(via uplink from MSC) such that the $X$-accelerometer was parallel to $\hat{g}$ for approximately 10 min. The process was then repeated with the $X$-accelerometer antiparallel to $\hat{g}$.

In position one we have [by Eq. (1)]

$$-g = g_X^{(1)} = [1 + SFE_X)(\Delta N_X^{(1)}/\Delta T^{(1)}] - B_X \quad (4)$$

where $\Delta N_X^{(1)} < 0$. For position two we have

$$g = g_X^{(2)} = [1 + SFE_X)(\Delta N_X^{(2)}/\Delta T^{(2)}] - B_x \quad (5)$$

with $\Delta N_X^{(2)} > 0$. Equations (4) and (5) can be combined to yield

$$g = \tfrac{1}{2}[\Delta N_X^{(2)}/\Delta T^{(2)} - \Delta N_X^{(1)}/\Delta T^{(1)}](1 + SFE_X) \quad (6)$$

Equation (6) was used to calculate $g$ for this experiment.

PIPA data and the associated times were sent to earth via telemetry. Subsequent reduction of the stored data yielded the raw data for the determination of $g$.

### Data, Results and Analysis

The Lunar Surface Alignment Program gave for $\beta$ and $\gamma$ the values of $-0.12183$ and $-0.33971$, respectively. From Eqs. (2) and (3) these values gave for the gimbal angles: $OGA_1 = OGA_2 = 0.00°$; $IGA_1 = +178.05°$; $IGA_2 = -1.95°$; $MGA_1 = MGA_2 = +7.00°$. These angles were passed to the Guidance Officer in Mission Control Center whose first command sequence placed the $X$-PIPA input axis in the down position within 0.016° of the desired position, resulting in a measurement of $g_X^{(1)} = -161.55$ cm/sec$^2$. Then the $X$-PIPA input axis was commanded to the up position with a final alignment accuracy of 0.012°, resulting in a measurement of $g_X^{(2)} = +164.06$ cm/sec$^2$. Compensating for the misalignment of the $X$-PIPA from the known direction of $g$ did not effect the final value since the correction was less than 0.01 cm/sec$^2$. Using Eq. (6) and a value for $SFE$ of $-945$ ppm, which was based on a linear regression fit of the preflight measurements, the value for $g$ was determined to be $g = 162.65$ cm/sec$^2$.

The alignment accuracies quoted previously were determined from the gimbal angles and the known orientation of the NB with respect to $\hat{g}$.

### Comparison with Models

Prior to the Apollo 14 mission, the value of $g$ for the Fra Mauro landing site was calculated using four lunar potential models,[2] resulting in a mean value of 162.57 cm/sec$^2$ with an uncertainty of 0.04 cm/sec$^2$.

The predicted values of $g$ at the landing site assumed a radius of 937.735 naut miles. This was arrived at from stereo photography and landmark tracking data obtained from previous lunar missions. From a series of experiments performed during Apollo 14, the radius of the landing site was later determined[2] to be 937.577 naut miles. This value changes the mean value of acceleration for the lunar potential models to 162.63 cm/sec$^2$.

The uncertainty associated with the gravity measurement is 0.02 cm/sec$^2$; hence, within experimental error, the measured value of $g$ ($g = 162.65 \pm 0.02$ cm/sec$^2$) equals the mean value of $g$, $\langle g_m \rangle$, calculated from the four potential models ($\langle g_m \rangle = 162.63 \pm 0.04$ cm/sec$^2$) employing the corrected value for the radius.

### Conclusion

The measured value of $g$ is consistent with the value of $g$ observed among four lunar potential models while the uncertainty in the measured value of $g$ is less than that of the models.

### References

[1] Miller, J. E. and Laats, A., "Apollo Guidance and Control System Flight Experience," *Journal of Spacecraft and Rockets*, Vol. 7, No. 5, May 1970, pp. 551–557.

[2] Savely, R. T., "Preliminary Apollo 14 LM Radius Analysis," MSC Internal Note 71-FM-91. (N.B. This preliminary report is also the final report.)

# Impulse Trajectory Correction by the "Cassiopee" System

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## Introduction

In PREPARATION for an astronomical observation program using sounding rockets, an attitude control device named Cassiopee[1] has been developed in France over the past few years. It is usually used to set the nose cones, previously separated from the rockets, on the celestial targets studied in a predetermined sequence.

