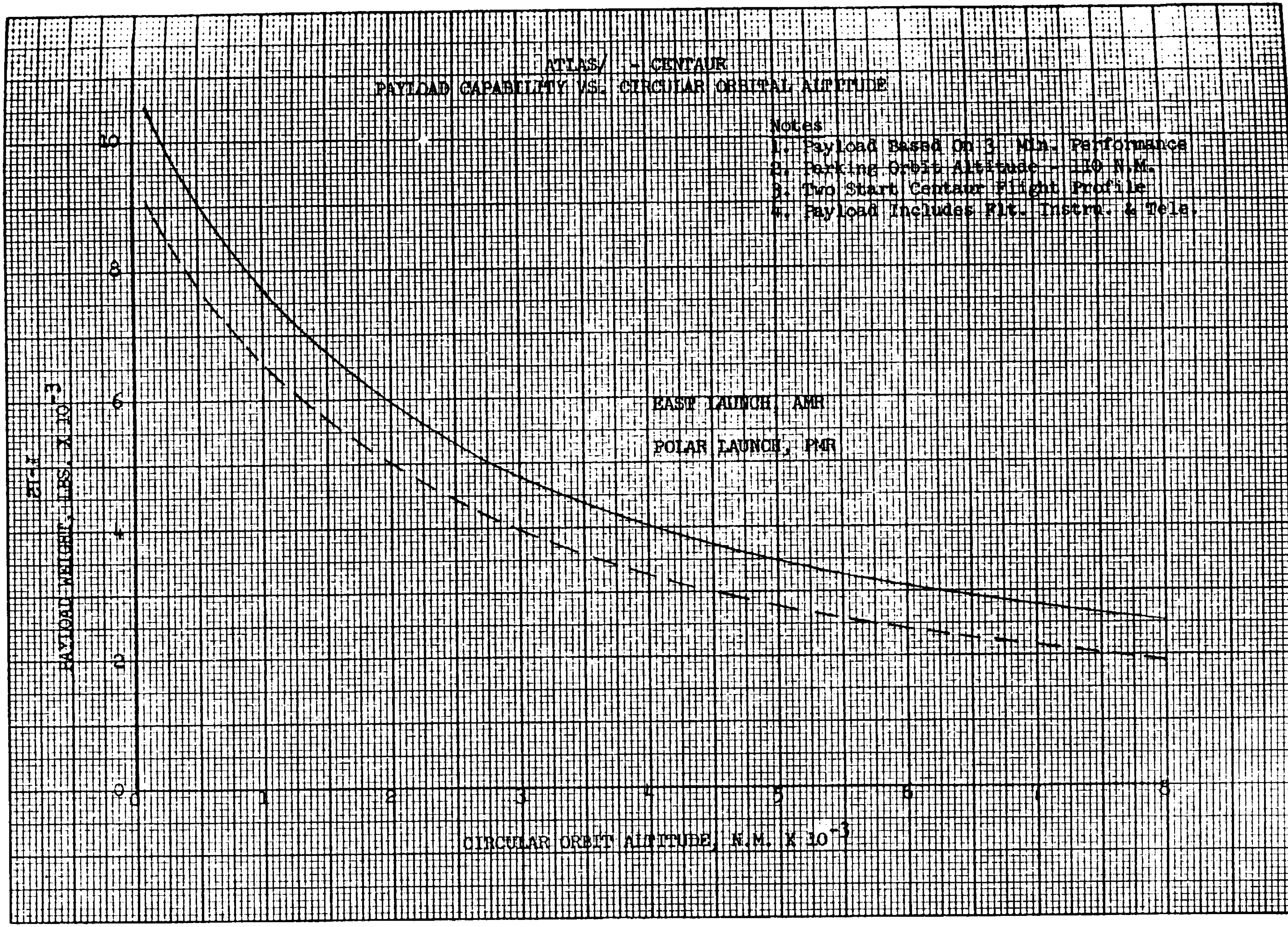
Remarks: (1) Inertial Reference from local vertical

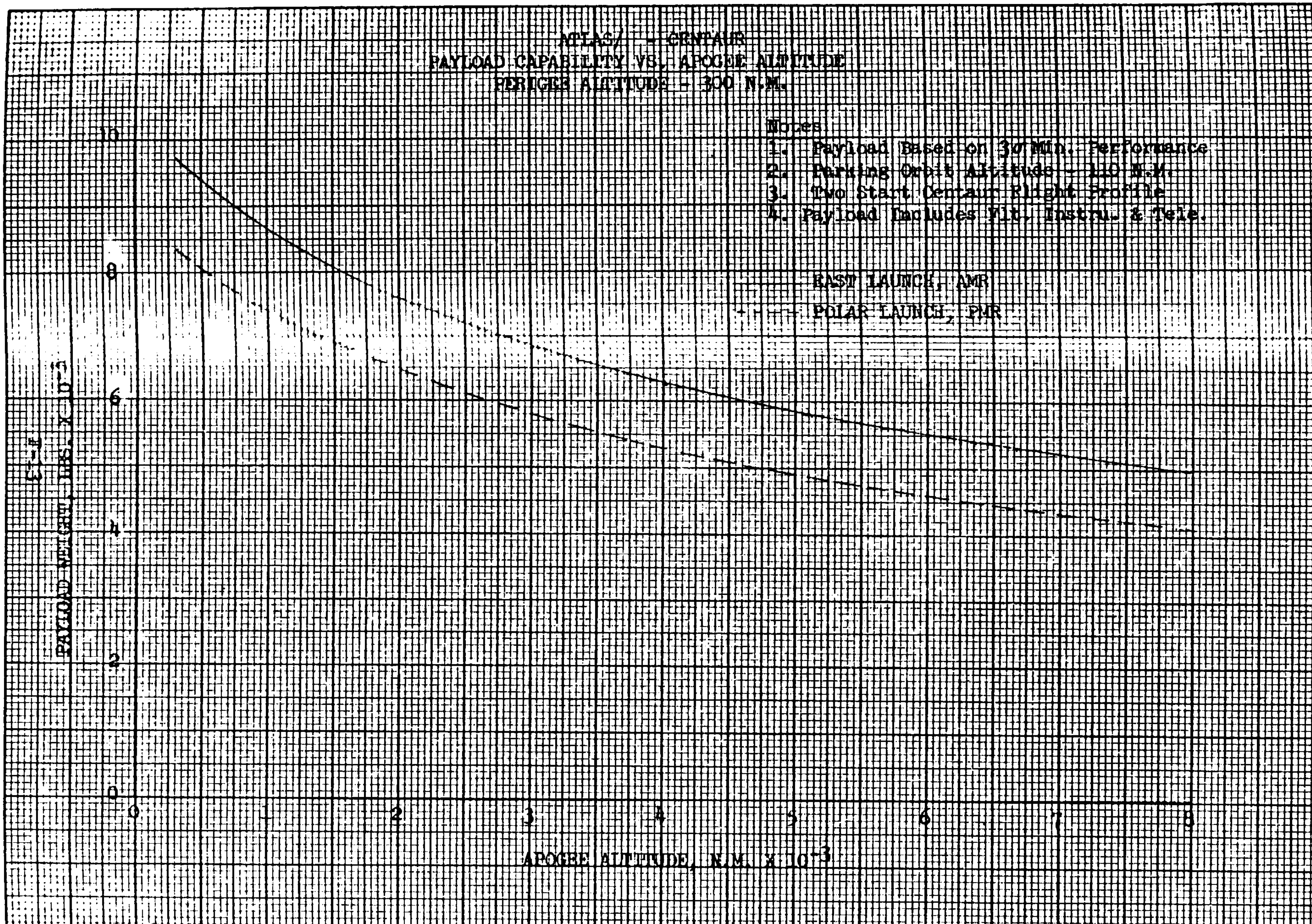
Parking Orbit Altitude = 110 n.mi.

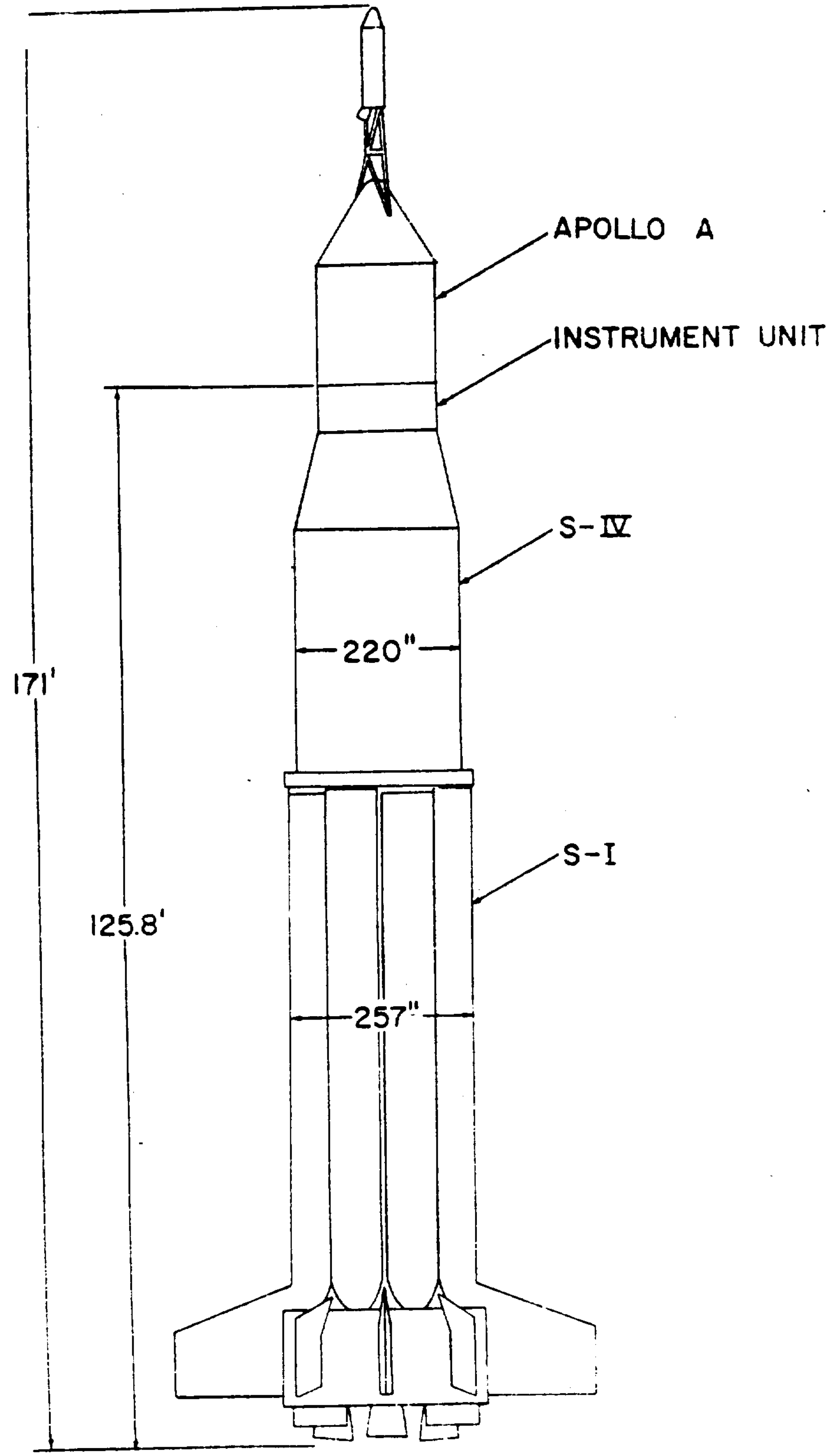
Launch Azimuth = 100° E

Launch Site = AMR (Pad 36)

Injection Longitude = 105° W


ATLAS/ - CENTAUR
PAYLOAD CAPABILITY VS. CIRCULAR ORBITAL ALTITUDE
Notes
1. Payload Based On 3 Min. Performance
2. Parking Orbit Altitude - 110 N.M.
3. Two Start Centaur Flight Profile
4. Payload Includes Flt. Instrm. & Tele.
EAST LAUNCH, AMR
POLAR LAUNCH, PMR
PAYLOAD WEIGHT, LBS. X 10^-3
0
2
4
6
8
10
0
1
2
3
4
5
6
7
8
CIRCULAR ORBIT ALTITUDE, N.M. X 10^-3


ATLAS/ - CENTAUR
PAYLOAD CAPABILITY VS. APOGEE ALTITUDE
PERIGEE ALTITUDE - 300 N.M.
Notes
1. Payload Based on 3σ Min. Performance
2. Parking Orbit Altitude - 110 N.M.
3. Two Start Centaur Flight Profile
4. Payload Includes Flt. Instru. & Tele.
EAST LAUNCH, AMR
POLAR LAUNCH, PMR
PAYLOAD WEIGHT, LBS. X 10-3
10
8
6
4
2
0
0
1
2
3
4
5
6
7
8
APOGEE ALTITUDE, N.M. X 10-3



SATURN C-1

G-1

SATURN C-1

The Saturn C-1 is a two-stage vehicle to be used primarily for low earth orbits. Both stages are designed to continue flight with one engine shut down. The reduced performance would in many instances result in an alternate mission.

Payloads for both 7 and 8 engine first stage operations are presented to show the effects of first stage engine-out on maximum payloads. All payloads presented are for trajectories which include 3% velocity flight propellant reserve.

The present second stage configuration does not have restart or coast capability. Therefore, circular orbit payload data is presented for directly ascending into the desired orbital altitude burning all the way. By the addition of a coast attitude control system and changes to the flight sequencer circuitry and the inflight pressurization schedules, the stage can be made restartable. It is estimated that an additional 300 lbs. of weight in the S-IV stage is required. Payloads are therefore presented for trajectories utilizing a 100 nautical mile parking orbit and a Hohmann transfer to orbit altitudes above the parking orbit. Payloads are also presented for elliptical orbits with a 100 n.mile perigee.

SATURN VEHICLE GUIDANCE SYSTEM - SALIENT FEATURES

The SATURN guidance system is being planned to use all inertial sensing components and a basic digital computer in combination with a mission navigation adaptor to meet the varied proposed SATURN missions.

This hardware will be used to steer the vehicle, utilizing a guidance scheme called the path adaptive and propellant minimizing approach (PA/PM). Rather than precomputing a best trajectory for a given vehicle to fly and then devising a guidance system to hold this trajectory the PA/PM scheme will continually determine the new optimum trajectory from on-board sending elements. This will then allow the vehicle to arrive at the injection point with the required velocity vector despite disturbances such as one engine out in the booster.

SATURN C-1

STAGE CHARACTERISTICS

STAGE CHARACTERISTICS	1	2
1. Stage designation	S-I	S-IV
2. Light-off weight (less payload)lb.	1,078,296	113,431
3. Loaded stage weight	964,865	113,431
4. Propellant weight (useable) lb.	847,135	100,000
5. Propellant Weight(Residual), lb.	15,799	982
a. Usable contingency for mixture ratio shift	5,100	500
b. Unusable	10,699	482
6. Stage Burnout weight, lb.	114,819(1)	15,682(3)
7. SBW - Jettisonable weight	117,730(2)	16,431(4)
8. Stage propellant fraction, $\frac{\text{Usable propellant wt}}{\text{Loaded stage wt}}$ $\frac{847,135}{964,865}$	.878	.882

SATURN C-1

(1) 1st stage burnout weight (SBW)

(a)	Stage dry weight with fins	90,000
(b)	S-I/S-IV interstage section	1,850
(c)	First stage retro rockets	3,000
(d)	Reserve for mixture ratio shift	5,100
(e)	Trapped propellants	10,699
(f)	Trapped pressurants	4,170
		114,819

(2) SBW + Jettisonables

(a)	SBW	114,819
(b)	Second stage chill-down and vented gases	357
(c)	Fuel for lubrication	552
(d)	Trapped propellants expended	319
(e)	Usable pressurants	1,683
		117,730

(3) 2nd stage burnout weight (SBW)

(a)	Stage dry wt	11,700
(b)	Reserve for mixture ratio shift	500
(c)	Residuals	482
(d)	Instrument unit	3,000
		15,682

(4) 2nd stage (SBW) + Jettisonables

(a)	SBW	15,682
(b)	Helium heater propellants	60
(c)	Lost propellants during SEP/start	689
		16,431

66

C-1
VEHICLE DESIGNATION

STAGE CHARACTERISTICS

PROPULSION

	Stage No. 1	2	
1. Engine designation	H-1	RL10-A-3	
2. Contractor	Rocketdyne Div. North American Aviation Corp.	Pratt & Whitney	
3. Propellants	LOX/RP-1	LOX/LH_2	
4. No. of chambers	8	6	
5. Thrust (S.L.S.) per chamber	188,000	--	
6. Thrust (vacuum) per chamber	--	15,000	
7. I_{sp} (S.L.S.) = $F_{sl}/dm/dt$	255.5 (System I_{sp})	--	
8. I_{sp} (vacuum)	--	420	
9. Nozzle expansion ratio	8:1	40:1	
10. Chamber Pressure (nominal)	645 psia	300 psia	
11. Nozzle exit area, A_e	11.42 ft^2	8.55 ft^2	
12. Relite Capabilities	None	None	
13. Propulsion Feed System	Turbopump	Turbopump	
TRAJECTORY LIMITATIONS			
1. Max q ($1/2\ \rho v^2$) at separation	26 kg/m^2	--	
2. Minimum separation time after burnout	1.7 sec		
3. Limitation (if any) on coast time	1.7 sec (control limitation)		
4. Line of sight requirements for guidance	None after launch -- all inertial system		