Most of the experiments planned within the framework of this program include some photography, thus making it essential to retrieve the scientific payloads. This in turn entails a special requirement for the nose cones to make an accurate splashdown—a demand all the more pressing because the Guyana Space Center (CSG), the site chosen for the experimental program, is a seaside launching base, and successful recovery hinges largely on quick access to the floating payload.

With no special provisions made (apart from initiation of rocket spin, and careful correction for wind), the standard deviation on the impact of the two-stage rocket Beridan which is capable of carrying 400 kg to an altitude of 340 km, as called for under the observation program, is 39 km in any one direction. This suggests the need to aim at a nominal impact point 100 km off the coast so that the probability of having to destroy the rocket by remote control in order to protect the mainland should not exceed 0.5%. The distance of the coast from the planned splashdown and the dispersion provide the data that determine the operational methods and facilities to be used for locating and recovering the payload.

One way of improving accuracy would be to equip the rocket with an auto-pilot as a means of reducing dispersion, thereby enabling the nominal impact point to be brought nearer the coast. Without the use of costly high-precision gyroscopes, there is little chance of the standard deviation

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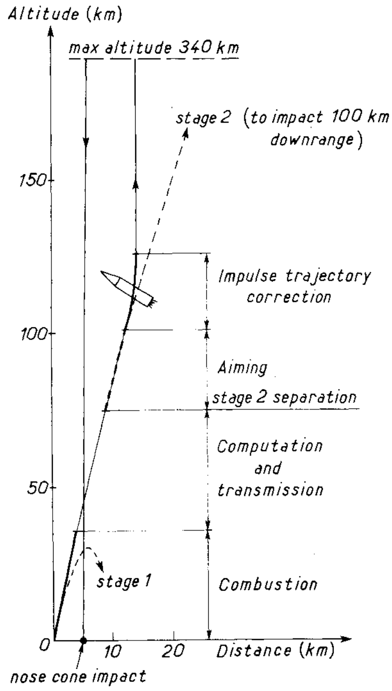


Fig. 1 Mission profile.

on the impact area of the Beridan rocket being kept below 12 km.

Another solution, which seems both more efficient and relatively less expensive in the context of this project, is to fire the rocket in the normal way, and then impart to the nose cone, after separation of the second stage, a corrective impulse delivered by an auxiliary propulsion unit mounted in the nose cone.

This is operated with the aid of a simple radio guidance system relying: 1) on the ground, upon the CSG real-time trajectory-measuring and computing facilities, which give the correction to be applied after completed combustion in the two main stages; and 2) aboard the missile, on the Cassiopee system, which sets the nose cone in the impulse direction assigned by remote control. This makes it possible both to compensate for the deviation of the observed trajectory and to choose for the nose cone a nominal impact point different from that of the rocket and closer to the coast.

## Mission Profile

Summarizing, a firing comprises the following phases (Fig. 1). 1) The rocket launched in the usual way, with the wind corrected to provide for a splashdown at the intended point preselected to lie 100 km off the coast. During combustion in the two phases, the rocket is tracked by the Bretagne radar, and the data so furnished are given real-time processing by the CSG computer to obtain an optimum estimate of position and velocity at any given instant. 2) At the end of the second-stage propulsion phase at an altitude of 36 km (beyond which perturbations are negligible) the computer determines the direction of the corrective impulse by means of a suitable algorithm from the ballistic condition of the rocket at that instant, bringing the nose-cone spashdown to the point chosen for recovery 5 km off the coast. The two angular coordinates of this direction are transmitted by radio in binary form, and put in storage inside the nose cone. 3) At a 75 km altitude, the nose cone separates from the second stage and the Cassiopee device sets it in the preassigned direction, using the airborne gyroscope platform for reference. Aiming is achieved, with the vehicle spinning by means of suitable switched nitrogen jets at the nose-cone rear. 4) At a predetermined instant (78 sec), the corrective propulsion unit ignites with the continuing spin of the nose cone maintained in, by and large, the desired direction by nonstop Cassiopee action. Combustion in the unit goes on for 15 sec until, with the solid propellant spent, correction has been achieved. 5) The corrective propulsion unit falls away, and the operation of point aiming at the first celestial target, as defined by "zero" direction on the gyroscope stage, is carried out to a high degree of accuracy by the use of stellar sensors. Further objects may also be examined in turn. 6) Upon re-entry into the atmosphere the nose cone, parabraked, splashes down and kept afloat by a self-inflating buoy, sends out radiolocation signals.