TRAJECTORY AND PAYLOAD DATA

(EASTWARD AMR LAUNCH)

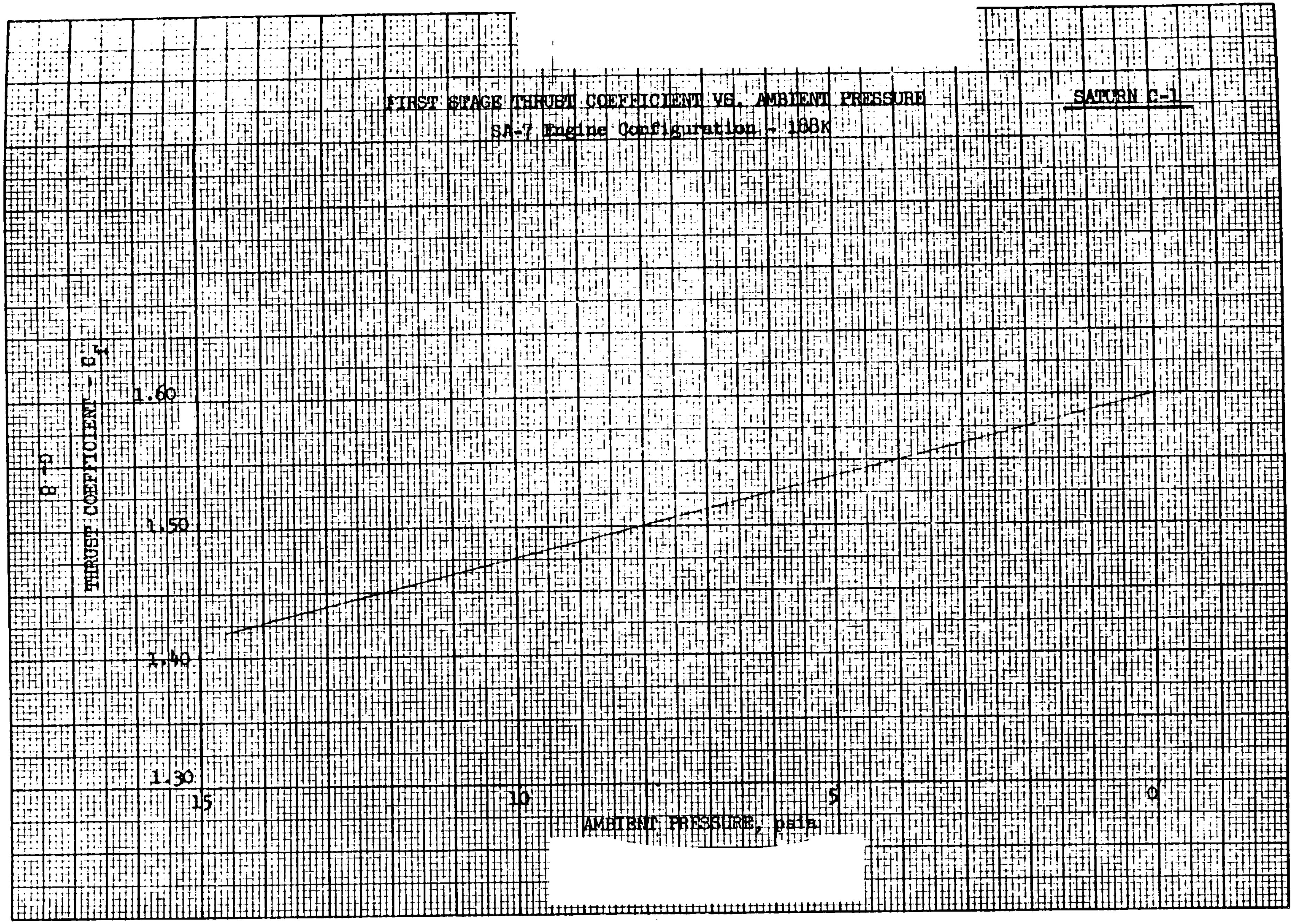
VEHICLE: SATURN C-1

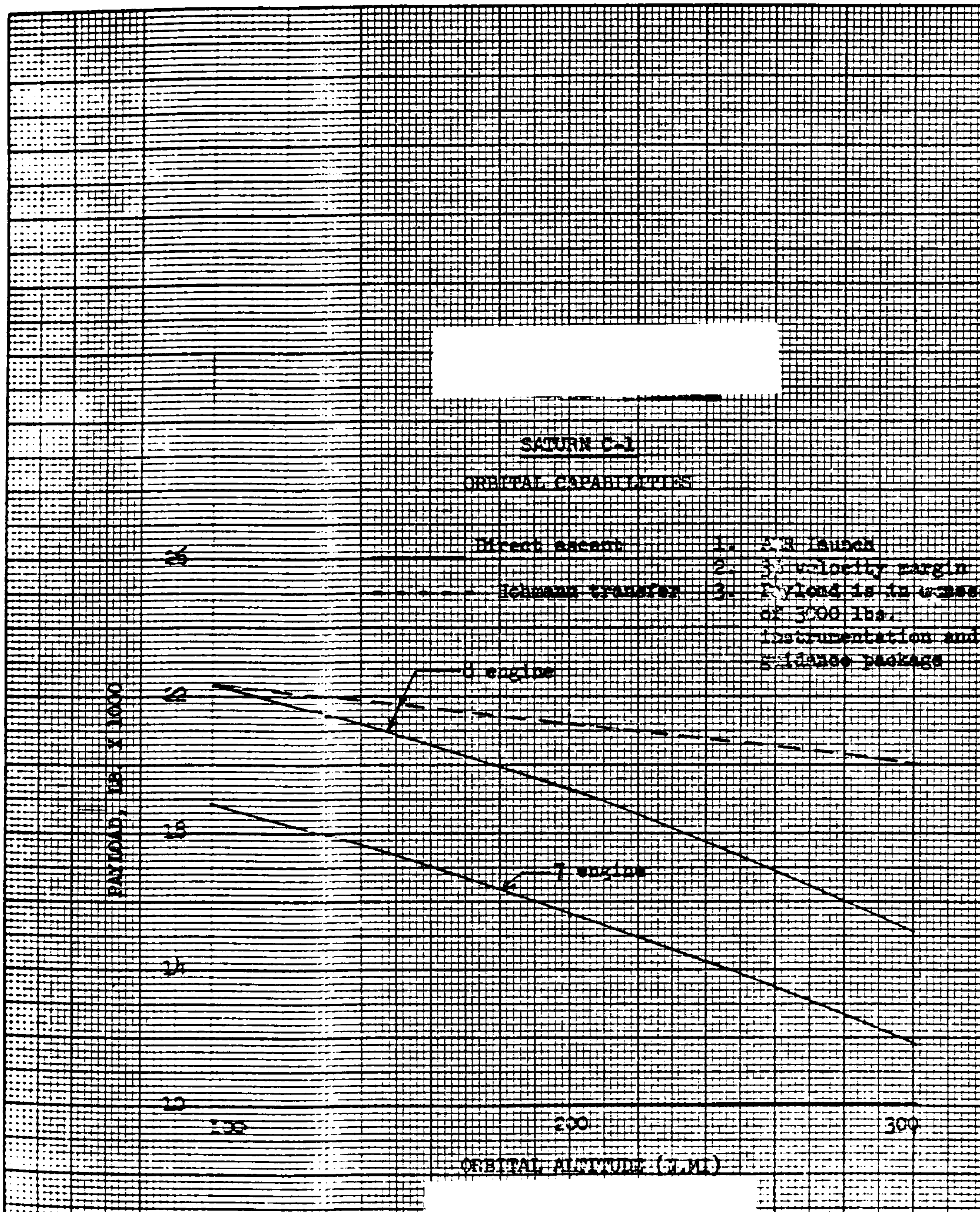
MISSION: 100 N.MI. CIRCULAR ORBIT

PAYLOAD: 22,300*

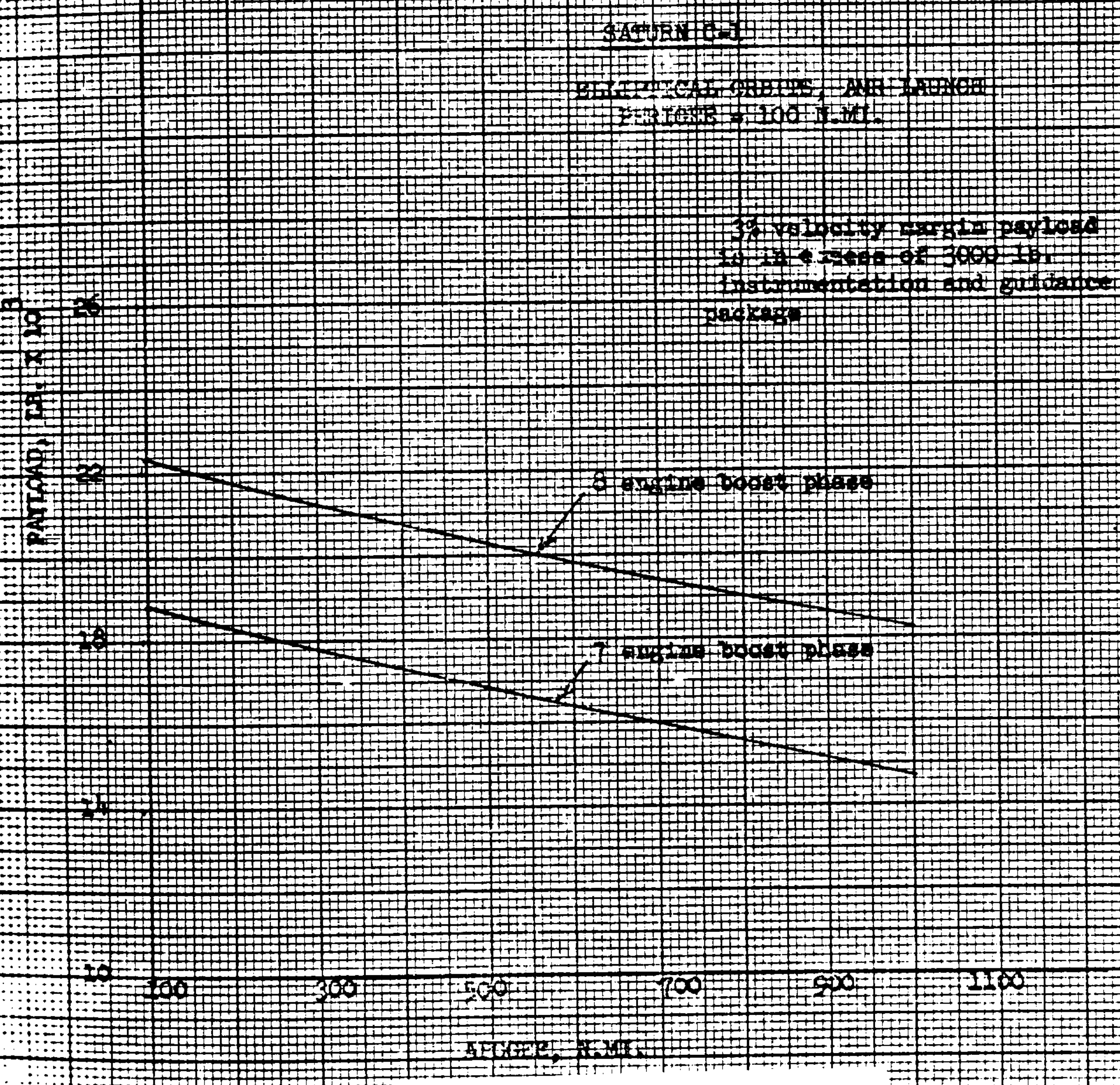
Position condition	Time (sec)	Altitude, ft.	Distance NM	Dynamic Pressure lb/ft^2, $\frac{1}{2}\rho v^2$	Speed ft/sec	Acc g's	Tilt angle (referred to local vertical), deg.	Impact points, d (nmi)
1. Launch	0	0	0	0	0	1.36	0	---
2. 1st stage max q	66	36,417	2.4	783.2	1503	2.05	35.8	---
3. 1st stage cut-off	140.5	202,427	46.3	21.3	8678	6.08	66.0	---
4. 2nd stage ignition	146.5	223,753	54.3	0	10470	.64	69.6	---
5. 2nd stage cut-off	609.2	607,282	1229.5	0	25568	2.21	90.0	---

* Does not include 3700 lbs available for abort tower and jettisoned at S-I burnout

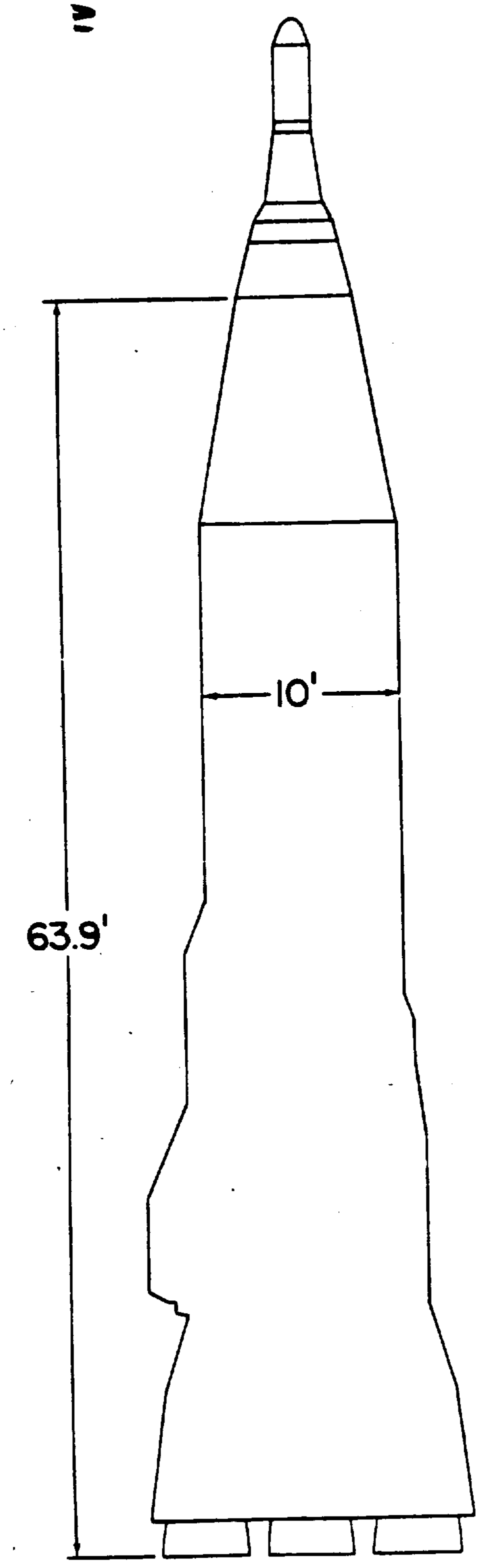

FIRST STAGE THRUST COEFFICIENT VS. AMBIENT PRESSURE
SATURN C-1
SA-7 Engine Configuration - 188K
THRUST COEFFICIENT - C_f
1.60
1.50
1.40
1.30
15
10
5
0
AMBIENT PRESSURE, psia


SATURN C-1
ORBITAL CAPABILITIES
Direct ascent
Hohmann transfer
1. launch
2. velocity margin
3. Payload is in excess of 3000 lbs. instrumentation and guidance package
8 engine
7 engine
26
22
18
14
10
PAYLOAD, LB. X 1000
100
200
300
ORBITAL ALTITUDE (N.MI)

7



SATURN C-1
ELLIPTICAL ORBITS, AND LAUNCH
PERIGEE = 100 N.MI.
3% velocity margin payload is in excess of 3000 lb. instrumentation and guidance package
8 engine boost phase
7 engine boost phase
PAYLOAD, LB. X 10^3
26
22
18
14
10
100
300
500
700
900
1100
APOGEE, N.MI.



ATLAS XSM-65E

ATLAS E/ALWASS CONFIGURATION

The Atlas E/Alwass configuration is a proposed modified Atlas E that could be used for low earth orbits. It utilizes the Centaur guidance system in place of the conventional Atlas E system. There are some modifications to the E series pod configurations.

The vehicle configuration is quite similar to the Atlas E and since no sketch of the Alwass configuration was available the Atlas E sketch is presented.

ATLAS E/ALWAAS

STAGE NO.

STAGE CHARACTERISTICS

	1	2	Remarks
1. Stage Designation	Booster Phase	Sustainer Phase	
2. Initial Weight (less payload)	262,564	NA	
3. Loaded Stage Weight	NA	NA	
4. Propellant weight (usable)			
a. Fuel weight, main	← 75,342 →		
b. Oxidizer weight, main	← 173,298 →		
5. Propellant weight (residual)			(1) 170 lbs jett. oil and 53 lb. propellant vented thru vernier vent system
a. Usable contingency	NA	327	
b. Unusable contingency	1,050	671	
6. Stage Burnout Weight	7,232	6140	
7. SBW + Jettisonable Weight	7,455 (1)	6469 (2)	(2) 292 lbs. jettisoned after booster separation, 37 lbs. jett. oil.
8. Stage Propellant Fraction, Usable propellant weight / Loaded stage weight	.948		

VEH[illegible] DESIGNATION - ATLAS E/ALWAAS

	AGE NO. 1	2	Remarks
STAGE CHARACTERISTICS PROPULSION (separate listing for main engine or engines and vernier engines)			
1. Engine designation	MA-3	MA-3	
2. Contractor	NAA	NAA	
3. Propellants	RP-1 & lox	RP-1 & lox	
4. No. of chambers	5	3	Stage 1 consists of (2) booster, (1) sustainer and (2) vernier chambers. (verniers canted out 30°. Stage 2 consists of (1) sustainer and (2) vernier chambers. (verniers canted out 50°).
5. Thrust (S.L.S.) per chamber	165K lbs.(2) 57K lbs.(1) 1K lbs.(2)	57K lbs(1) 1K lbs(2)	
6. Thrust (vacuum) per chamber	199K lbs. (2) 79K lbs. (1) 1K lbs. (2)	79K lbs.(1) 1K lbs.(2)	
7. I_{sp} (S.L.S.)	253.5 sec.(2) 215.5 sec.(1) 207.5 sec.(2)	215.5 sec.(1) 207.5 sec.(2)	

VEHICLE DESIGNATION - ATLAS E/ALWAAS

STAGE CHARACTERISTICS PROPULSION (Continued)	STAGE NO. 1	STAGE NO. 2	Remarks
8. I_{sp} (vacuum)	290.0 sec.(2) 308.9 sec.(1) 238.8 sec.(2)	308.9 sec.(1) 238.8 sec.(2)	
9. Nozzle expansion ratio	8:1 (2) 25:1 (1) 5:1 (2)	25:1 (1) 5:1 (2)	
10. Chamber pressure	533 psia (2) 650 psia (1) 344 psia (2)	650 psia (1) 344 psia (2)	
11. Nozzle exit area, A_c	1640 in^2 (2) 1680 in^2 (1) not available	1680 in^2 (1) not available	
12. Relite capabilities	None	None	
13. Propulsion feed system	Turbo pump	Turbo pump	
TRAJECTORY LIMITATIONS			
1. Max. q ($\frac{1}{2}\rho V^2$) at separation	See remarks	NA	Maximum allowable αq = 240 psf deg (α = angle of attack)
2. Minimum separation time after burnout of stage	NA	NA	
3. Limitation (if any) on coast time	NA	NA	
4. Line of sight requirements for guidance	None	None	

TYPICAL TRAJECTORY DATA

VEHICLE ATLAS-E

MISSION 100 N.M. ALT. CIRC. ORBIT

PAYLOAD 3460 LBS.

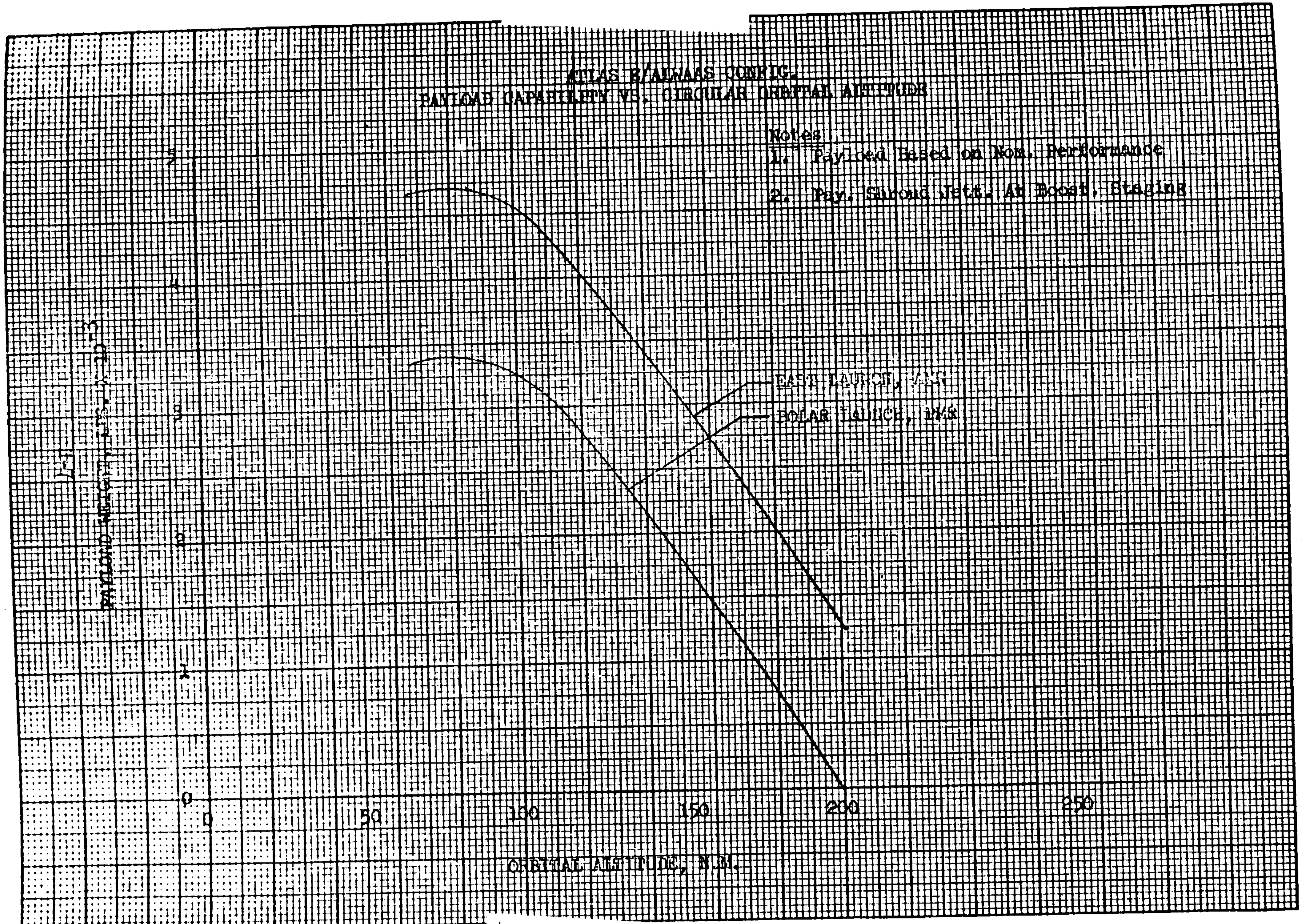
CONDITION	TIME (sec)	ALT (n.m.)	RANGE (n.m.)	DYN. PRESS. (p.s.f.)	SPEED(1) (f.p.s.)	ACC. (g's)	TILT ANGLE(1) (deg)	IMPACT (n.m.)
LAUNCH	0	0	0	0	1,254	1.47	90	-
MAX. DY. PRESS.	56	5.4	2.2	1010	1,982	2.28	50.6	-
PHASE 1 BURNOUT (Booster)	120	29.6	39.4	43	8,479	6.60	64.2	500
STAGE 1 BURNOUT (Sustainer)	329	99.4	506.9	0	25,586	8.09	90	-

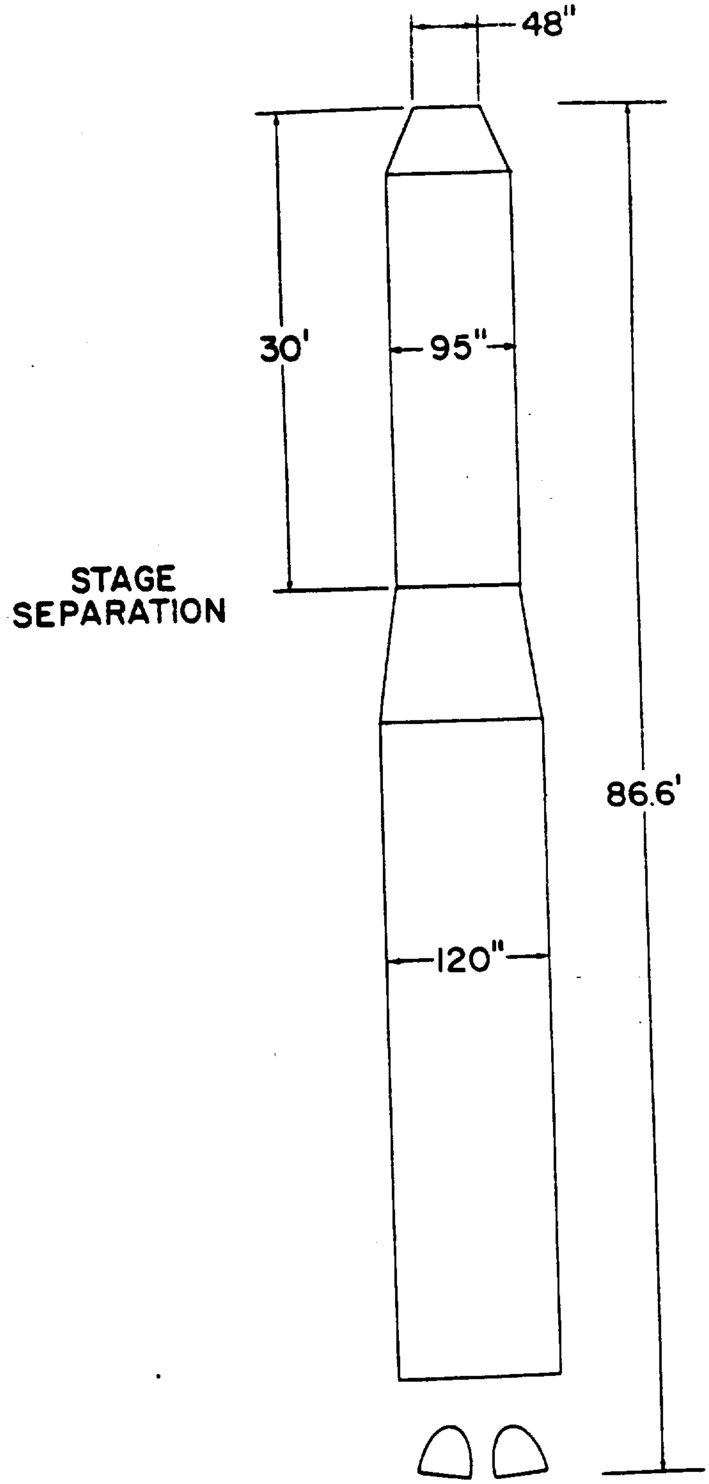
Remarks: (1) Inertial Reference from local vertical

Launch Site - PMR

Launch Azimuth = 183°E

Polar Orbit


ATLAS B/ALWAAS CONFIG.
PAYLOAD CAPABILITY VS. CIRCULAR ORBITAL ALTITUDE
Notes
1. Payload Based on Nom. Performance
2. Pay. Shroud Jett. At Boost. Staging
EAST LAUNCH, AMR
POLAR LAUNCH, PMR
PAYLOAD WEIGHT, LBS. X 10-3
0
1
2
3
4
5
0
50
100
150
200
250
ORBITAL ALTITUDE, N.M.



TITAN I

TITAN VEHICLE DATA

I GENERAL

The Titan data presented herein is for SM-68 and XSM-68B. The terms SM-68, Titan I, and Titan A are used interchangeably as are the terms Titan B, Titan II and XSM-68B.

The information presented represents current best estimates. However, some changes are still to be expected in the basic ICBM descriptions (particularly for XSM-68B) and further variations will result from redesign to accommodate specific space missions.

TABLE I

TITAN I or A, VEHICLE DESIGNATION SM-68

STAGE CHARACTERISTICS

	First Stage	Separation Phase	Second Stage	Vernier Phase	Remarks
1. Stage Designation	SM-68 First Stage	---	SM-68 Second Stage	---	
2. Light-off Weight (Less Payload)	217,163	45,296	45,082	5,344	Note 1
3. Loaded Stage Weight	171,250	45,136	44,922	5,184	Note 2
4. Propellant Weight (Usable)	161,688	152	39,738	364	Note 3
(a) Fuel Weight	49,750	---	12,227	264	
(b) Oxidizer Weight	111,938	---	27,511	100	
4A. Propellant Weight Expended Without Impulse Credit During First Stage	---	---	266	---	Note 4
(a) Lox bleed	---	---	134	---	
(b) Vernier Operation During Stage I	---	---	132		
5. Propellant Weight (Residual)	2071	---	407	---	
~~(a)~~ Usable Contingency	451	---	110	---	
(b) Unusable	1620	---	297	---	
6. Stage Burnout Weight	9,562	44,984	5,184	4,820	
7. S.B.W. + Jettisonable Weight	9,913	45,046	5,184	4,980	Note 5
8. Stage Propellant Fraction	0.944	---	0.885	---	Note 6

NOTES:

1. Total weight of vehicle at lift-off and at ignition of succeeding powered phases less payload.
2. First stage loaded weight does not include 3964 lbs. start and ground losses. Lo[illegible] stage weights do not include propellant weight expended without impulse credit during first stage.
3. Second stage propellant weight includes 44 lbs. of start propellant.
4. 134 lbs. of lox is bled through the 2nd stage gas turbines during Stage I flight. Verniers are ignited prior to Stage I burnout and consume 132 lbs. of propellant before Stage I burnout.
5. Weights jettisoned are as follows:

Time (Ref. lift-off) (sec)	Event	Jettisoned Weight (lb)
134.47	First stage burnout	9,913
134.47	End Separation phase	62
140.97	Second stage burnout	0
341.46	Vernier burnout	4,980

First stage burnout weight and jettisonable weight does not include propellant wt expended from Stage II tank without impulse credit during 1st stage operation.

6. First stage: 161,688/171,250 ; second stage: 39,738/44,922

TABLE II
TITAN A (SM-68)
PROPULSION CHARACTERISTICS

Stage No.	1	Separation		2	Vernier
Stage Designation	Titan			Titan	
Stage Contractor	Martin			Martin	
Engine Designation	LR-87-AJ-3	Rato (5) Bottles	Stage II (5) Turbine Exhaust	LR-91-AJ-3	Stage II Turbine Exhaust
Engine Contractor	A.G.C.			A.G.C.	
Propellants	LP_2/RP-1			LP_2/RP-1	
No. of Chambers	2			1	
Thrust/Chamber (lb)					
S.L.	150,000 (1)	---		---	---
Vac.	172,193 (1)	9300	642	80,603 (6)	975 (8)
Specific Impulse					
S.L.	249.5	---		--- (7)	---
Vac.	286.0	214.6	88.2	310	134
Nozzle Area Ratio	8			25	
Chamber Pressure (psia)	587 (2)			685	
Nozzle Exit Area (in^2)	1462 (3)			1668	
Relite Capability	None			None	
Propulsion Feed System	Pump Fed			Pump Fed	
Total Axial Thrust (lb)					
S.L.	299,655	---		---	---
Vac.	343,900	8844.8	603.2	80,603	916
Total Decay Rate (lb/sec)	1203.4 (4)	43.33	7.28	258.1	7.28

NOTES:
(1) Includes 600# turbine exhaust thrust
(2) Injector face
(3) In addition each first stage engine has a turbine
(4) Includes 1#/sec second stage lox bleed
(5) Separation is accomplished with two rato bottles canted at 18 degrees and the Stage II turbine gases exhausted through 4 nozzles canted at 20 degrees
(6) The chamber thrust is 80,000#. The axial thrust of the four verniers is 603#.
(7) Defined as chamber thrust over total flow
(8) Stage II turbine gases exhausted through 4 nozzles canted at 20 degrees

TOLERANCES SM-68

TABLE III

Parameter	First Stage	Second Stage
Thrust	± 3.0% (Sea Level)	± 3.0% (Vacuum)
Specific Impulse	± 1.8% (Sea Level)	± 0.97% (Vacuum)
Propellant Loading	± 3500 lb	± 375 lb
Propellant Utilization	Note (1)	Note (2)
Drag Coefficient	± 10%	± 10%
Dry Weight	± 100 lb	± 70 lb
Wind and Guidance	± 100 ft/sec (3)	

(1) Nominal Outage = 0.28% (Fuel Bias), Max Outage = 1%

(2) Nominal Outage = 0.28% (Fuel Bias), Max Outage = 1%

(3) Combined Effect of Wind and Guidance on Burnout Velocity

II TRAJECTORY CONSTRAINTS FOR TITAN A

Performance evaluations must recognize several trajectory constraints imposed by hardware limitations. These constraints are sensitive to upper stage configurations as well as operating ground rules and are therefore difficult to define in general. The following paragraphs discuss some of the more significant considerations.

(a) Second Stage Separation

For any given configuration at staging time and for any given staging method, permissible combinations of dynamic pressure (q) and angle of attack (α) may be determined. The nominal trajectory must be such that $q\alpha$ limits are not exceeded when the effects of trajectory dispersions are considered.