## Corrective Capability and Loss of Performance

Prior to correction the uncertainty factor of the rocket trajectory can be described in first approximation by the difference, $\delta\mathbf{V}_3$, between actual velocity $\mathbf{V}_3$ and the rated velocity $\mathbf{V}_{3N}$ at a certain reference time $t_3$ subsequent to burn out in stage 2. The $\delta\mathbf{V}_3$ is a random vector of zero central value; its two horizontal components, $\delta V_{x_3}$ and $\delta V_{y_3}$, result mainly from the angular trajectory deviation and have the same standard deviation (82 m/sec). The component $\delta V_{z_3}$ primarily derives from an impulse difference in stages 1 and 2 and shows a lower standard deviation (30m/sec).

### *Corrective capability*

With the above approximation in use, it is sufficient to consider horizontal-velocity perturbations only. Those trajectories capable of full correction are the one where the horizontal projection $V_{H_3}$ of the end of $\mathbf{V}_3$ lies inside the circle of $\Delta V$ radius centered on $V_{H_3}'$.

The probability of a trajectory being fully corrected accordingly depends on the correction capability $\Delta V$ and the distance $D$ between the nominal rocket impact point and the impact point chosen for nose-cone recovery. To approach the nose-cone splashdown to within 5 km of the coast with a 99.9% probability, while launching the rocket under normal safety conditions, requires a corrective capability of 480 m/sec, which in turn calls for the use of a propulsion unit containing 100 kg of propellant. The requisite corrective capability would be less (300 m/sec) if the nose-cone impact point aimed at were to coincide with the nominal impact point of the rocket.

### *Loss of performance*

At a given vertical velocity $V_{z_3}$, any correction other than full-scale is attended with a gain in vertical velocity. It is not, however, a "something-for-nothing" gain, for the weight of the corrective propulsion unit detracts from the initial vertical velocity $V_{z_3}$. Again, the corrective operation takes a period of 27 sec to complete, which is set apart at the expense of useful observation time. This points up the importance of evaluating the system in terms of loss of performance as well, the performance rating being in terms of the flight time available after correction.§ It appears that

**Table I Deviations due to elementary errors**

| Trouble sources | Impact errors, km |
|---|---|
| 1) Estimate of initial state (3m/sec) | 1.4 |
| 2) Cassiopee gyroscopic pointing (0.5°) | 1.6 |
| 3) Cassiopee jets (1m/sec) | 0.5 |
| 4) Propulsion-unit alignment error (10′) | 0.8 |
| 5) Propulsion-unit impulse (1.5%) | 1.9 |
| 6) Wind gaging (2m/sec) | 0.6 |
| 7) Inaccuracy of linear algorithm | 0.2 |
| Resulting deviation | 3.1 |

§ The limit to this time is given by the descent to 100 km altitude, below which observations are considered nonsignificant.

the result is increasingly favorable with an increasing size of auxiliary propulsion unit. At the same time the greater inertia and the rearward displacement of the center of gravity due to the weight of the propulsion unit located at the nose-cone rear do not make the working of the Cassiopee system any easier. This sets the necessary restriction of using a propulsion unit of 100 kg propellant capacity, which enables the loss in the useful flying time of approximately 450 sec to be kept down below 15 sec in 50% and 45 sec in 95% of cases.

***Control efficiency in the propulsion phase***

In the propulsion phase, the nose cone is perturbed by the misalignment of the propulsive jet. For the projected propulsion unit (thrust 16,500 N), the standard deviation on this unwanted moment is three times as high as the Cassiopee control moment. All the same, the average direction of the nose cone hardly deviates from the assigned direction and with the cone made to spin at a speed of over 0.5 rps the disturbance in lateral velocity stays below 4m/sec. Nitrogen consumption in the propulsion phase is 0.750 kg.

### Accuracy of Impact

Impulse trajectory correction following combustion in the two main rocket stages obviates previous errors, but introduces new, though fortunately slight, ones. The residual scatter of impact results from a combination of them; they are summarized in Table 1, together with their standard deviations and respective effects, for the projected propulsion unit containing 100 kg propellant and having a correction capability of 480 m/sec.

The effects of 2, 4, 5, and 7 on the impact point depend on the correction to be applied, and are computed in mean square value terms for all trajectories capable of correction.

The uncertainty factor in the impulse of the auxiliary propulsion unit has a more important effect than the rough-aiming errors, 2 and those of attitude control during the propulsion-unit combustion phase 4. Accordingly, the Cassiopee system appears to measure up in accuracy (0.5°) to the new mission expected of it.