Staging sequence for the Titan A ICBM is as follows:

Booster cutoff signal at time T
Rato bottle ignition at time T + 2.5 seconds
Blow separation bolts at time T + 2.8 seconds
Stage II engine ignition at time T + 3.8 seconds

The above timer controlled sequence allows for tolerances in thrust build-up and decay variations and assures positive acceleration at all times. A minimum separation of approximately 10 feet exists between stages when the second stage engine ignites.

Since the second stage is unstable prior to sustainer engine ignition, aerodynamic pitching moments must be kept within allowable limits. For the ICBM, q = 50 psf and α = 5 deg or any combination producing the same $q\alpha$ product is considered acceptable for the nominal trajectory. For space configurations, allowable $q\alpha$ will generally be reduced due to forward translation of the aerodynamic center of pressure.

Staging α may be large if booster flight is controlled only by an autopilot involving large gyro drift rates. However, if guidance is employed during booster operation, α may be kept small at staging and q may be allowed to increase accordingly. The final limiting value for q is that which produces negative g's during separation due to excessive aerodynamic drag. In its present design, Titan A cannot tolerate negative g's which tend to cause pump cavitation.

(b) Aerodynamic Heating

Acceptable nominal trajectories for the Titan ICBM's are limited by aerodynamic heating to a value of approximately 100 x 10^6 for the quantity $\int_0^t q V_{rw} dt$ where V_{rw} = relative wind velocity. Detailed analysis is necessary to determine the effects of trajectory dispersions and atmospheric variations on the maximum skin temperatures.

Forward transition sections are most subject to aero heating. In general, these transitions would be replaced with special adapters for space applications. Tolerance to heating is thereby increased if the adapters are appropriately designed. Further increases to allowable heating are possible by addition of insulation or ablative finishes to critical areas.

(c) Structural Limitations

Flight through a jet stream wind profile imposes severe structural bending loads as well as dynamic stability requirements. In general, the Titan autopilots can operate satisfactorily on any trajectory which does not violate structural capabilities. In some cases, gain changes may be necessary to accommodate changed body bending characteristics. Very high angles of attack present difficulties due to extreme nonlinear behavior of aerodynamic normal force coefficients.

For a given configuration, permissible trajectories may be related to the maximum value of the product $q\alpha$. Acceptable values for $(q\alpha)_{max}$ for the ICBM nominal trajectory are about 6500 lb-deg/ft^2 for Titan A flown through the 1956 Sissenwine wind profile. The value is based on detailed studies of all applied loads and allows for trajectory dispersions.

$(q\alpha)_{max}$ normally occurs at or very near max q (by definition in some cases). Permissible q then depends directly on the severity of the wind profile which must be endured. For some space applications, wind requirements may be relaxed considerably from those imposed on the ICBM. Consequently, lower trajectories may be flown with higher values of q_{max}.

Permissible values of $(q\alpha)_{max}$ for space applications depend on the upper stage configuration. Total bending loads must be calculated for each specific application (aerodynamic moments and engine thrust moments) and superimposed on axial loads at each missile station to determine structural integrity.

(d) Radio Guidance

The Titan A utilizes a BTL pulse radar guidance system. To obtain acceptable accuracies for the ICBM mission, the BTL guidance system requires that elevation angle (angle between line of sight to the missile and the local horizontal measured at the ground tracking station) be not less than 13 deg and that look angle (angle between line of sight to the ground tracking station and the missile centerline measured at the airborne antenna) be no less than 5 deg. In addition slant range between the ground tracking station and the airborne antenna is limited to 700 n.mi.

Shaping trajectories for space missions to satisfy these stringent constraints results in substantial performance penalties (even if radio guidance is employed only through second stage operation). However, some relaxation is possible if guidance accuracy requirements for space missions are less severe. In particular, elevation angle may be allowed to drop as low as 3 deg for some applications. More information on this subject is contained in the following discussions of Titan guidance system characteristics.

III GUIDANCE SYSTEM DESCRIPTION FOR TITAN A

1. Description

Titan A uses a BTL pulse radar guidance system. Airborne equipment is located in the front compartment of Stage II. Ground equipment consists of a radar transmitter and receiver, tracking antenna, computer and associated equipment. The airborne system contains a transmitter, receiver, decoder and associated wave guides and slot antennas.

A few of the characteristics of the ground equipment are listed below:

Coverage - Range	Azimuth	Elevation
0-700 n.mi.	6400 mil	-180 to 1600 mil

Standard deviation. These deviations are for S/N of 35 db. Propagation errors not included.

	Range	Azimuth	Elevation
Relative (1 sec. smoothing)	2 mil	.03 mil	.03 mil
Absolute	50 ft	.07 mil	.08 mil

Maximum Tracking rates.

Range	Azimuth	Elevation
25,000 ft/sec	100 mil/sec	100 mil/sec

Transmitter.

Frequency	x-band (tunable)
Peak Power	250 kilowatts (minimum)
Pulse Length	0.25 micro second
Repetition Rate	99.2 pulse groups/sec

Antenna.

Type	Parabolic reflector
Size	94 3/8 inches diameter
Beamwidth	1 degree one way, -3 decibel level
Gain	44 decibel

Receiver.

Noise Figure	14 decibel
Bandwidth	10 megacycles

Data Output.

Coordinates	Range, Azimuth, Elevation
Display	3"A" scopes, 10 inch TV boresight monitor, rectangular plotting board

Communications to Missile.

Address codes	4
Discrete commands	5 plus steering channels
Modulation	4 pulse, pulse-position code

Computer. The computer is a fixed binary-point digital machine with the following characteristics:

(1) Drum storage - 8 groups - 1024, 24 bit words per group;
(2) 200 kilocycle clock for permanent storage drum indexing and computer operation;
(3) Internal computer operations synchronized with ground guidance equipment;
(4) Storage slot addresses provided by256 word coincident-current magnetic core matrix.

The airborne transmitter operates in the x-band at a peak power of 3 KW. The receiver has a threshold of -60 dbm. The airborne system uses two flush mounted slot antennas. The dorsal antenna, used at long range, has a gain of 20 db; the ventral antenna has a gain of 0 db. The airborne system weighs 150 lbs. and draws 220 watts of power.

2. Flight Operation

Shortly after liftoff, a programmer switch connects the guidance decoder yaw output to the control system input. Yaw steering signals from the ground radar will cause the missile to roll at 12^{0}/sec until it is aligned to the proper azimuth. A maximum roll of $\pm$ 200 deg. can be executed. When the roll program is completed, pitch and yaw steering outputs are connected to their respective control system inputs. Open loop pitch programming will then be performed by signals from the ground radar. Shortly after maximum q, first stage closed loop pitch and yaw guidance are initiated. Steering signals will be limited to keep the missile from attaining large angles of attack during the high q phase of flight. Guidance signals are terminated at Stage I cutoff and initiated again at Stage II ignition. Closed loop pitch and yaw steering are performed throughout Stage II flight. Stage II cutoff is initiated by a guidance signal. A vernier flight phase is employed after Stage II cutoff to correct the remaining position and velocity errors. Vernier thrust is terminated by the guidance system.

3. Performance

Accuracy of this system on typical space missions employing a 20 second vernier phase is estimated below:

Elevation Angle	Azimuth & Elev. Velocity Errors	Range Rate Error	Error in Thrust Vector Determination
12^{0}	$\pm$ 7 ft/sec	$\pm$ 1 ft/sec	2.5 mil
7^{0}	$\pm$ 9.8 ft/sec	$\pm$ 1.4 ft/sec	3.5 mil
3^{0}	$\pm$ 14 ft/sec	$\pm$ 2.5 ft/sec	6.8 mil

Position will be known with $\pm$ 300 ft. Guiding below a 3^{0} elevation angle is not recommended.

IV PERFORMANCE CAPABILITY

Efficient utilization of the Titan vehicles for space applications generally requires the addition of upper stages with restart capability. However, some missions are within the capability of the basic two stage vehicle and further improvements are possible by incorporation of small propulsion stages sized for apogee acceleration maneuvers.

Tables IV and V present the configuration data and the ground rules applicable to the performance summary curves. Some slight differences exist with the nominal configuration data quoted earlier in this document. Fluctuation of basic design estimates as well as variations in ground rules are responsible for these differences. Performance quoted is on a nominal basis.

Typical trajectory characteristics are shown for the 300 n.mi. circular orbit launched due east from AMR. A small solid rocket was used for acceleration to circular velocity at apogee of a transfer ellipse established by the two stage Titan vehicles.

TABLE IV
CONFIGURATION DATA USED FOR PERFORMANCE STUDIES

TITAN A
Powered Flight Phase

	First Stage	Separation Phase	Second Stage	Vernier Phase
Total Axial Thrust (lb)	299656 (s.l.)	9996	80,667	916
Effective Specific Impulse (sec)	249.214 (s.l.)	174.76	312.62	127.4
Total Weight Decay Rate (lb/sec)	1202.771	57.2	258.065	7.19
Total Usable Propellant (lb)	164,471	171.6	40,896	71.9
Jettison Weight (lb)	10,099	42	0	4980
Thrust Gradient (lb/psi)	-3024.8	---	---	---
Duration of Phase (sec)	136.74	3.0	158.47	10.0
Diameter (in)	120	95	95	95

TABLE V
GROUND RULES FOR PERFORMANCE STUDIES

The assumptions used in the calculation of the performance data presented are tabulated below:

1. Nominal engine performance in all stages

2. Nominal usable propellant contingency (outage) in all stages.

3. Spherical, rotating earth.

4. Constant flow-rate engines.

5. Linear variation of thrust with ambient pressure for first stage; thrust gradient neglected for subsequent powered phases.

6. Full impulse credit for start and shutdown propellant in Titan stages except for first stage start propellant.

7. Instantaneous thrust changes at start and shutdown of all powered phases.

8. ICAO Standard Atmosphere (1956) with extensions by Minzner; no wind.

9. Vertical flight for 20 seconds after lift-off.

10. Instantaneous tilt of the relative wind velocity vector and the total thrust vector at 20 seconds.

11. Zero - lift flight from 20 seconds to 140 seconds.

12. Constant inertial thrust attitude rate from 140 seconds to vernier burnout.

TABLE VI

TRAJECTORY AND PAYLOAD DATA
(EASTWARD AMR LAUNCH)

VEHICLE: TITAN A (SM-68)

MISSION: 300 n.mi. orbit

PAYLOAD: 3760 lbs.

Position Condition	Time (Sec)	Altitude (n.mi.)	Distance (n.mi.)	Dynamic Pressure (lb/ft^2)	(Inertial) Speed (ft/sec)	Acc (g's)	Tilt Angle (Referred to Local Vertical) (Degrees)	Impact Points (n.mi.)
1. Launch	0	0	0	0	1340.5	1.33	---	---
2. Stage 1 Burnout	136.74	24.3	44.9	121.1	9366.5	5.71	76.9	284
3. Stage 2 Ignition	139.74	25.3	48.7	93.9	9364.6	1.62	77.5	284
4. Stage 2 Burnout	298.22	50	407	1.3	26153	8.92	90.0	--
5. Vernier Burnout	308.22	50	447	1.3	26207	0.1	90.0	--

Remarks: Horizontal burnout at 50 n.mi.; payload on eccentric orbit (300 n.mi. apogee) is 3970 lbs. Circular orbit is obtained at this point through application of impulse by a hypothetical solid motor (specific impulse = 270 sec). As a result, the weight available for scientific payload is reduced to 3760 lbs.

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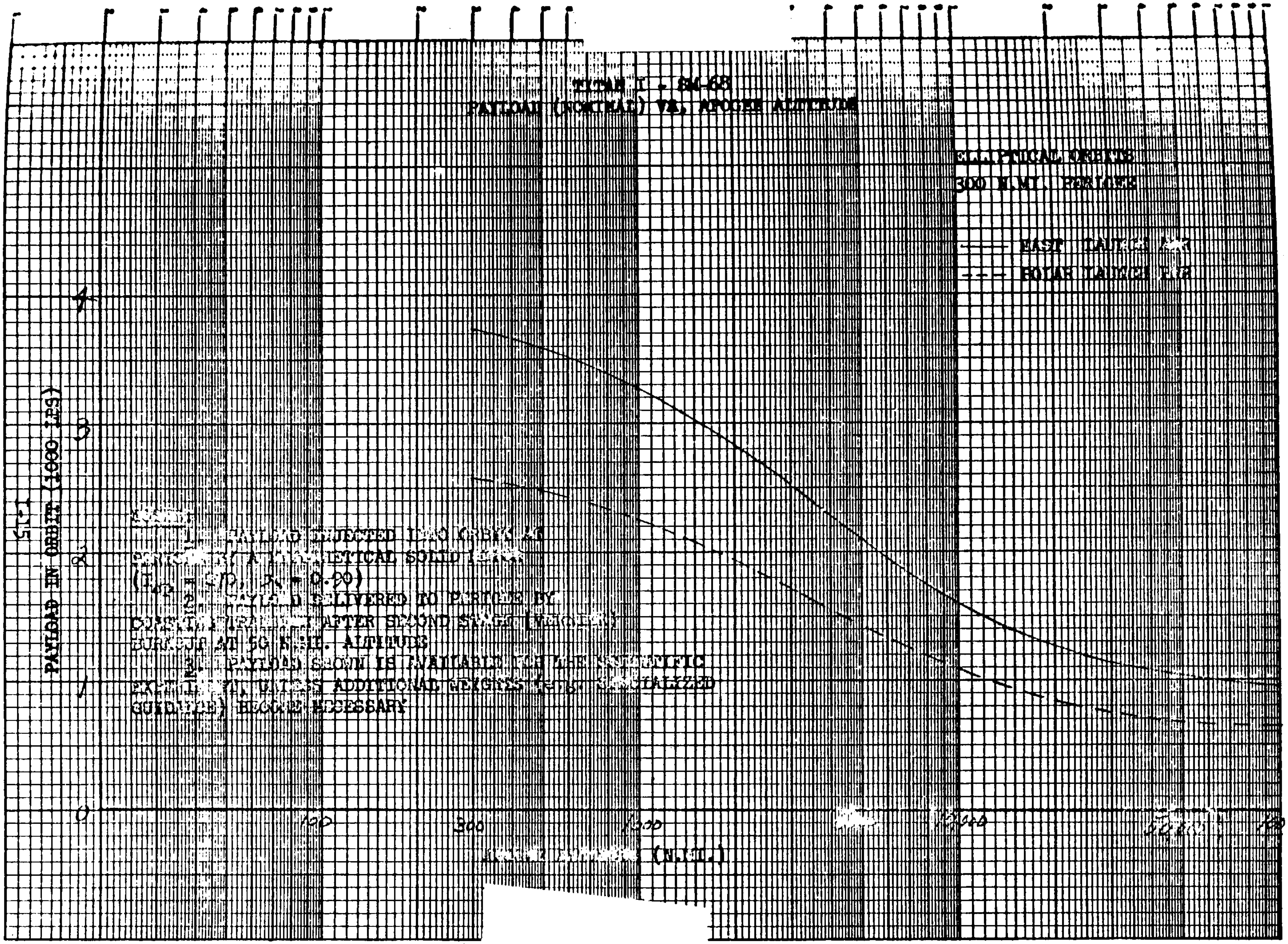
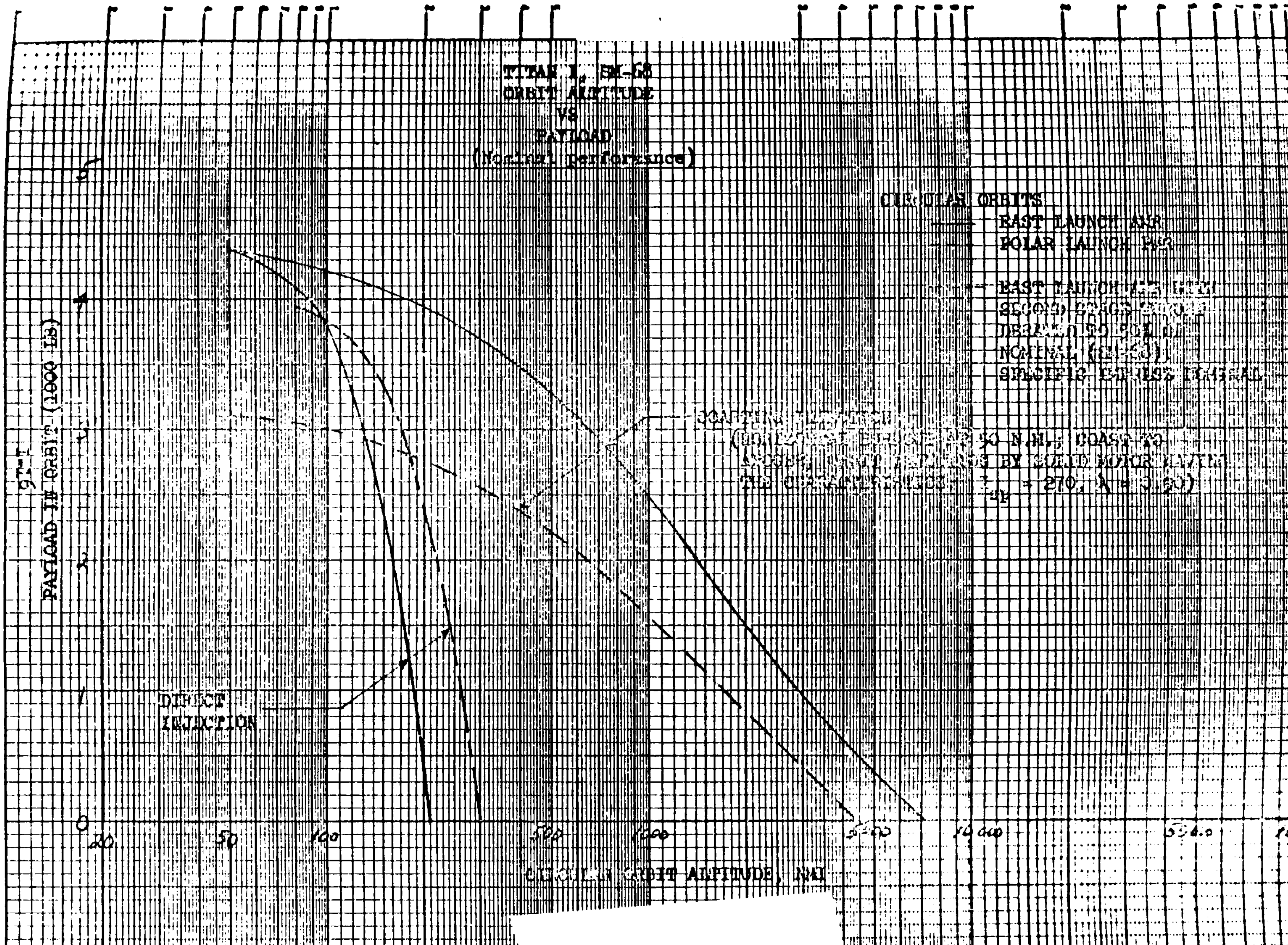
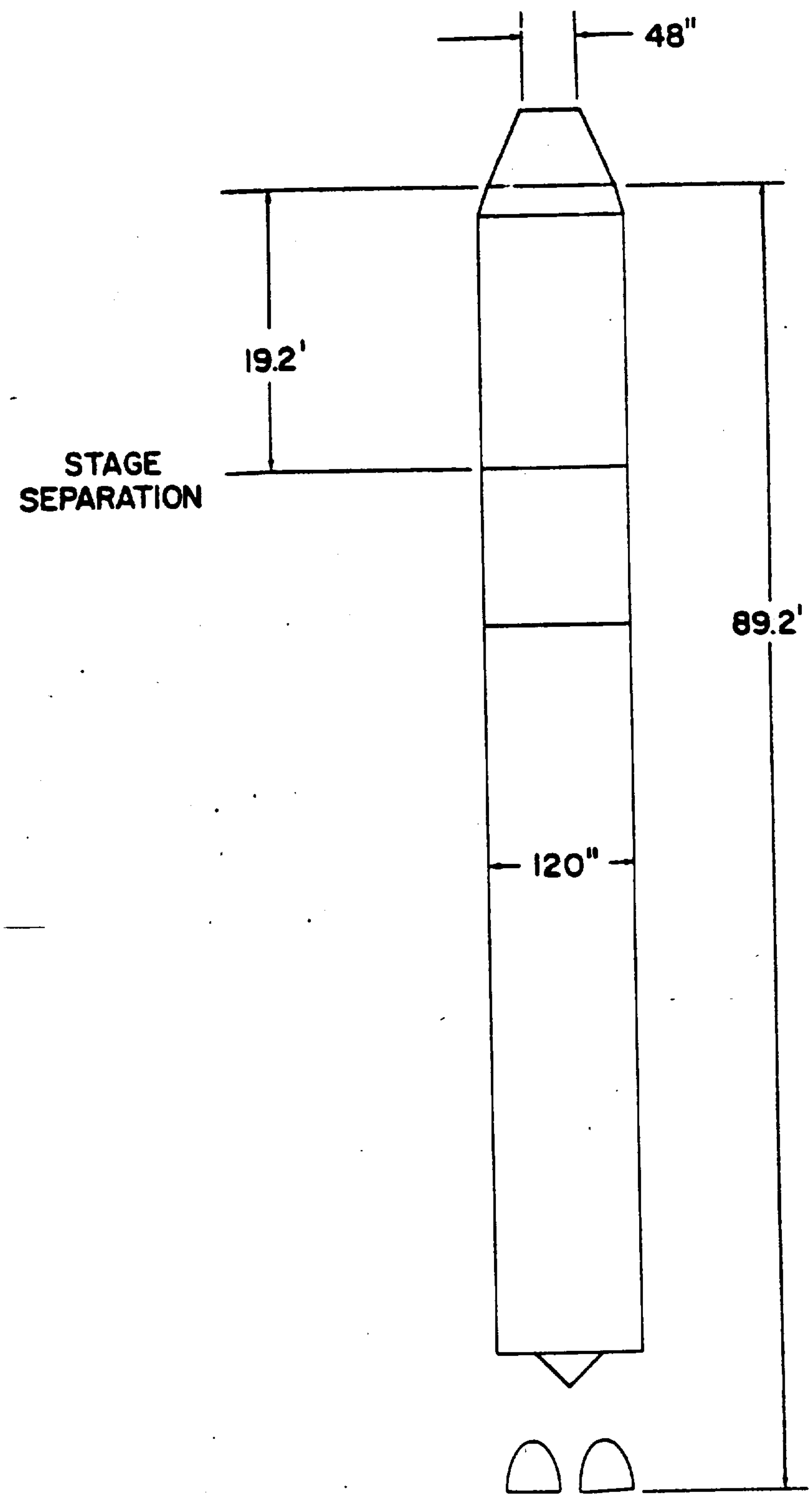
TITAN I - SM-68
PAYLOAD (NOMINAL) VS. APOGEE ALTITUDE



NOTES:
1. PAYLOAD INJECTED INTO ORBIT BY A HYPOTHETICAL SOLID STAGE. (I_{sp} = 270, λ = 0.90)
2. PAYLOAD DELIVERED TO PERIGEE BY ... AFTER SECOND STAGE ... BURNOUT AT 50 N.MI. ALTITUDE
3. PAYLOAD SHOWN IS AVAILABLE FOR THE SCIENTIFIC EXPERIMENT, UNLESS ADDITIONAL WEIGHTS (e.g. SPECIALIZED GUIDANCE) BECOME NECESSARY


TITAN I, SM-68
ORBIT ALTITUDE
VS
PAYLOAD
(Nominal performance)
CIRCULAR ORBITS
EAST LAUNCH AMR
POLAR LAUNCH PMR
EAST LAUNCH
NOMINAL
SPECIFIC
N.M., COAST TO
BY
= 270,
DIRECT
INJECTION
PAYLOAD IN ORBIT (1000 LB)
CIRCULAR ORBIT ALTITUDE, NM
0
1
2
3
4
5
20
50
100
500
1000
10 000



TITAN II

TABLE I
TITAN II or B, VEHICLE DESIGNATION SM-68B
STAGE CHARACTERISTICS

	First Stage	Separation Phase (Fire-in-hole)	Second Stage	Vernier Phase	Remarks
1. Stage Designation	SM-68B Stg I	---	SM-68B Stg II	---	
2. Light-off Weight (Less Payload)	320,480	63,351	63,252	5,350	Note 1
3. Loaded Stage Weight	256,516	63,030	62,931	5,029	Note 2
4. Propellant Weight (Usable)	246,846	99	57,902	132	
(a) Fuel Weight	84,249	---	20,679	---	
(b) Oxidizer Weight	162,597	---	37,223	---	
5. Propellant Weight (Residual)	1,387	0	312	20	
(a) Usable Contingency	284	0	63	---	
(b) Unusable	1103	0	249	20	
6. Stage Burnout Weight	9,670	62,931	5,029	4,897	
7. S.B.W. + Jettisonable Weight	10,283	62,931	5,029	5,218	Note 3
8. Stage Propellant Fraction	0.962		0.920		Note 4

NOTES:

1. Total weight of vehicle at lift-off and at ignition of succeeding powered phases less payload.

2. First stage loaded weight does not include 1343 lbs. start and ground losses, second stage start propellant contained in 99 lbs. consumed during separation phase.

3. Weights jettisoned are as follows:

Time (Ref. lift-off)	Event	Jettison Weight
149.26	First stage burnout	10,283
159.26	End separation phase	99
341.65	Stage II burnout	0
362.85	Vernier burnout	5,218

4. First stage: 246,846/256,516; Second Stage: 57,902/62,931

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TABLE II

TITAN B (SM-68B)

PROPULSION CHARACTERISTICS

Stage No.	1	Separation Phase	2	Vernier Phase
Stage Designation	Titan B		Titan B	
Stage Contractor	Martin		Martin	
Engine Designation	LR 87-AJ-5		LR 91-AJ-5	
Engine Contractor	AGC		AGC	
Propellants	N_2O_4/Aerozine*		N_2O_4/Aerozine*	
No. of Chambers	2	FIRE IN HOLE TECHNIQUE USED	1	2
Thrust/Chamber (lb)				
S.L.S.	215,000		---	
Vac.	237,500		100,000	865
Specific Impulse (sec)				
S.L.S.	260		---	
Vac.	284		315	278
Nozzle Area Ratio	8		45	30
Chamber Pressure	806**		845**	
Nozzle Exit Area	1462		2943	
Relite Capability	None		None	
Propulsion Feed System	Pump Fed		Pump Fed	Solid Motors
Total Axial Thrust (lb)				
S.L.S.	429,734		---	
Vac.	474,706		100,000	
Total Weight Decay Rate (lb/sec)	1653.85		317.46	

* Aerozine consists of 50% hydrazine, 50% UDMH
** At injector face

TABLE III

TOLERANCES SM-68B

Parameter	First Stage	Second Stage
Thrust	± 3.0% (Sea Level)	± 3.0% (Vacuum)
Specific Impulse	± 1.0% (Sea Level)	± 1.0% (Vacuum)
Propellant Loading	± 0.5%	± 0.5%
Propellant Utilization	Note (1)	Note (2)
Drag Coefficient	± 10%	± 10%
Dry Weight	± 1.5%	± 1.5%
Wind and Guidance	± 100 ft/sec (3)	

(1) Nominal Outage = 0.115% (Fuel Bias), Max Outage = 1%

(2) Nominal Outage = 0.109 (Fuel Bias), Max Outage = 1%

(3) Combined Effect of Wind and Guidance on Burnout Velocity

II TRAJECTORY CONSTRAINTS FOR TITAN B

(a) Second Stage Separation

Titan B employs the "fire-in-the-hole" staging technique. Stage II engine shutdown and bolt detonations are simultaneous and the stages are clear of one another about one second later. The permissible product of $q\alpha$ for the nominal trajectory is 150 lb-deg/ft^2. Most of the second stage separation constraints for Titan A apply to Titan B as well. In particular, no period of negative g can be tolerated in the present design. The Titan B sustainer engine can be modified for restart capability. Ullage rockets would then be necessary prior to the second engine burning period.

(b) Aerodynamic Heating

Acceptable nominal trajectories for the Titan ICBM's are limited by aerodynamic heating to a value of approximately 100×10^6 for the quantity $\int_0^t qV_{rw}\,dt$ where V_{rw} = relative wind velocity. Detailed analysis is necessary to determine the effects of trajectory dispersions and atmospheric variations on the maximum skin temperatures.

Forward transition sections are most subject to aero heating. In general, these transitions would be replaced with special adapters for space applications. Tolerance to heating is thereby increased if the adapters are appropriately designed. Further increases to allowable heating are possible by addition of insulation or ablative finishes to critical areas.

(c) Structural Limitations

The comments of Section II c on Titan A apply to Titan B with the one exception that the acceptable $(q\alpha)_{max}$ for the ICBM nominal trajectory is about 8500 lb-deg/ft^2 for the Titan B flown through the 1959 Sissenwine wind profile.

(d) The Titan B ICBM employs an all inertial guidance system and consequently does not impose trajectory restraints.

III GUIDANCE SYSTEM DESCRIPTION-FOR TITAN B

1. Description

Titan B contains an A.C. Spark Plug inertial guidance system weighing 238 lbs. The inertial measurement unit is made up of a gyro stabilized platform, integrating accelerometers and associated electronics. It is housed in a package 19" in diameter by 27" long. The computer contains the circuitry and storage required to solve the guidance equations and process gimbal and accelerometer outputs into steering commands. The digital computer is housed in a package 20" x 22" x 12". Both pieces of equipment are located between tanks on Stage II.

Characteristics of the system are listed below:

a. Gyro Characteristics — 2 FBG Type

Random drift	± .05 deg/hr
Constant drift (reaction torques)	± .05 deg/hr
Acceleration sensitive drifts	± .1 deg/hr/g
Acceleration^2 sensitive drifts	± .05 deg/hr/g^2
Maximum torquing rates	± .1 deg/sec

b. Accelerometer Characteristics PIGA 25 Type

Range	13 g
Bias	+ 6 (10^{-5}) g
Scale Factor	.14 ft/sec/pulse - digital output
Non linearity	+ .006%
Alignment Accuracy	± .003 radian before compensation .18 (10^{-4}) radian after compensatio
Cross Coupling	< .002 radian/g
Threshold	2 (10^{-5}) g
Dynamic lag	.2 ft/sec/g

c. Gimbal Characteristics

Freedom of Motion (on pad axis)	Range
Roll	± 130 deg
Pitch	- 130 to +60
Yaw	± 20 deg

d. Platform

Alignment Accuracy	Azimuth ± 10 sec of arc Vertical ± 4sec of arc

e. Computer. General Purpose, Magnetic Drum, Welded Module, Digital.

1. Magnetic Drum Characteristics:

70 tracks; 64 words/track; 27 bit/word; 4 clock pulses/bit

In the present application these tracks are divided as below:

30 tracks for instructions
15 tracks for constants storage
2 time tracks
2 intermediate data tracks
5 resolvers
16 spares

The drum operates at 6000 rpm.

2. Computer input-digital velocity, time, and gimbal angles. Loaded by punched tape.

3. Computer output - Pitch, yaw and roll steering signals are issued as digital steps 20 times/sec. 32 steps of missile attitude error between 0 and 5.45 degrees can be issued to the control system. Twelve discrete commands are available.

4. Calculating Rate - One complete trajectory calculation every 0.5 sec. Velocity to be gained is computed 20 times/sec during Stage II flight and 40 times/sec during vernier flight.

2. System Operation

The platform is aligned on the pad to the azimuth and vertical reference. Shortly after liftoff the inertial system rolls the missile at rates up to 12 deg/sec to the required flight azimuth. Maximum roll azimuth is ± 115 degrees. After the roll program is completed, a pitch program is initiated. Open loop steering then continues until after maximum q. During this period the computer is continuously calculating velocity to be gained (Vg) from accelerometer inputs. When dynamic pressures drop to a sufficiently low level, closed loop Vg steering is initiated and continues until vernier cutoff. When velocity to be gained approaches a small value, the guidance system orders sustainer cutoff. Both pitch and yaw steering are performed during the short vernier period. When Vg reaches zero, the guidance system cuts off the vernier engine.

3. Performance

Accuracy of this system in a typical space booster application with a 10 sec vernier period is given below:

Flight Path velocity error - 1 ft/sec
Flight Path angle error - 1 mil
Cross Range velocity error - 2 ft/sec
Position error - 200 ft

The inertial guidance system has no elevation angle limitations.