Errors 2, 4, and 5, which are bound up with the corrective capability of the propulsion unit, could be lessened by the use of a propulsion unit with a corrective capacity of 380 $ms^{-1}$ only. The resultant deviation of the correctible trajectories could be brought down to 2.6 km, but the penalty to pay for this gain would be more trajectories unamenable to correction (5% instead of 0.1%).

The use of a linear corrective algorithm is justified by the slight error so introduced.

### Conclusions

An accurate correction of the trajectories of Beridan sounding rockets after completed combustion in the two propulsive states has been made possible by the Cassiopee system installed aboard observation nose cones and by the use of CSG equipment.

The result is a reduction of impact scatter to 3.1 km (instead of 39 km as before), with the recovery area brought within 5 km of the coast (instead of a former 100 km).

To achieve this improvement, the base of the nose cone (of 400 kg initial weight) must be equipped with a corrective propulsion unit containing 100 kg propellant, and be able to impart to the cone a velocity variation of 480m/sec. This enables practically all (99.9%) of the deviating trajectories to be corrected at no great sacrifice of time available for subsequent scientific observation.

### References

[1] Remondière A., "Dispositif de controle d'attitude Cassiopee—Premiers essais en vol," *XXth International Astronautical Congress*, TP ONERA 762, Mar del Plata, Oct. 5–11, 1969.

# Churchill Research Range Auroral Launcher Facility

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### Background

OUT of a total of 630 scientific rockets launched at the Churchill Research Range since its inception in 1957, 350 were fired under subzero temperature conditions. The most favored time for rocket launching at Churchill is during the winter months when clear skies and long nights prevail. The existing subarctic conditions in northern Manitoba provide an excellent opportunity for the study of geophysical events such as the aurora borealis, but they also create deplorable working conditions. This situation requires continued efforts to improve the environmental control systems of the launcher facilities. When a new launcher was constructed in 1968, it incorporated design features based on considerable experience in operating in the north. This facility, which was designated the "Auroral Launcher," has proven to be the most favored of all launchers in use at Churchill because of the superiority of its temperature control system.

### Design Requirements

The basic requirement was to provide reasonably comfortable conditions for the launch crew during loading operations and the maintenance of the rocket at a satisfactory temperature during prolonged periods when it would be held in an upright position in readiness for launching at the onset of a desired geophysical event. In the earlier facilities, such as the Universal Launcher, these requirements were met by enclosing the launcher in its lowered position (Fig. 1) in a building with large roof doors. When the roof doors were open and the rocket elevated, temperature control for the rocket was provided by forcing heated air through clam-shell heat shields which encased the vehicle.

This arrangement had several serious drawbacks. The large roof doors were of necessity slow acting and, as a con-

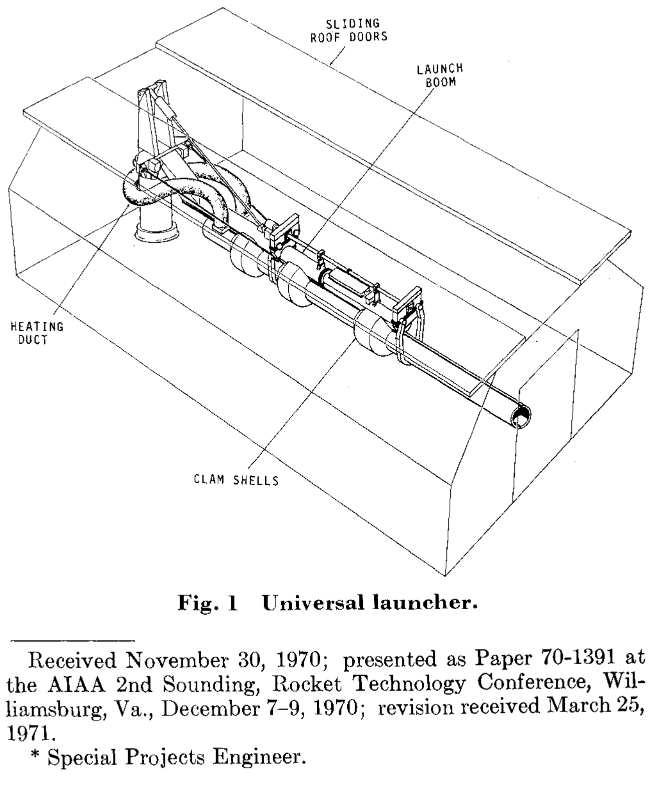


**Fig. 1 Universal launcher.**

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