IV PERFORMANCE CAPABILITY

(See Section IV on Titan A)

TABLE IV
CONFIGURATION DATA USED FOR PERFORMANCE STUDIES

TITAN B
Powered Flight Phase

	First Stage	Separation Stage	Second Stage	Vernier Phase
Total Axial Thrust (lb)	430,000 (s.l.)	0	100,000	1550
Effective Specific Impulse (sec)	260 (s.l.)	0	315	220
Total Weight Decay Rate (lb/sec)	1653.846	0	317.46	7.045
Total Usable Propellant (lb)	246846	0	57,902	155
Jettison Weight (lb)	10,500	92	0	5205
Thrust Gradient (lb/psi)	-3056.46	---	---	---
Duration of Phase (sec)	149.26	1.0	182.39	22.0
Diameter (in)	120	120	120	120

TABLE V
GROUND RULES FOR PERFORMANCE STUDIES

(SEE TABLE V FOR TITAN A)

TABLE VI

TRAJECTORY AND PAYLOAD DATA
(EASTWARD AMR LAUNCH)

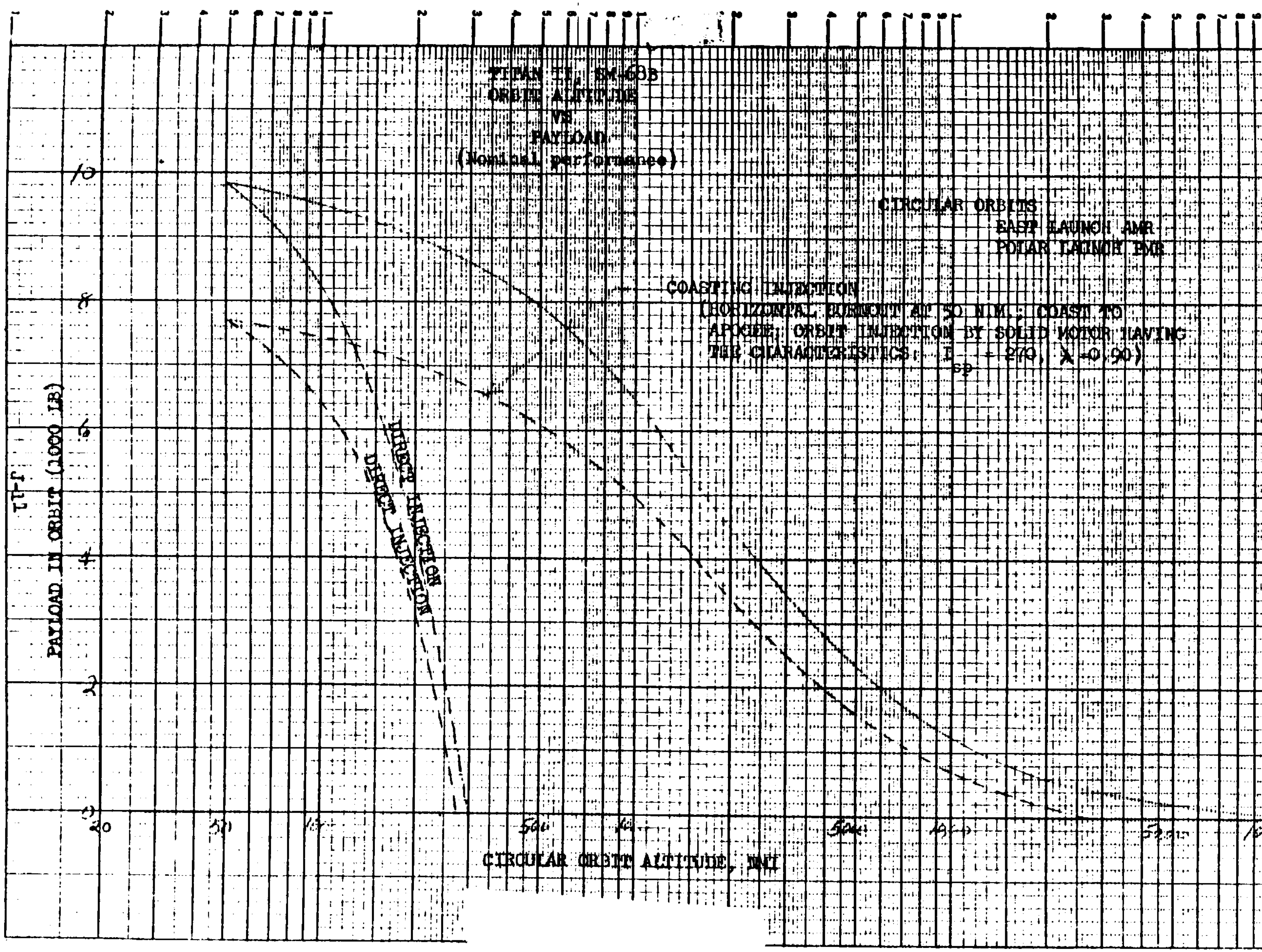
VEHICLE: TITAN B (XSM-68B)

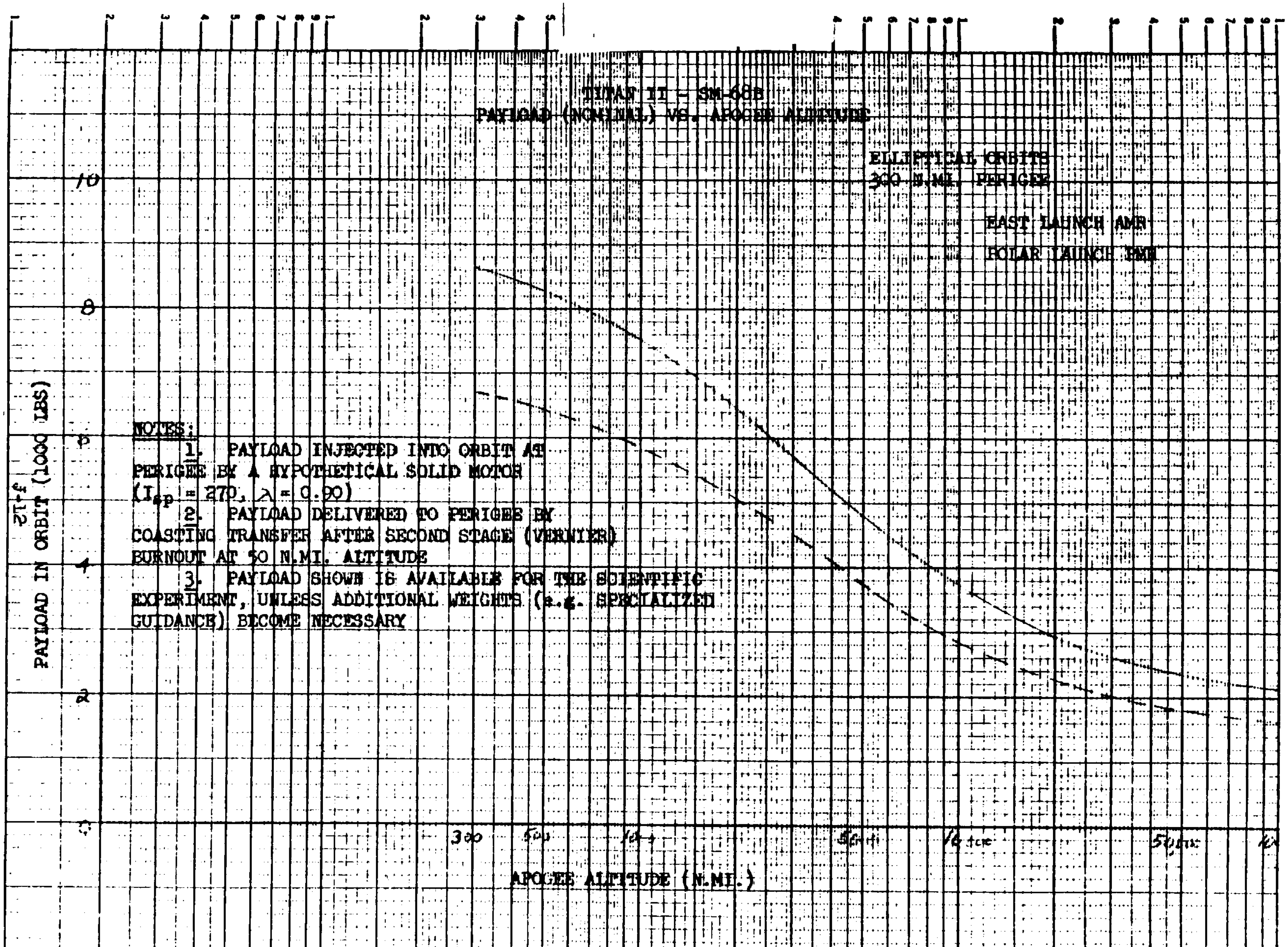
MISSION: 300 n.mi. orbit

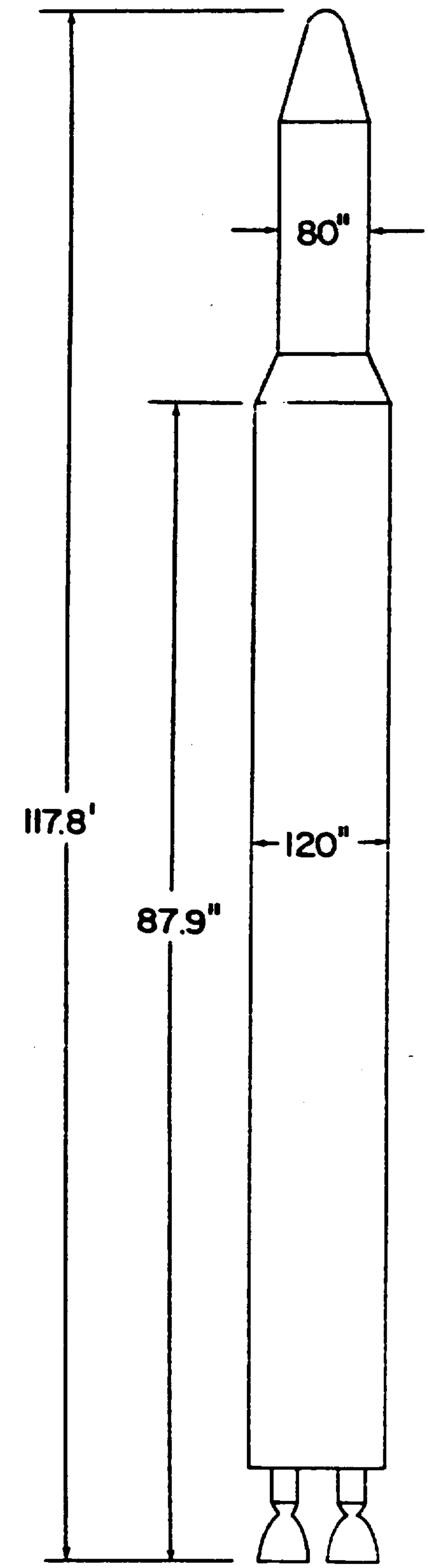
PAYLOAD: 8650

Position Condition	Time (Sec)	Altitude (n.m.)	Distance (n.m.)	Dynamic Pressure (lb/ft^2)	(Inertial) Speed (ft/sec)	Acc (g's)	Tilt Angle (Referred to Local Vertical) (Degrees)	Impact Points (n.m.)
1. Launch	0	0	0	0	1340.05	1.30	---	---
2. Stage 1 Burnout	149.26	26.8	54.3	82.5	10164.5	5.72	80.2	327
3. Stage 2 Burnout	332.65	50	484.8	1.3	26112.2	6.92	89.8	---
4. Vernier Burnout	354.65	50	573.3	1.4	26207	0.1	90.0	---

Remarks: Horizontal burnout at 50 n.mi.; payload on eccentric orbit (300 n.mi. apogee) is 9135 lbs. Circular orbit is obtained at this point through application of impulse by a hypothetical solid motor (specific impulse = 270 sec.) As a result, the weight available for scientific payload is reduced to 8650 lbs.)


TITAN II, SM-68B
ORBIT ALTITUDE
VS
PAYLOAD
(Nominal performance)
CIRCULAR ORBITS
EAST LAUNCH AMR
POLAR LAUNCH PMR
COASTING INJECTION
(HORIZONTAL BURNOUT AT 50 N.M.; COAST TO
APOGEE; ORBIT INJECTION BY SOLID MOTOR HAVING
THE CHARACTERISTICS: I_{sp} = 270, λ = 0.90)
DIRECT INJECTION
DIRECT INJECTION
PAYLOAD IN ORBIT (1000 LB)
CIRCULAR ORBIT ALTITUDE, NMI


TITAN II - SM-68B
PAYLOAD (NOMINAL) VS. APOGEE ALTITUDE
ELLIPTICAL ORBITS
300 N.MI. PERIGEE
EAST LAUNCH AMR
POLAR LAUNCH PMR
NOTES:
1. PAYLOAD INJECTED INTO ORBIT AT PERIGEE BY A HYPOTHETICAL SOLID MOTOR (I_{sp} = 270, λ = 0.90)
2. PAYLOAD DELIVERED TO PERIGEE BY COASTING TRANSFER AFTER SECOND STAGE (VERNIER) BURNOUT AT 50 N.MI. ALTITUDE
3. PAYLOAD SHOWN IS AVAILABLE FOR THE SCIENTIFIC EXPERIMENT, UNLESS ADDITIONAL WEIGHTS (e.g. SPECIALIZED GUIDANCE) BECOME NECESSARY
PAYLOAD IN ORBIT (1000 LBS)
10
8
6
4
2
0
300
500
1000
5000
10,000
50,000
APOGEE ALTITUDE (N.MI.)



TITAN II – AGENA B MOD I

TITAN SM68B/Agena B-Mod I

This vehicle has been studied by Lockheed Aircraft Company. Though not currently under development, performance estimates are presented.

Trajectory and Payload Data

The Titan ascent trajectory used is similar to that of the Atlas until first stage separation. However, after ignition of the second stage, the trajectory consists of a phase of constant attitude rate in inertial space rather than a consta attitude path in local space. This continues throughout the remainder of the boost and Agena phase of powered flight, again resulting in the velocity vector aligned with the local horizontal at the end of the Agena first burn.

Although the constant attitude rate trajectory is incompatible with the proposed guidance concept of the Agena B-Mod I (constant pitch angle), it was concluded that the differences in payloads due to trajectory simulation are negligible and therefore did not justify the expenditure of recalculating performance.

No specific trajectory information is presently available with the Agena B-Mod I. The following table lists the -3σ payloads obtainable for typical missions requested.

MISSION	PAYLOAD
300 NM AMR	8895 lb.
Escape AMR	2315 lb.
24 hr. synch. equat.	1095 lb.
Soft lunar landing	150 to 250 lb.

Typical lunar soft landing trajectories can vary in trip time from about 40 to 90 hours. They vary in departure velocity by several hundred ft/sec, and the retro-velocity prior to touchdown varies from about 7,600 to 9,000 ft/sec.

The trajectory chosen consists of departure from a parking orbit with 100 ft/sec less than escape velocity. After a trip duration of 66 hours, 8,600 ft/sec (includi 100 ft/sec assumed for midcourse correction) retro-velocity is required for a direct impact soft landing with a relative velocity of less than 50 ft/sec.

The total weight at departure from the parking orbit equals 3,770 lb. If no weight is dropped, then the total weight at the lunar surface equals 1,605 lb, of which a maximum of 150 lb. can be considered instrument payload. If, however, some weight can be jettisoned prior to the application of the 8,600 ft/sec retro-velocity, a considerably higher payload may result. As an example, dropping 945 lb. consisting of structures, propulsion system, tanks and trapped propellants, and assuming a separate engine and tanks weighing 350 lbs., the instrument payload on the moon will be about 250 lbs. A detailed mission analysis is necessary to determine what the optimum conditions for achieving maximum payloads will be.

Titan SM68B/Agena B Mod-I Guidance System

The A.C. Spark Plug inertial guidance system is used for Titan SM68B.

A summary of the inertial system operation follows. The platform is aligned prior to liftoff. As the missile moves along the trajectory, velocity information is gathered by three digital integrating accelerometer which are mounted to the stabilized platform. The three components of velocity along with gimbal angle displacements are fed into the airborne digital computer where they are transformed into the computational coordinate system. The computer then takes this information and calculates the velocity-to-be-gained (Vg) from which steering and cutoff commands are generated. Steering signals enter the autopilot as positive or negative d-c voltages. The amplitude of the output signal is dependent on the missile attitude displacement error. The scale factor is one volt output per degree missile attitude error. Steering signals deflect the engines which cause the missile to rotate about its center of gravity and null out the attitude error signal.

For a more complete description, see the Titan B section.

The Agena B Mod-I guidance equipment is essentially the same as utilized in the Agena B. The guidance system provides the basic vehicle attitude reference during all phases of Agena ascent, and the event timing and measurement and control of the incremental velocity additions required during Agena powered flight phases. The Titan SM68B inertial guidance establishes the required Agena attitude, time, and trajectory initial references, with the Agena guidance subsystem constraining the trajectory as required following separation from Titan SM 68B.

Agena B Mod I Structural Limitations

The Agena B Mod-I vehicle may be made as strong as necessary for general requirements. In all cases the strength of the Agena B Mod-I will be such that the booster strength will be critical before the Agena structure becomes critical. This may be considered to be the design criteria on Atlas boosted missions. For other boosters such as Titan B, this criteria may have to be modified for the sake of payload capability.

The Agena B Mod-I is proposed to be capable of 60,000 lb-g's of payload. Payload here refers to everything forward of the Agena tanks. The Atlas booster is presently limited to 140,000 lb-g's for booster burnout.

The present design limits for the SM68B are $q = 710$ lb/ft^2 and $\alpha = 11.8$ degrees. Analysis with Agena weights up to 27,000 lbs. indicate that the Stage II between-tanks section is critical when aerodynamic heating is also considered. This section must also be strengthened to satisfy Stage burnout condition loads. Modification to the Stage I fuel tank structure and the Stage II forward skirt structure is also indicated.

TABLE I

TITAN SM68B/AGENA B-MOD I

STAGE CHARACTERISTICS	1	2	3
1. Stage Designation	Titan Stage I	Titan Stage II	Agena B-Mod I
2. Light-off Weight (less payload) lb.(a)	338,749	N/A	N/A
2.' Separation Weight (less payload), lb.	N/A	81,839	17,465
Jettisoned Weight	N/A	99 6)	26(12)
3. Loaded Stage Weight (=ignition weight for Agena) (b)	257,009	63,790	17,439
4. Propellant Weight Usable(1) (lb)	243,598	57,349 7)	15,920(13)
a. Fuel			
b. Oxidizer			
5. Propellant Weight Residual, lb.	3,631	840	211
a. Usable contingency(2)	2,014	478	80(14)
b. Unusable contingency	1,517 3)	362 8)	131(15)
5.' Non-Impulse Expendables, lb.	300	137	
a. Propellants	0	0	68
b. Other	300 4)	137 9)	19(16)
6. Stage Burnout Weight (SBW), lb.(c)	11,097 5)	5,727 10)	1,352(17)
7. SBW + Jettisoned Weight, lb. (d)	11,097	6,212 11)	N/A
8. Stage Propellant Fraction, K_p $\frac{\text{Usable propellant wt}}{\text{Loaded stage wt}}$	243,598/257,009=.947	59,349/63,790=93	-----

Footnotes:

(a) (b) (c) (d) (See Thor DM 21/Agena B footnote section)

1. Does not include item 5a.

2. Not included in item 6. It is assumed used in -3σ case.

3. Includes the following items:

1,073 lb	trapped propellant
444 lb	fuel bias for mixture ratio shift (=.18% of items 4 plus 5a)
1,517 lb	

4. Includes the following items:

282 lb	pressure gas
18 lb	lubricant
300 lb	

5. Includes the following items:

5,288 lb	structure
3,700 lb	propulsion
219 lb	guidance and control
114 lb	electrical system
217 lb	telemetering and range safety
42 lb	miscellaneous
1,517 lb	see footnote 3.
11,097 lb	

6. Non-impulse propellant used during separation phase

7. Includes the following items:

57,154 lb	usable propellant stage 2
155 lb	vernier propellant (solid)
40 lb	retro-thrust propellant (solid)
57,349 lb	

8. Includes the following items:

264 lb	trapped propellant
98 lb	fuel bias for mixture ratio shift (=17% of items 4 plus 5a
362 lb	

9. Includes the following items:

129 lb	pressure gas
8 lb	lubricant
137 lb	

Footnotes (Contd)

10. Includes the following items:

2,435 lb	structure
1,505 lb	propulsion
786 lb	guidance and control
231 lb	electrical system
331 lb	telemetering and range safety
77 lb	miscellaneous
262 lb	see footnote 8.
5,727 lb	

11. Includes the following items:

5,727 lb	as above
470 lb	adapter and attachments
15 lb	destruct system
6,212 lb	

12. Includes the following items:

8 lbs	horizon sensor fairings
12 lbs	ullage orientation propellant
3 lbs	attitude control propellant
3 lbs	pre-flow 1st burn
26 lbs	

13. Does not include item 5a.

14. Not included in item 6. It is assumed used in -3σ case

15. Consists of 109.5 lb trapped propellant & allowance for mixture ratio shift plus 21.5 lb expended after 3rd burn

16. Includes the following items:

12 lbs	ullage orientation propellant 2nd burn
7 lbs	attitude control propellant
19 lbs	

17. Includes the following items:

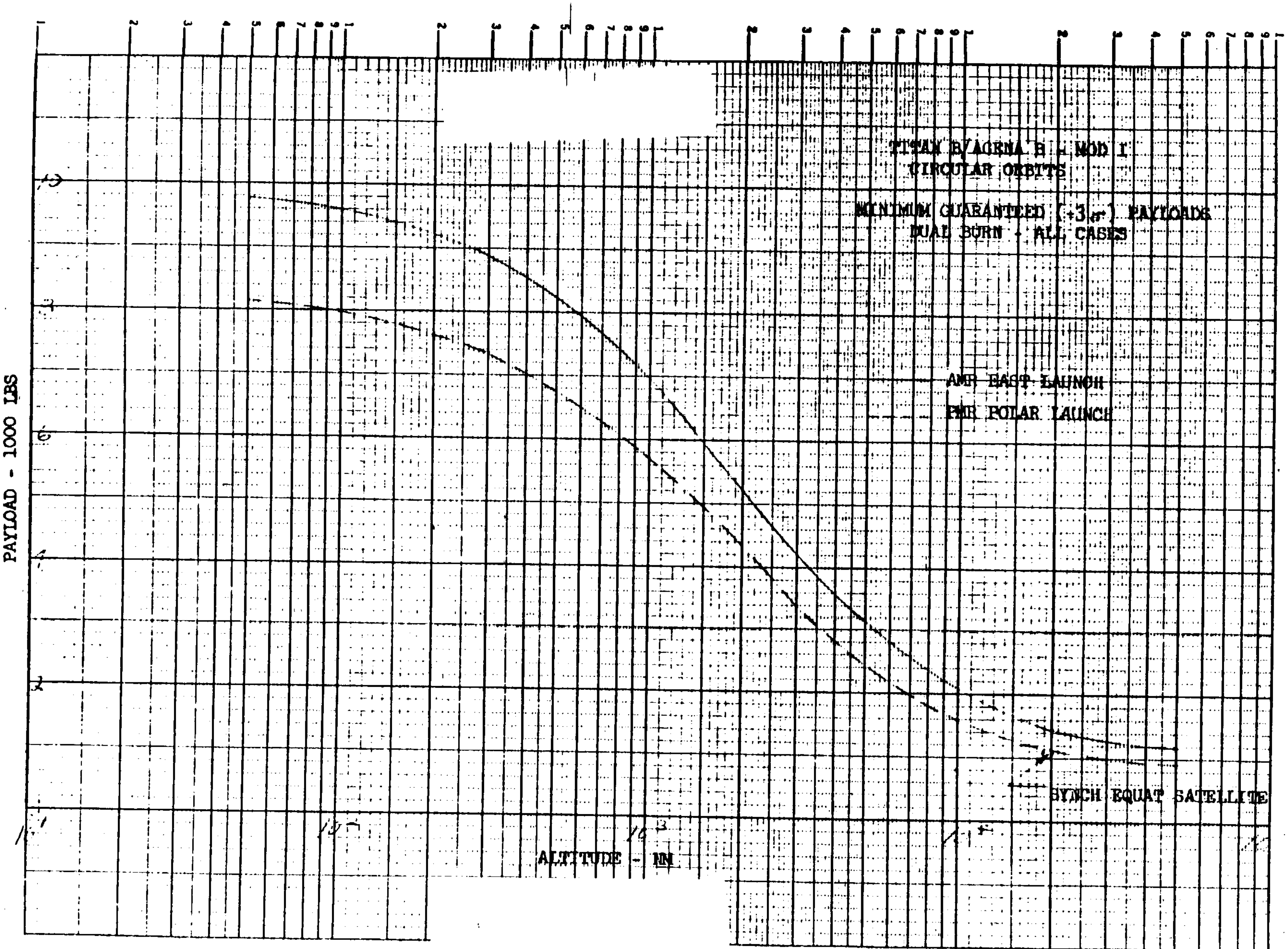
429 lbs	structure (including tanks)
412 lbs	propulsion
81 lbs	controls
112 lbs	guidance
138 lbs	APU
49 lbs	communications
131 lbs	unusable contingency (item 5b)
1352 lbs	

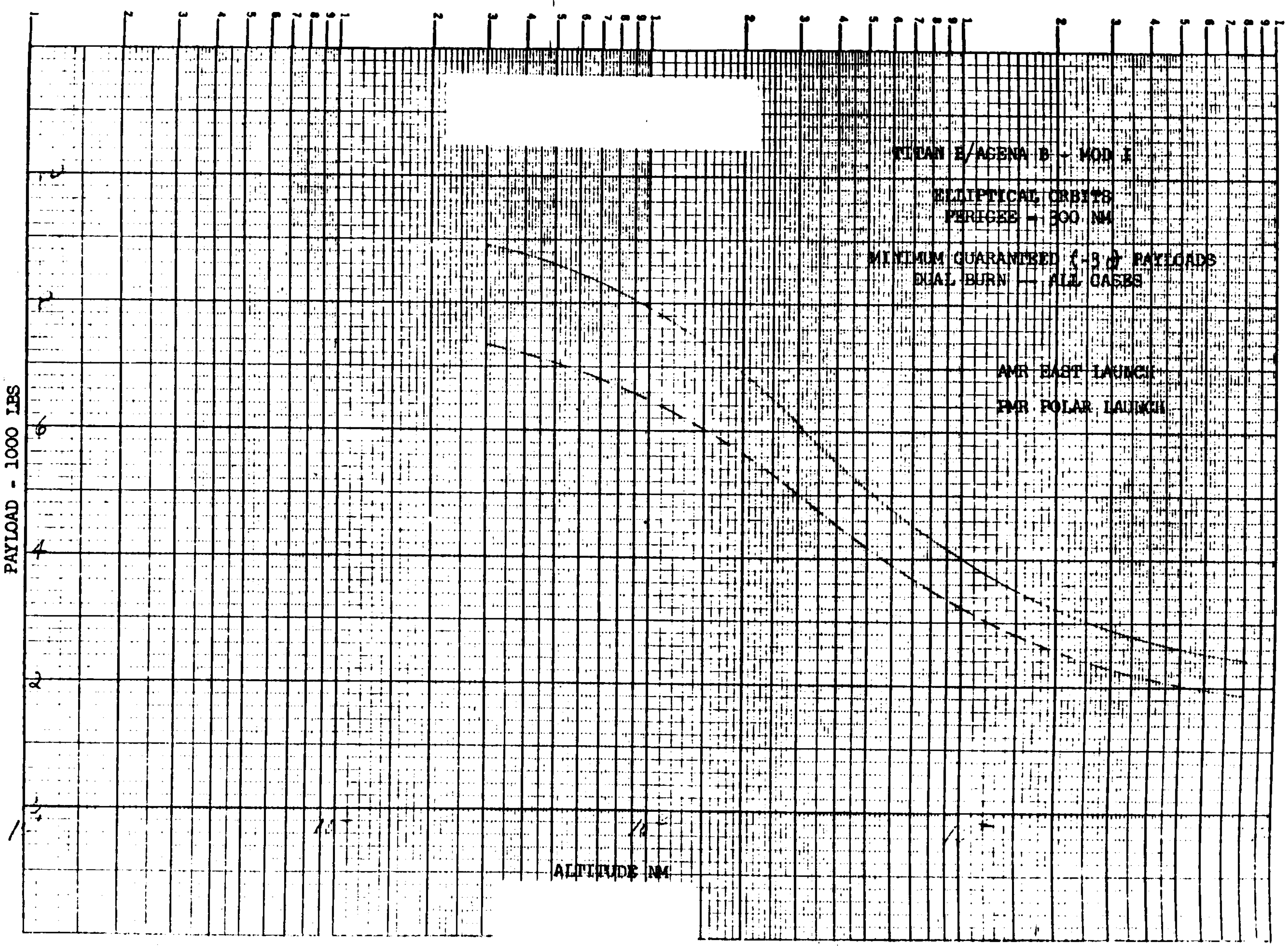
TABLE II

TITAN SM 68B/AGENA B-MOD I

STAGE PROPULSION CHARACTERISTICS

	1 Titan Stage I	2 Titan Stage II	3 Vernier	4 Agena B-Mod I
1. Engine designation	XLR 87-AJ-5	XLR 91-AJ-5		BAC Model D8133
2. Contractor	Aerojet Gen. Co.	Aerojet Gen. Co.		Bell Aerosystems Co.
3. Propellants	N_2O_4,50/50 UDMH/HZN	N_2O_4,50/50 UDMH/HZN		N_2O_4,50/50 UDMH/HZN
4. No. of chambers	2	1	2	1
5. Thrust (SLS) per chamber, lb	215,000	N/A	N/A	N/A
6. Thrust (vacuum) per chamber, lb	237,500	100,000	775	16,166
7. I_{sp} (SLS, -3σ), sec	257	N/A	N/A	N/A
8. I_{sp} (vacuum, -3σ), sec	281	312	220	313 (3σ)
9. Nozzle expansion ratio	8:1	45:1		45:1
10. Chamber pressure , psia	806	845		500
11. Nozzle exit area, A_c , in^2				771
12. Relite. capabilities	None	None	None	Unlimited
13. Propellant feed system	Turbopump	Turbopump	Solid	Turbopump


TITAN B/AGENA B - MOD I
CIRCULAR ORBITS
MINIMUM GUARANTEED (-3σ) PAYLOADS
DUAL BURN - ALL CASES
AMR EAST LAUNCH
PMR POLAR LAUNCH
SYNCH EQUAT SATELLITE
ALTITUDE - NM
PAYLOAD - 1000 LBS


TITAN E/AGENA B - MOD I
ELLIPTICAL ORBITS
PERIGEE - 300 NM
MINIMUM GUARANTEED (-3 σ) PAYLOADS
DUAL BURN - ALL CASES
AMR EAST LAUNCH
PMR POLAR LAUNCH
ALTITUDE NM
PAYLOAD - 1000 LBS

TITAN - CENTAUR

No Titan-Centaur vehicle is currently under development. However, a study of the payload capabilities of vehicles using Titan A and B as the first two stages and the Centaur stage currently under development as the third stage has been made and will be presented in this section.

The ground rules used in the trajectory calculations are outlined in Table I. Tables II and III present the configuration data used. A detailed description of the Centaur stage can be found in the Atlas-Centaur section. Table IV summarizes nominal performance capabilities for a variety of missions.

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TABLE I

GROUND RULES FOR PERFORMANCE STUDIES

1. Nominal engine performance and outage in all stages

2. Spherical, rotating earth

3. ICAO Standard Atmosphere (1956) with extensions by Minzner, no wind

4. Constant flow-rate engines

5. Linear variation of thrust with ambient pressure for first stage; thrust gradient neglected for subsequent powered phases

6. Full impulse credit for start and shutdown propellant in Titan stages except for first stage start propellant

7. Instantaneous thrust changes at start and shutdown of all powered phases

8. Vertical flight for 20 seconds after liftoff

9. Instantaneous tilt of the relative wind velocity vector and the total thrust vector at 20 seconds

10. Zero-lift flight from 20 seconds to 140 seconds

11. Constant inertial thrust attitude rate from 140 seconds to end of first Centaur powered phase

12. First Centaur burnout occurs at 50 n mi altitude with inertial velocity vector in the local horizontal

13. Guidance and solo vernier systems were not carried on the Titan stages

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TABLE II

CONFIGURATION DATA FOR PERFORMANCE STUDIES

TITAN A - CENTAUR

NOMINAL CONDITIONS

	FIRST STAGE	SEPARATION PHASE	SECOND STAGE	THIRD STAGE
Thrust (Lb)	299,656 (Sea level)	9996	80,677 (Vacuum)	30,000 (Vacuum)
Specific Impulse (Sec)	249.214 (Sea level)	174.76	312.62 (Vacuum)	419 (Vacuum)
Weight Decay Rate (Lb/Sec)	1202.771 (1)	57.2	258.065	71.6
Total Useable Propellant (Lb)	164,421 (2)	171.6	40,920	27,486
Jettison Weight (Lb)	9873 (3)	42	4931	3574
Diameter (In)	120	95	95	120
Thrust Gradient (Lb/Psi)	-3024.8	0	0	0
Duration of Phase (Sec)	136.7	3	158.6	383.9

(1) Includes Lox-bleed rate of 0.366 lb/sec from second stage tanks through APDA unit

(2) Includes APDA bleed

(3) Includes 128.4 lb of second stage propellant consumed by verniers during first stage operation and dropped with no credit for impulse

TABLE III

CONFIGURATION DATA FOR PERFORMANCE STUDIES

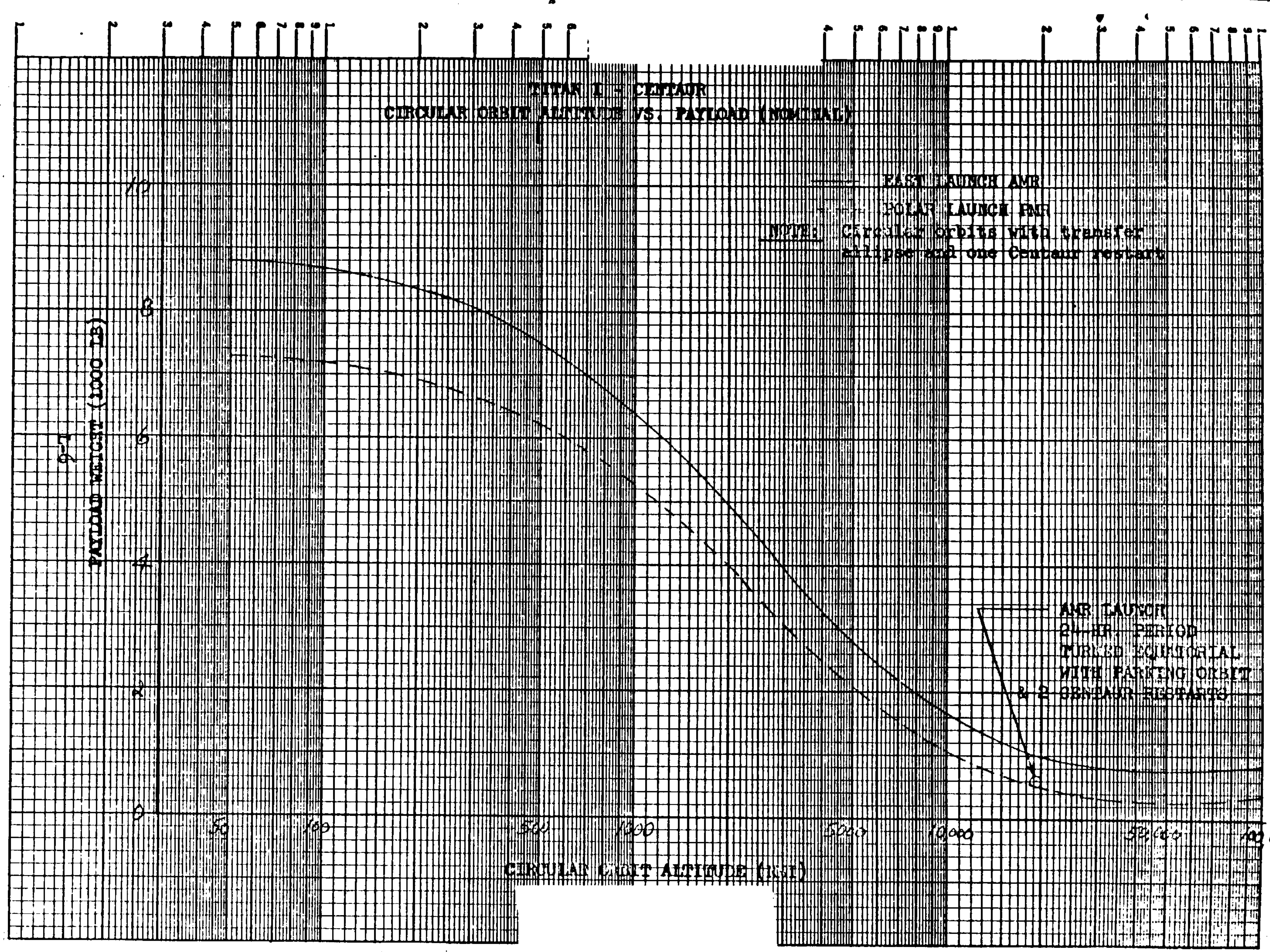
TITAN B - CENTAUR

NOMINAL CONDITIONS

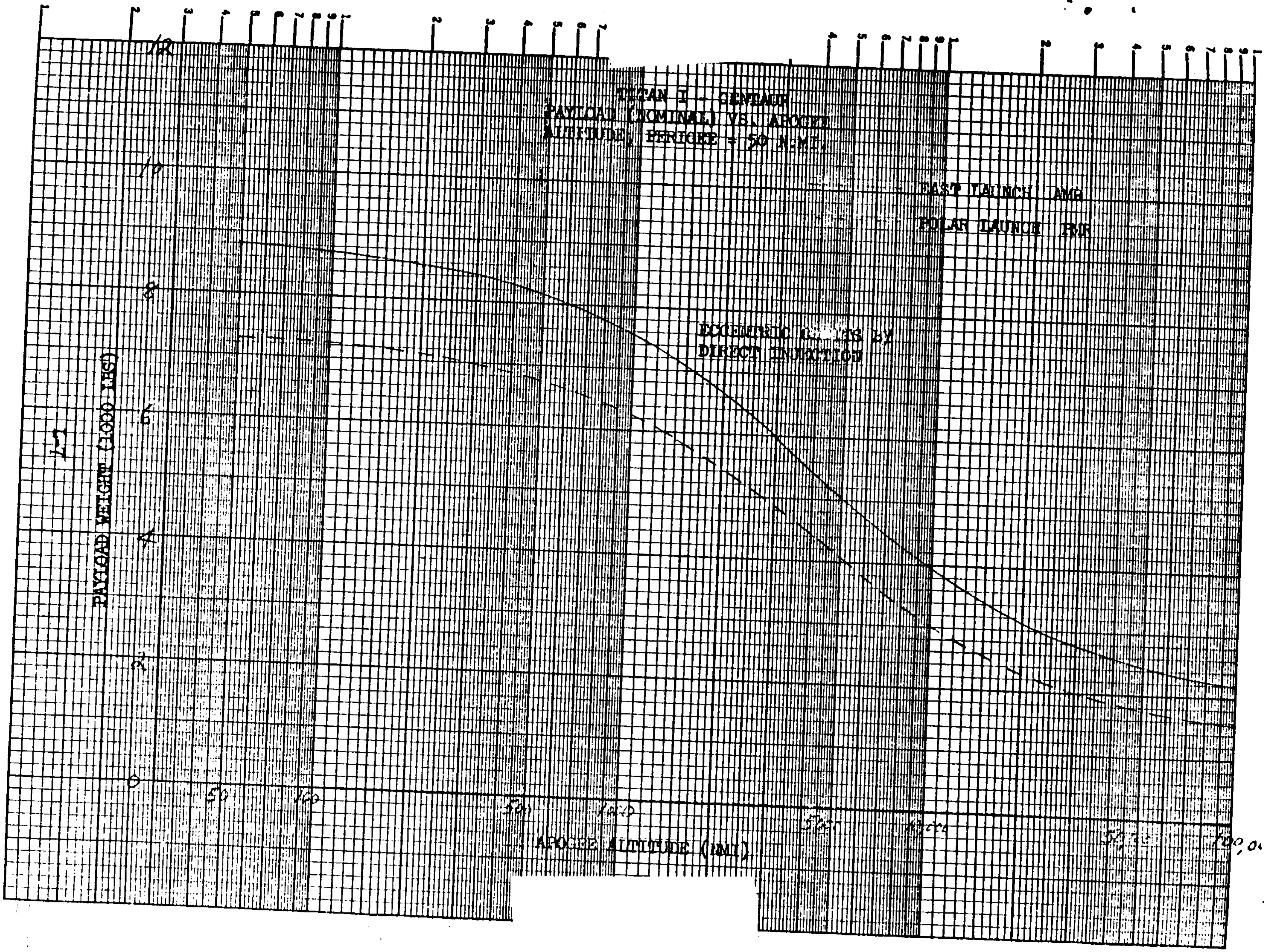
	FIRST STAGE	SEPARATION PHASE	SECOND STAGE	THIRD STAGE
Thrust (Lb)	430,000 (Sea Level)	0	100,000 (vacuum)	30,000 (Vacuum)
Specific Impulse (Sec)	260 (Sea Level)	0	315 (vacuum)	419 (Vacuum)
Weight Decay Rate (Lb/Sec)	1653.846	0	317.46	71.6
Total Useable Propellant (Lb)	246,846	0	57,902	27,486
Jettison Weight (Lb)	10,608	99	5,791	3,574
Diameter (In)	120	120	120	120
Thrust Gradient (Lb/Psi)	-3056.464	0	0	0
Duration of Phase (Sec)	149.3	1	182.4	383.9

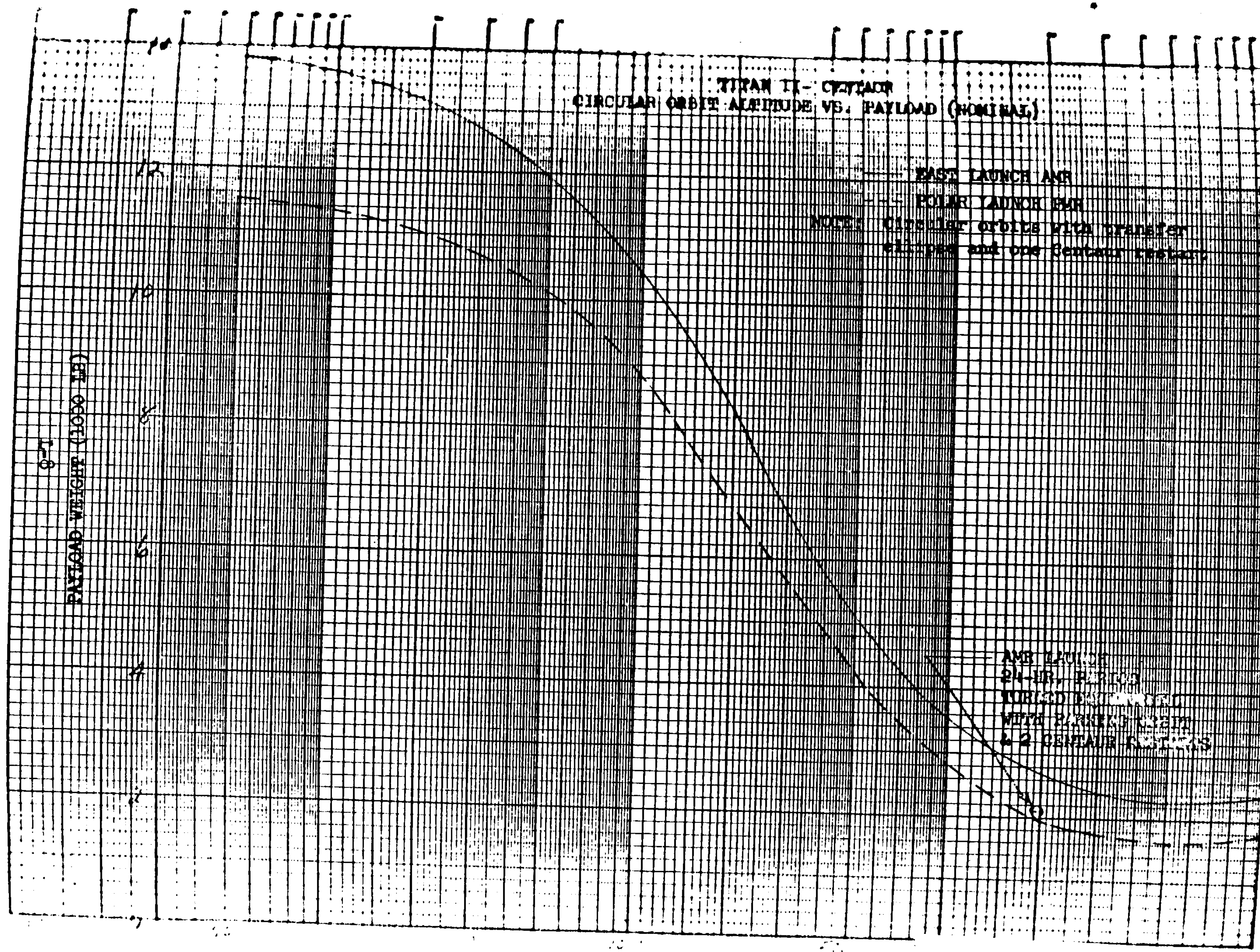
TABLE IV
NOMINAL PERFORMANCE CAPABILITIES

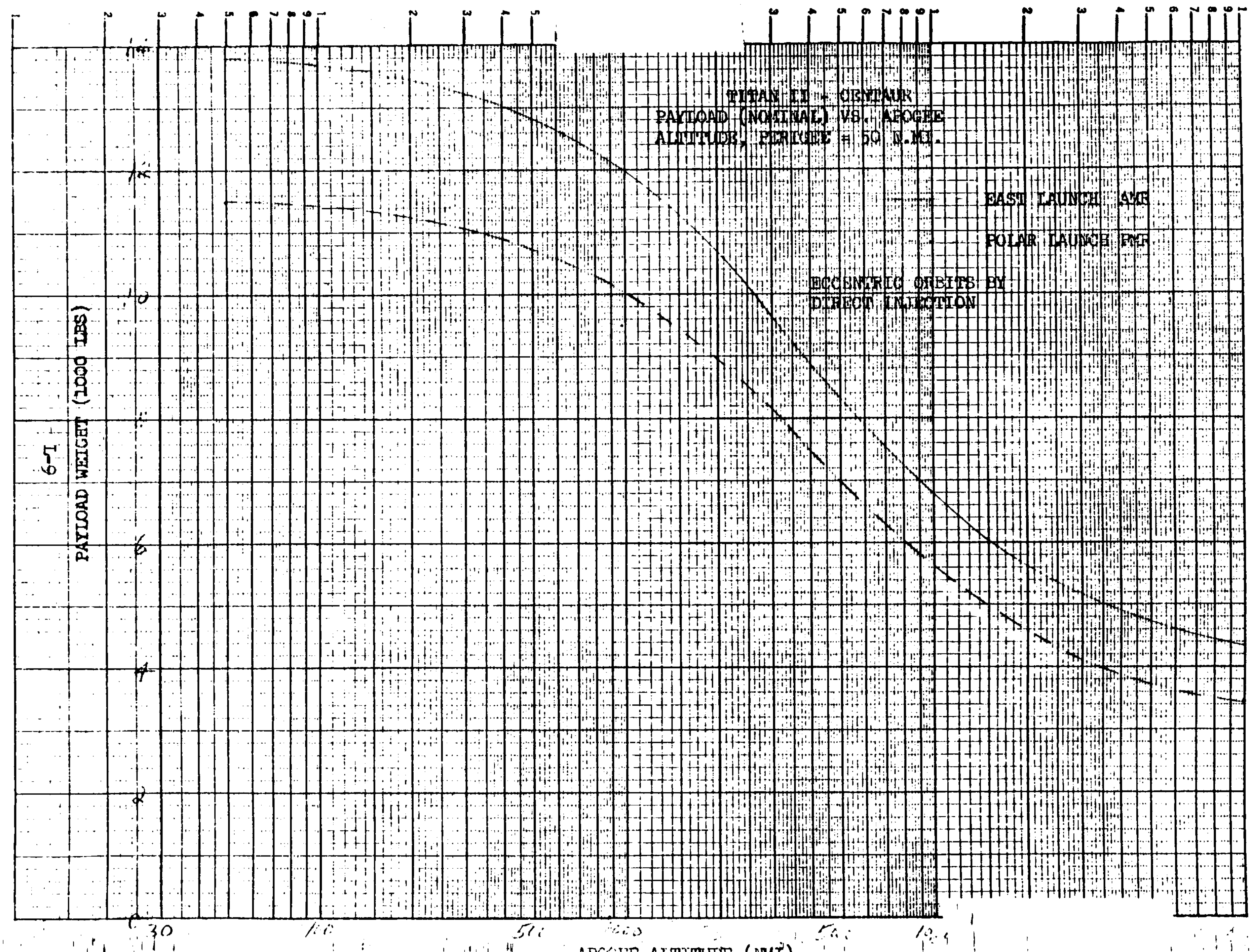
MISSION	PAYLOAD WEIGHT (LB) EAST LAUNCH FROM AMR TITAN A CENTAUR	TITAN B CENTAUR	POLAR LAUNCH FROM PMR TITAN A CENTAUR	TITAN B CENTAUR
50 n mi Circular Orbit by Direct Injection	8825	13,775	7255	11,475
300 n mi Circular Orbit by Coasting Injection	7585	12,685	6665	10,655
2000 n mi Circular Orbit by Coasting Injection	4965	8,195	3945	6,825
24 hour Circular Orbit Turned Equatorial with Parking Orbit and Two Centaur Re-Starts	620	2,105	--	--
Escape from Earth by Direct Injection	1935	3,975	1325	3,125
Minimum Energy Venus Probe by Direct Injection	1565	3,425	935	2,655
Minimum Energy Mars Probe by Direct Injection	1425	3,225	865	2,485
Minimum Energy Lunar Probe by Direct Injection	2075	4,155	1435	3,305



TITAN I - CENTAUR
CIRCULAR ORBIT ALTITUDE VS. PAYLOAD (NOMINAL)
EAST LAUNCH AMR
POLAR LAUNCH PMR
NOTE: Circular orbits with transfer ellipse and one Centaur restart
AMR LAUNCH
24-HR. PERIOD
TURNED EQUATORIAL
WITH PARKING ORBIT
& 2 CENTAUR RESTARTS
PAYLOAD WEIGHT (1000 LB)
CIRCULAR ORBIT ALTITUDE (N.MI)
10
8
6
4
2
0
50
100
500
1000
5000
10,000
50,000
KEUFFEL & ESSER CO.
4 CYCLES X 70 DIVISIONS


TITAN I - CENTAUR
PAYLOAD (NOMINAL) VS. APOGEE
ALTITUDE, PERIGEE = 50 N.MI.
EAST LAUNCH AMR
POLAR LAUNCH PMR
ECCENTRIC ORBITS BY
DIRECT INJECTION
PAYLOAD WEIGHT (1000 LBS)
APOGEE ALTITUDE (NMI)
L-7
KEUFFEL & ESSER CO. MADE IN U.S.A.
4 CYCLES X 70 DIVISIONS


TITAN II - CENTAUR
CIRCULAR ORBIT ALTITUDE VS. PAYLOAD (NOMINAL)
EAST LAUNCH AMR
POLAR LAUNCH PMR
NOTE: Circular orbits with transfer
AMR LAUNCH
& 2 CENTAUR
PAYLOAD WEIGHT (1000 LB)
L-8
14
12
10
8
6
4
2
0


TITAN II - CENTAUR
PAYLOAD (NOMINAL) VS. APOGEE
ALTITUDE, PERIGEE = 50 N.MI.
EAST LAUNCH AMR
POLAR LAUNCH PMR
ECCENTRIC ORBITS BY
DIRECT INJECTION
PAYLOAD WEIGHT (1000 LBS)
APOGEE ALTITUDE (NMI)

L-9